Technical Memorandum 4075

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This publication is one of four documents describing work performed in fiscal year 1988 under the auspices of the newly formed Office of Exploration. The first in the series, titled, "Beyond Earth's Boundaries . . . Human Exploration of the Solar System in the 21st Century" provides an overall programmatic view of the goals, opportunities, and challenges of achieving a national goal for human exploration. The technical details and analyses are described in a three-volume set titled: "Office of Exploration: Exploration Studies Technical Report (FY 1988 Status)." Volume I is a Technical Summary; Volume II is the Study Approach and Results; and Volume III is a collection of trade study results, indepth systems assessments, and workshop reports which describe aspects of FY 1988 analyses in more depth.

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NASA

VOLUME III: SPECIAL REPORTS, STUDIES AND INDEPTH SYSTEMS ASSESSMENTS

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VOLUME III: SPECIAL REPORTS, STUDIES AND INDEPTH SYSTEMS ASSESSMENTS

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1988 OEXP Annual Report Draft Transmittal

Lunar Node for Mars Evolutionary Expansion Missions

July 8, 1988

by

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Contract NAS1-18267 Task 14

LUNAR NODE OPTION

1.0 Introduction

The Lunar Node Option refers to the utilization of a lunar orbiting vehicle to support a sequence of manned expeditions to Mars and the moons of Mars in the 2010-2025 time period. This mission set has been referred to as the Mars Evolutionary Expansion scenario. The principal operational function of the Lunar Node is to provide a storage and transfer point for lunar derived liquid oxygen (LLOX). An additional function of the Lunar Node is to provide temporary storage of cargo such as crew, crew supplies, hydrogen propellant, and other material for subsequent delivery to the lunar surface outpost. In the event of a contingency, the Lunar Node would also provide a life support habitat for stranded lunar or Orbital Transfer Vehicle crews pending rescue from LEO. The specific mission mode studied was that of a Lunar Node operating in low lunar orbit (LLO) in the era beyond 2011, after the establishment of a lunar based LLOX production capability. The LLOX would be transported from the Lunar Node to LEO where on-orbit assembly and transfer of propellant to the Mars bound vehicles would take place. Follow-on studies may indicate alternate locations, such as libration points, for transfer of LLOX to the Mars vehicles, but the Lunar Node would provide the same functions.

2.0 Requirements and Assumptions

This conceptual study of a Lunar Node was principally guided by the requirements given in the Scenario Requirements Document (SRD), NASA Document No. Z-MAS-SRD Draft 5/12/88 (Reference 1). Specifically, the requirements for the Lunar Node for support of the Evolutionary Expansion Missions were applied. The SRD version issued on June 2, 1988 (Reference 2) eliminated the requirement for Lunar Node to be included in the 1988 Office of Exploration Annual Report. However, the concept presented in the following sections describes a node which could support any of several mission modes for Mars exploration using lunar derived liquid oxygen and a lunar outpost.

The requirements and related assumptions applied to the Lunar Node for Evolutionary Expansion Missions of Reference 1 are presented in Figures 2-1 through 2-3. The development of the propellant logistics schedule required a combination of related data from the SRD Draft (Reference 1) and some assumptions, since Appendix F of Reference 1 containing the Lunar propellant logistics schedule was unavailable at the time that this evaluation was conducted.

The baseline concept is configured to supply LLOX for a manned Mars expedition program based upon; (1) the use of cryogenic chemical propulsion for all velocity changes except for aerobraking at Earth and Mars, and (2) the assembly and propellant loading of the Mars-bound vehicles at the LEO node. Transfer of LLOX to Mars-bound vehicles directly from the Lunar Node or from an Earth-Moon libration point would cause a revision to the propellant logistics

LUNAR NODE: EVOLUTIONARY EXPANSION

<u>Study Guidelines Sources</u>

- Baseline Lunar Node to meet SRD (May 12, 1988) draft input
  - LLOX logistics not available (Appendix F)
- SRD (June 2, 1988) eliminates Lunar Node
  - LLOX logistics (Appendix F) per new scenario
  - Implicit application of a Lunar Node by functions described
  - Lunar Node equivalent to be defined

Figure 2-1 Study Guidelines Sources

6-13-88 TBC #2

LUNAR NODE: EVOLUTIONARY EXPANSION

Requirements from Scenario Requirements Document (SRD)

Source SRD 2.4

- Lunar Node (LN) supports 12 people on Lunar Surface (LS)
- 6 months stay time per person on LS
- Lunar tanker available for LLOX from LS to LN
- LN to support tank farm in Low Lunar Orbit (LLO)

Source SRD 6.2.1

- LN in 100 NM circular polar orbit
- LN to be man-tended, 2 crew members

Figure 2-2 Study Requirements

6-13-88 TBC #3

LUNAR NODE: EVOLUTIONARY EXPANSION

<u>Requirements from SRD</u> (con't)

SRD Operations Guidelines

- Safe Haven
  - Minimum of 2 habitable elements
  - EVA support capabilities
- Node Capability
  - Maintain trajectory altitude, attitude
  - Maintain general systems health
  - Minimum of 2 earth resupply cycles
  - On board navigation capability to determine inertial state and that of approaching S/C

Figure 2-2 Study Requirements (continued)

6-13-88 TBC #4

LUNAR NODE: EVOLUTIONARY EXPANSION

Requirements from SRD (con't)

SRD Operation Guidelines

- Communications links to:
  - TDRSS
  - OTV
  - LS
  - LEO Node

Figure 2-2 Study Requirements (concluded)

6-13-88 TBC #5

LUNAR NODE: EVOLUTIONARY EXPANSION

Fuel Requirements and Assumptions

Source SRD 2.4

- LLOX to be delivered to LEO (do not consider other locations such as L2 for FY88 study)

Study Assumptions:

- Do not consider boil-off on first cut
- Assume LLOX production infrastructure (i.e. after the first 10 years of evolutionary expansion)
- All propulsion by H_2/O_2 at ISP = 470 sec and 7:1 mixture
- Aero braking for OTV LLO to LEO (non-propulsive)

Figure 2-3 Propellant Requirements and Assumptions

6-13-88 TBC #6

LUNAR NODE: EVOLUTIONARY EXPANSION

<u>Fuel Requirements and Assumptions</u> (con't)

- 6 OTV Flights Per Year to LN from LEO
- 18 Lunar Ascents to LN Per Year from LS
- 20 Metric Tons Per Year Cargo Mass to LS (3:1) *
- 25 Metric Tons Per Year Hydrogen at LLO from LEO (SRD) *
  - For Use with LLOX Exclusively
  - Provides Propulsion of Cargo and Tanks from LLO to LS
  - Provides Lift of LLOX to LLO
  - Provides LLOX Transfer from LLO to Entry at LEO
- Equates to Approximate LLOX Annual Production Rate of 800 Metric Tons without Losses

* Derived from SRD (5/12/88) Draft Appendix Bar Charts

Figure 2-3 Propellant Requirements and Assumptions (continued)

6-14-88 TBC #6A

LUNAR NODE: EVOLUTIONARY EXPANSION

Fuel Requirements and Assumptions (con't)

- LLOX

Start from LLO with	100.0 MT
Arrive LEO with	81.0 MT
Reserve for next flight	19.9 MT

Available at LEO	61.1 MT

- LH_2

Leave LEO with	9.6 MT

Figure 2-3 Propellant Requirements and Assumptions (concluded)

6-13-88 TBC #7

manifest developed in Section 3. Application of a Mars node and/or Mars propellant production would also impact the Lunar Node LLOX logistics.

3.0 Propellant Logistics

The 25 year program plan for manned Mars exploration includes an initial ten to 12 year period with total dependence upon Earth provided liquid oxygen and liquid hydrogen for all propulsion functions. This early era is also the period for assembly of the infrastructure including; the Lunar Node, the lunar outpost, and the lunar oxygen production facility. Once these facilities are in place, the Lunar Node becomes operational in its role as a propellant transfer facility. It is this later period that defines the functional requirements of the Lunar Node.

The propellant logistics must consider both the liquid hydrogen fuel provided from Earth via the LEO node and the lunar derived oxygen returned to LEO. At each step, or stage, of the transport some of each propellant is consumed in the process.

The functions of the Lunar Node provide the basis for defining the propellant logistics. The determination of the masses which will be exchanged involves certain assumptions relative to the space vehicles to be serviced, the number or frequency of flights, and the velocity increments associated with each phase of the spacecraft flight profile. These parameters and assumptions are summarized in Table 3-1. The mass transfers for propellant and cargo together with the number of flights for the Orbital Transfer Vehicle (OTV) represent values listed in the SRD's (References 1 and 2; see Figure 2-2). The general configuration for the OTV depicted as a tank cluster using aerobraking for return to LEO follows from a previous study (Reference 3). The mass of the OTV system represents an estimated fraction of the maximum payload transported. The selection of a tank size was a result of the mass transfer requirement study. The propulsion capability provided by 25 MT of hydrogen at LLO combined with the ascent-descent orbit insertion-landing velocities associated with a 100 km lunar orbit altitude define the Lunar Ascent-Descent Vehicle (LADV) design parameters once the number of supply flights for each OTV trip is selected. The propulsion systems were assumed to operate with a seven to one mass ratio for O_2/H_2 and a specific impulse of 4310 N-sec/kg. The velocity increments for each phase of a transfer were extracted from orbital calculations, and the velocity changes used the following criteria.

a. Acceleration due to thrusting does not exceed 0.9 g (8.62 m/sec^2).

b. Propellant masses were calculated from a simplified rocket equation.

c. The lunar decent-ascent phase was limited to 103 second burns, and assumed a constant mass for levitation against lunar gravity.

The criteria for defining the velocity changes are considered conservative, in that higher thrust levels with correspondingly shorter burn times would consume less total fuel. On the other hand, the thrust levels selected maintain the launch of an H_2 tank from Earth as the worst case load condition. The analysis to define the masses transferred calculated the propellant requirements for each phase of an OTV transfer flight from LEO to/from lunar orbit and LADV flights from lunar orbit to/from the lunar surface. Table 3-1 summarizes the propellant and cargo transfer logistics in terms of quantities and numbers of tanks. The values shown in Table 3-1 apply the simplifying assumption that exchanges with the Lunar Node involve only the transfer of containers as tanks of cryogenic liquids and cargo in packages. The OTV and LADV will draw propellant from the tanks transferred, but exchanges with the Lunar Node do not involve the transfer of cryogenic liquids between tanks in a microgravity environment. The definition of the OTV propellant transfer first established the hydrogen constituent necessary to leave the lunar gravity field. This mass increment became part of the delivery to LLO. The propellant requirements for the OTV braking to node rendezvous and OTV escape from the Earth's gravity field established the hydrogen requirement at departure from LEO plus the amount of lunar oxygen that would have to be retained on-board the OTV. The flight sequence for the OTV is summarized in Figure 3-1 and shows the usage of on-board propellant for each of the burns.

The definition for the LADV sequence assumed three flights in a 55 day interval as realistically achievable. Figure 3-2 illustrates the sequence. The transfer of 33.3 MT of O_2 per flight became the principal parameter in defining the LADV system. The hydrogen exchanged must serve to land the LADV with the empty tanks necessary for oxygen fill, and provide sufficient fuel to lift the next flight. The masses and volumes that resulted indicated that a common tank size of 10 m^3 capacity would serve the requirements. This tank has a diameter of 2.67 m and is of a size compatible with aluminum construction. The mass estimate suggests that the full

TABLE 3-1 PROPELLANT TRANSFER LOGISTICS - SUMMARY OF PARAMETERS

Fuel and Cargo Transfer

1. Lunar Supplied Oxygen (LLOX): Liquid O_2, 600 MT/yr to LLO (90 K, 1140 kg/m^3)
2. Earth Supplied Hydrogen (LH_2): Liquid H_2, 25 MT/yr to LLO, (20 K, 70 kg/m^3)
3. Cargo to Lunar Base: Packaged, 20 MT/yr to lunar surface

Orbital Transfer Vehicle (6 flights/yr)

Transport: Propellant in 16 tanks of 10m^3 capacity (all tanks identical)

Thrust: O_2-H_2 engines supplied from on-board tankage

Mass and Configuration: 30 MT total, structure includes aeroshell for LEO deceleration

Lunar Ascent-Descent Vehicle (18 flights/yr)

Transport: Propellant in six tanks of 10 m^3 capacity

Thrust: O_2-H_2 engines supplied from on-board tankage

Mass and Configuration: 4.8 MT total; open frame structure with lunar landing pads

Velocity Change Requirements

LEO to lunar gravity equilibrium	2.67 km/sec
LLO to Earth gravity equilibrium	0.667 km/sec
LLO to lunar surface	1.62 km/sec
Levitation in lunar gravity requires	1578 N/MT

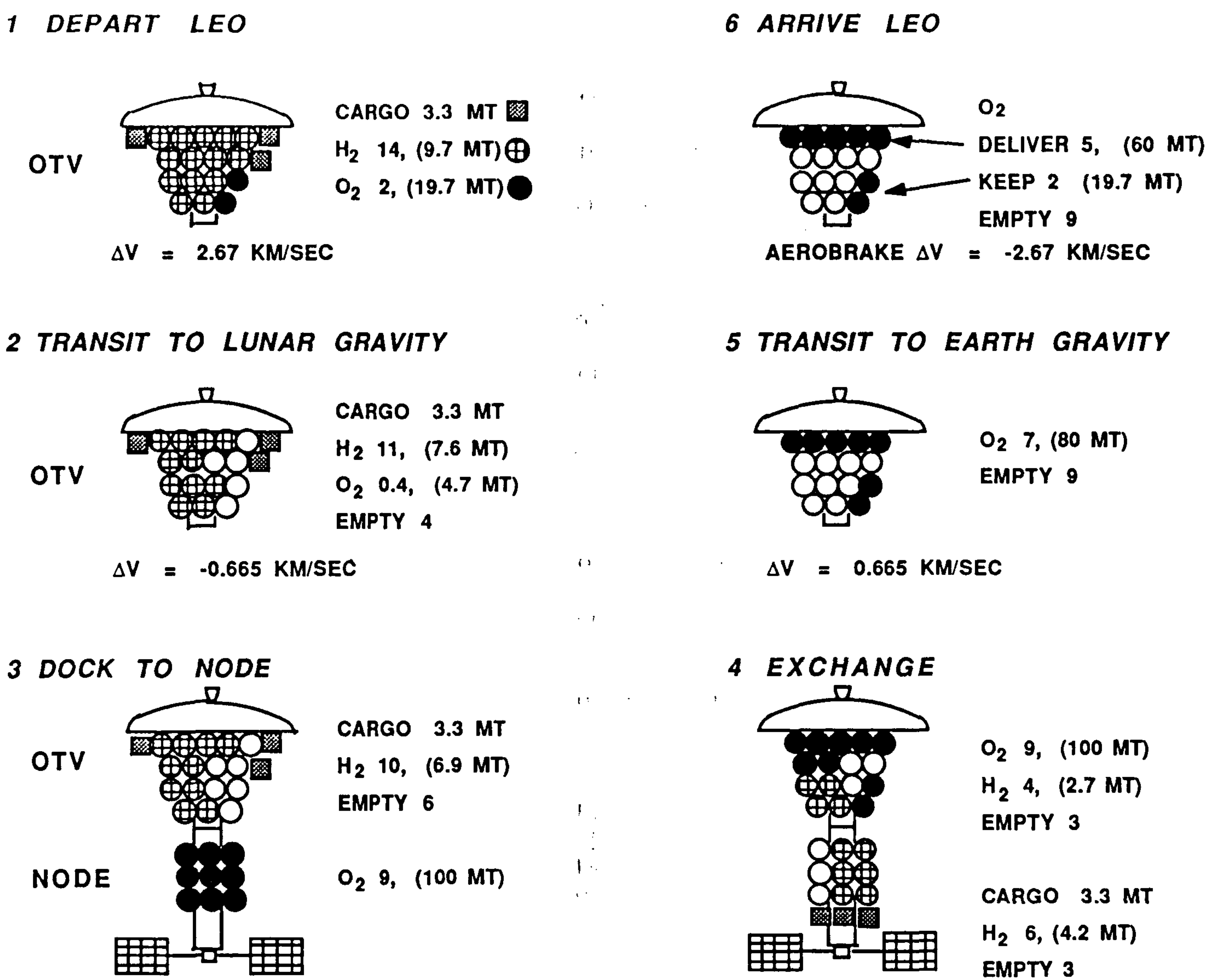

Figure 3-1 Flight Sequence for the OTV Showing Propellant Usage and Transfer Logistics

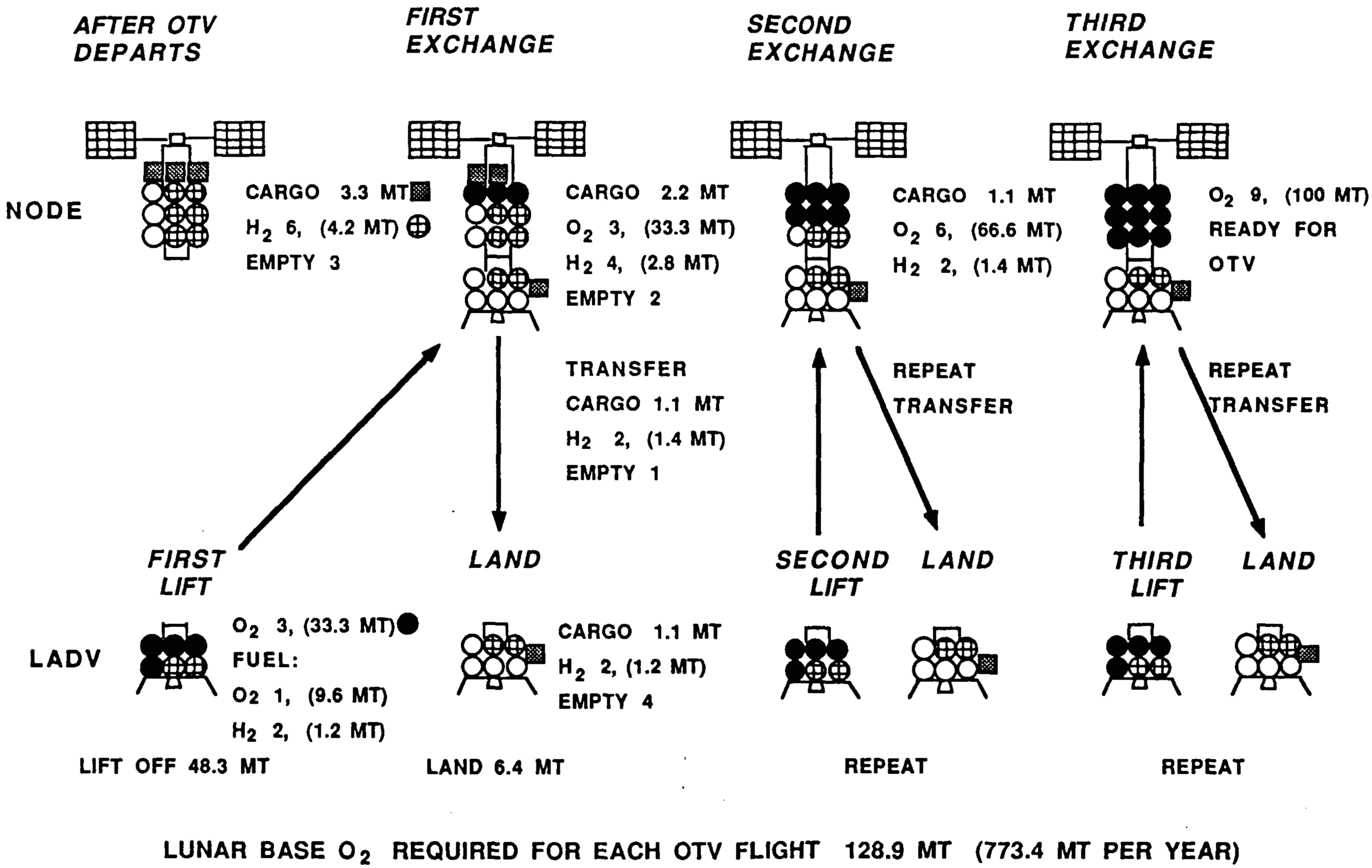

Figure 3-2 Flight Sequence for the LADV Showing Propellant Usage and Exchange Logistics

H_2 tank on Earth represents a handling weight of about 1.2 MT, while the same tank filled with O_2 on the lunar surface represents a handling weight of about 1.9 MT. Both values appear reasonable for local handing, storage and transport. The definition of the fuel transfer masses did not include boil-off losses. The actual OTV and LADV flight would be relatively short (i.e., days) and transit losses correspondingly small. Cryo-retention capability as reliquification is considered as part of the operation availability at LEO, and lunar base. A cryo-retention capability has been included for the Lunar Node.

The summary of propellant transfer logistics as shown in Table 3-2 and indicated by Figures 3-1 and 3-2, suggest that about 60 percent of the O_2 delivered to lunar orbit will become available at LEO and about 75 percent of the lunar surface O_2 production can be delivered to LLO. The combined effect is to deliver approximately 47 percent of the LLOX production to LEO for the Mars mission. The H_2 usage shows that 25 percent of the total is required to transfer the O_2 from LLO to LEO with an aerobraked OTV and about 40 percent of the H_2 is used to lift the O_2 from the lunar surface to LLO.

TABLE 3-2 PROPELLANT TRANSFER LOGISTICS, FLIGHT SEQUENCES FOR CONSTITUENTS, CARGO AND TANKAGE

	CONSTITUENTS AND CARGO TRANSFER					
OTV SEQUENCE (6 per year)	O_2 (MT)	Active Tanks	H_2 (MT)	Active	Cargo Tanks	Empty (MT) Tanks
1. Depart LEO	19.7	2	9.7	14	3.3	0
2. Post 2.67 km/sec Burn	4.7	1	7.6	11	3.3	4
3. Lunar Orbit Rendezvous	0	0	6.9	10	3.3	6
Transfer from Node	+100	+9	-4.2	-6	-3.3	-3
4. Depart LLO	100	9	2.7	4	0	3
5. Post 0.667 km/sec Burn	79	7	0	0	0	9
6. LEO Rendezvous	79	7	0	0	0	9
7. Transfer at LEO	60	5				
LADV SEQUENCE (18 per year, 3 each OTV flight)						
1. Depart Lunar Surface	42.9	4	1.2	2	0	0
2. LLO Rendezvous	34.8	4	0	0	0	2
Transfer to Node	-33.3	-3	+1.4	+2	+1.1	+1
3. Depart Rendezvous	1.5	1	1.4	2	1.1	3
4. Lunar Landing	0	0	1.2	2	1.1	4

LLOX SUMMARY

Lunar Surface Production	778 MT/yr
Lunar Orbit Transfer	600 MT/yr
LEO Delivery	360 MT/yr

LH_2 SUMMARY

Delivery to LEO	58.2 MT/yr
Delivery to LLO	25.0 MT/yr
Delivery to Lunar Surface	21.6 MT/yr

4.0 Crew Safe-Haven Option

One of the SRD (Reference 1) requirements for the Lunar Node is to provide a safe-haven for stranded crew from the lunar base and/or the transportation vehicles until rescue can be effected from the LEO node. The Lunar Node offers a site for the establishment of a lunar crew safe-haven intermediate in location between the lunar surface and the LEO node. There are several contingency events which could require such a temporary emergency shelter capability. The lunar habitat could become inoperative or an orbital transfer vehicle with exchange crews aboard could malfunction at the node and be unable to return to LEO. In the current context, a safe-haven is construed to be a habitat providing air, food, water and minimal creature comforts sufficient to sustain the crew until rescue. Safe-haven as used here does not include protection from solar flares. The safe-haven crew capacity and duration of stay are not explicit in the SRD requirements. For the purpose of sizing the Lunar Node, two bounding conditions were examined. On the low-end of the scale, the crew operational capability is provided by the equivalent of one Space Station Resource Node and one scaled-down Logistics Module. This combined pressurized volume permits; EVA, multiple vehicle docking capability, and life support for a two-person crew during man-tended periods of up to five days. This is referred to as the Man-Tended configuration. At the high-end of the safe-haven capability, an emergency condition for a crew of 14 was assumed for a stay time of 110 days (capacity to skip one visit from the LEO node transfer vehicle). This is referred to as the Safe-Haven configuration, it includes a Space Station Habitat Module as well as the Resource Node. This provides a capacity for 1500 man-days of emergency occupancy. This option incorporates a pressurized volume providing amenities such as exercise equipment, showers, separate sleeping areas, a galley, and other features associated with long duration, large crew missions.

The survival resources of oxygen, food, water, and personal items were estimated from manned flight experience and are presented in Figure 4.1. The life support concept is an open cycle system. For example, carbon-dioxide is absorbed from the air, but not recycled. Water is not recycled, but stored as liquid waste. Clothing, bedding, and food service utensils are used once and then stored as trash. The overall mass and volume quantities of material estimated for the 1500 man-day survival capability is presented in Figure 4.2.

The electrical power capacity for the man-tended option was set at the 20 kw level. This level would sustain the crew and meet operations requirements. For the safe-haven habitat option the electrical power demand was selected to be 35 kW, based largely on the safe-haven requirement. Further evaluation of energy use and energy storage for dark periods is required to arrive at a confident estimate.

LUNAR NODE: EVOLUTIONARY EXPANSION

Assumed Emergency Shelter Consumables
(lb/man day)

Breathing Oxygen	1.84
Carbon Dioxide Removal	2.20
Drinking Water	4.09
Food Water	2.48
Food Solids	1.36
Food Packaging	1.04
Other Water	17.1
Clothing and Personal Hygiene	3.0
Bedding, Towels	2.0

	35.11 lb/day (16 kg/day)

Figure 4-1 Emergency Shelter Consumables

6-14-88 TBC #14

LUNAR NODE: EVOLUTIONARY EXPANSION

1500 Man-Day Survival Capacity (12 Lunar Crew and 2 LN Crew @ 110 days)

ITEM	MASS (kg)	VOLUME (m^3)
Fresh Water	16095	16.0
Fresh Water Tanks	3210	
Waste Water Tanks	3600	18.0
Breathing Oxygen	1260	1.5
Carbon Dioxide Removal	1500	
Oxygen Tank	252	
Carbon Dioxide Removal Tankage	150	1.0

(con't)

Figure 4-2 Emergency Shelter Requirements

6-14-88 TBC #15

LUNAR NODE: EVOLUTIONARY EXPANSION

1500 Man-Day Survival Capacity (12 Lunar Crew and 2 LN Crew @ 110 days) (con't)

ITEM	MASS (kg)	VOLUME (m^3)
Atmospheric Nitrogen	750	1.0
Nitrogen Tanks	68	
Food (Dry) & Packaging	1650	6.0
Dry Goods	3675	15.0
Dry Goods Storage Lockers	368	
Trash & Waste Storage	368	7.5
	-----	----
TOTAL	32946 kg	66.0 m

Figure 4-2 Emergency Shelter Requirements (concluded)

6-14-88 TBC #16

5.0 Lunar Node Description

The two conditions for on-board crew accommodations for the Lunar Node result in two base-line configurations which share a number of common features. Table 5-1 summarizes the principal features and requirements for the node. The concepts for the Lunar Nodes are shown in Figure 5-1 through 5-3. Figure 5-1 and 5-2 show the safe haven node configuration with a large habitat capability docked with the OTV and LADV, respectively. Figure 5-3 is the Man Tended configuration docked with the LADV. All three figures show the concepts for transferring propellant in tanks and cargo in packages. The maximum propellant transfer quantities associated with Mars missions are identified in Reference 2. The principal design factors for the Lunar Node are described in the following, first addressing the mutual items followed by the unique features.

A. <u>Docking Capability</u> The docking adapter is universal to the OTV, LADV, and the Lunar Node, and permits the simultaneous docking of both vehicles to the Node. Figure 5-4 shows such a concept. A Resource Node from the IOC Space Station has been adapted to provide the docking function, in addition to ECLSS support for the crew.

B. <u>Robotic Transfer Unit</u> The transfer unit moves the tanks and cargo between the LADV, Node and OTV. The unit compares to the RMS on the NSTS and would need to position a filled O_2 tank, (mass - 12 MT), at a distance of 15 meters.

C. <u>Communication and Tracking</u> The node communication system links assume a relay satellite in higher orbit. The links would include voice, video, and housekeeping data telemetry.

TABLE 5-1 SUMMARY OF LUNAR NODE REQUIREMENTS AND FEATURES

REQUIREMENTS	FEATURES
1. Lunar Orbit Storage of Cryogenic Liquids • LH_2, 4.2 MT total, 1.4 MT for 55 days max • LLOX 100 MT total, 33.3 MT for 55 day max	Remotely actuated tank attachments for 10 spherical tanks of 10 m^3 capacity. Attachments include provisions for maintaining both O_2 and H_2 as cryogenic liquids.
2. Lunar Orbit Storage of Cargo Pallets 3.33 MT max	Remotely actuated package attachments for three packages of 1.1 MT each.
3. Docking Capability for OTV and LADV Individually and Simultaneously	Universal docking unit based upon IOC Space Station resource node accommodates up to 3 spacecraft at one time.
4. Tank and Cargo Transfer Node to OTV: 9 tanks exchanged and 3.3 MT cargo transferred Node to LADV: 3 tanks exchanged and 1.1 MT cargo transferred	Teleoperated boom and end effectors with capabilities that include: • Reach up to 15 m. • Mass up to 12 MT. • Position within 20 cm during transition with provisions for fine motion within a 1 cm diameter during engagement.
5. Communication and Rendezvous Tracking Links for both manned and teleoperated docking	RF links via communication satellite relay include: Command, control, telemetry data, video and relay of docking support data from radars and lasers to show ranging and positioning.
6. Spacecraft Function as Emergency Haven for Lunar Crew (1500 man-days support capacity 110 day operation) with power, ECLSS, GN&C, and expendables included on-board	Node structure is a pressurized cabin based upon the IOC Space Station habitat module. Exterior modified for tanks and cargo attachment.
7. Spacecraft Functions as Man Tended (Two man crew 5 days max) with power ECLSS GN&C and limited expendable	Node structure is open gridwork of beams with attachment points for tanks and cargo. Resource node plus an auxiliary enclosure (sealed down logistics module) houses all operating subsystems.

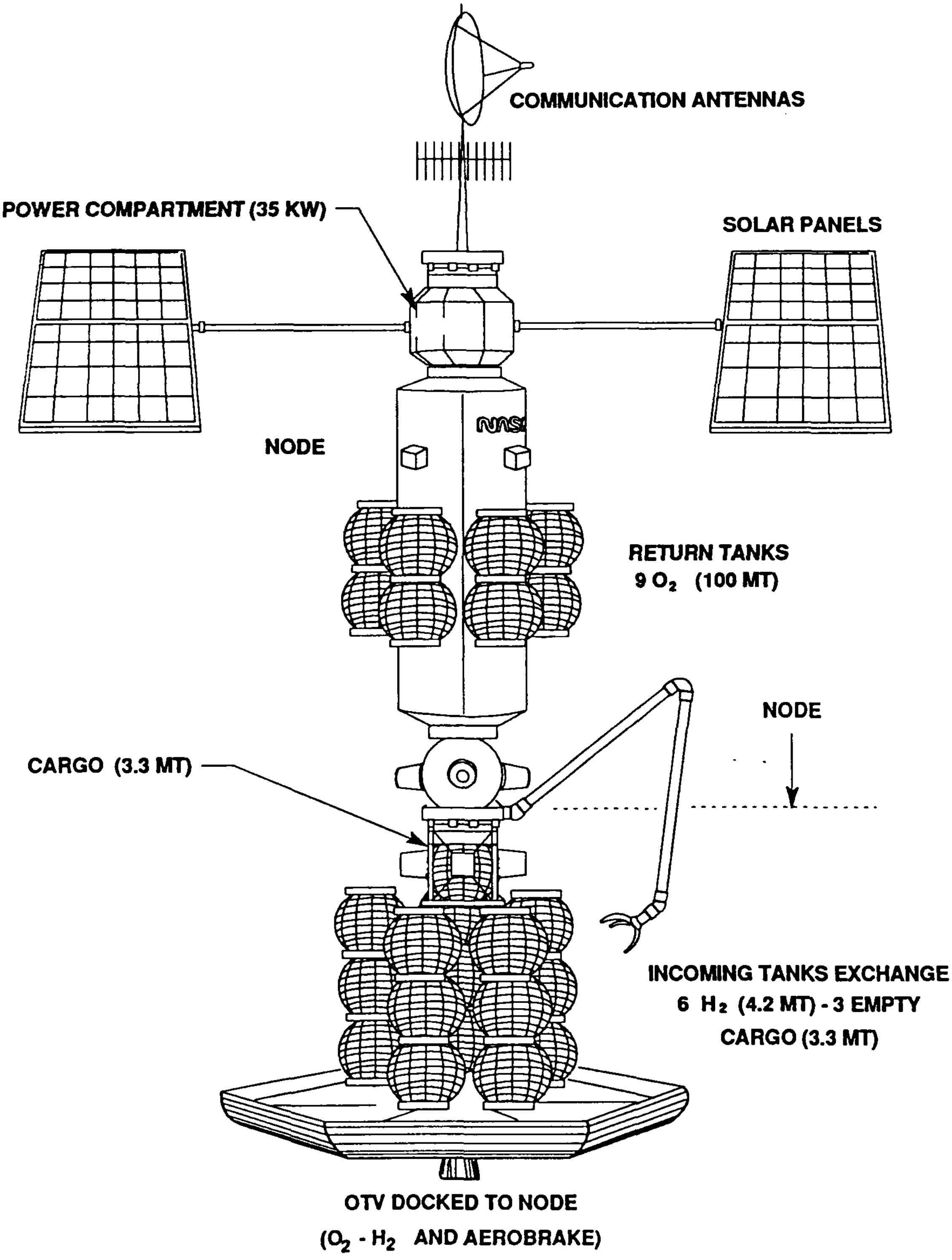

Figure 5-1 Concept for the Safe Haven Lunar Node Docked to the OTV for Transfer of Propellants and Cargo

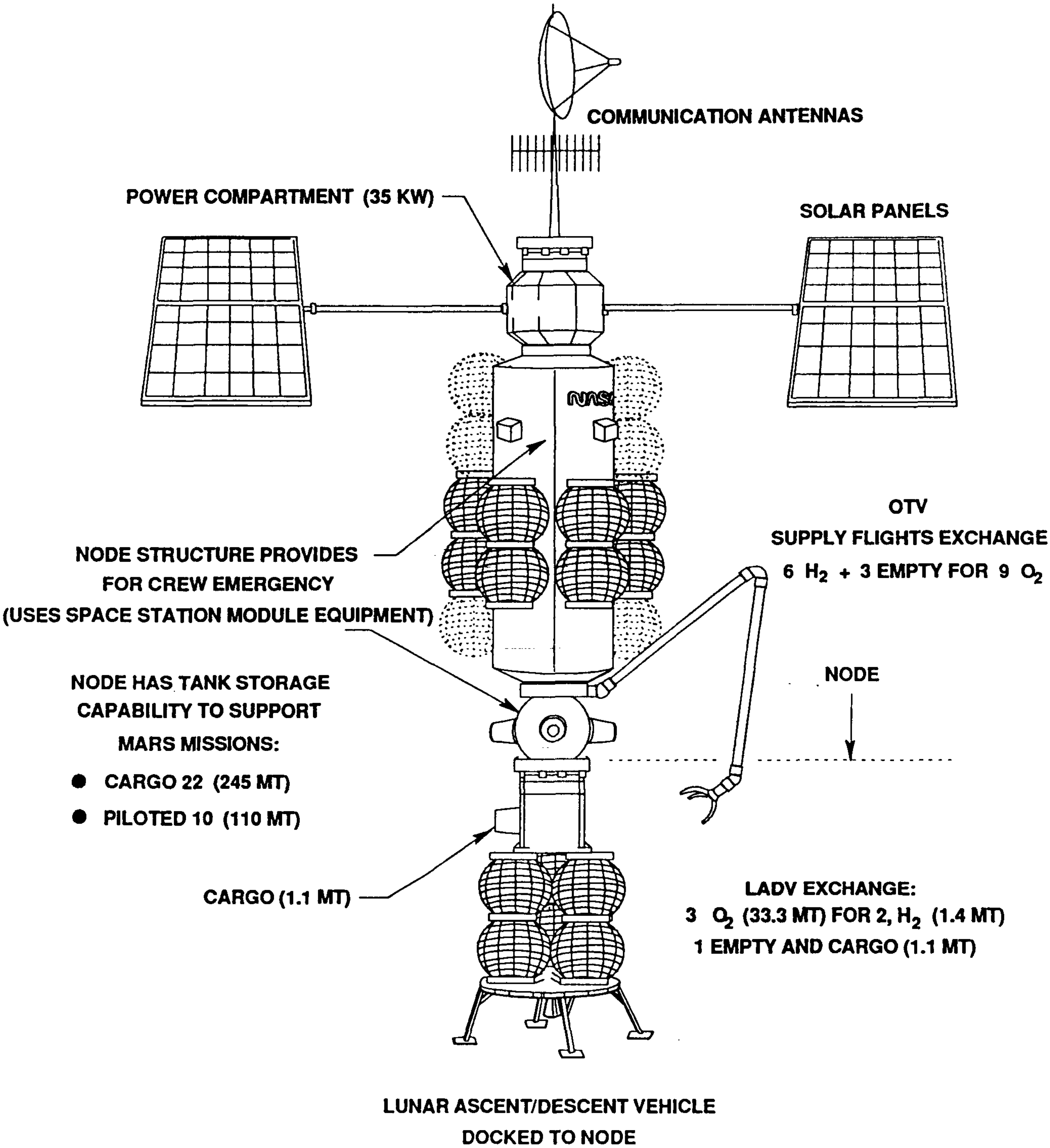

Figure 5-2 Concept for the Safe Haven Lunar Node Docked to the LADV for Exchange of Propellants and Cargo

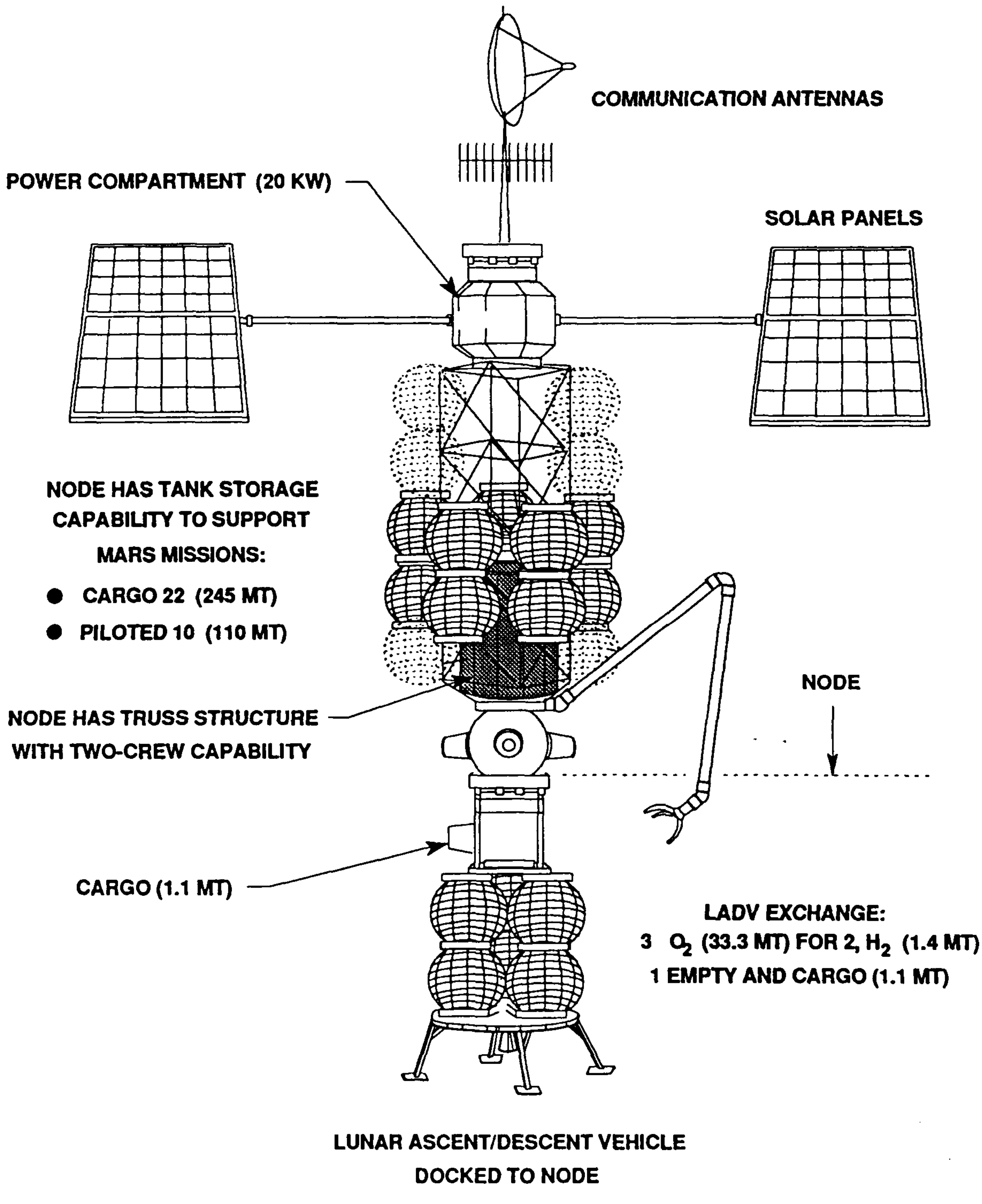

Figure 5-3 Concept for the Man Tended Lunar Node Docked to the LADV for Exchange of Propellants and Cargo

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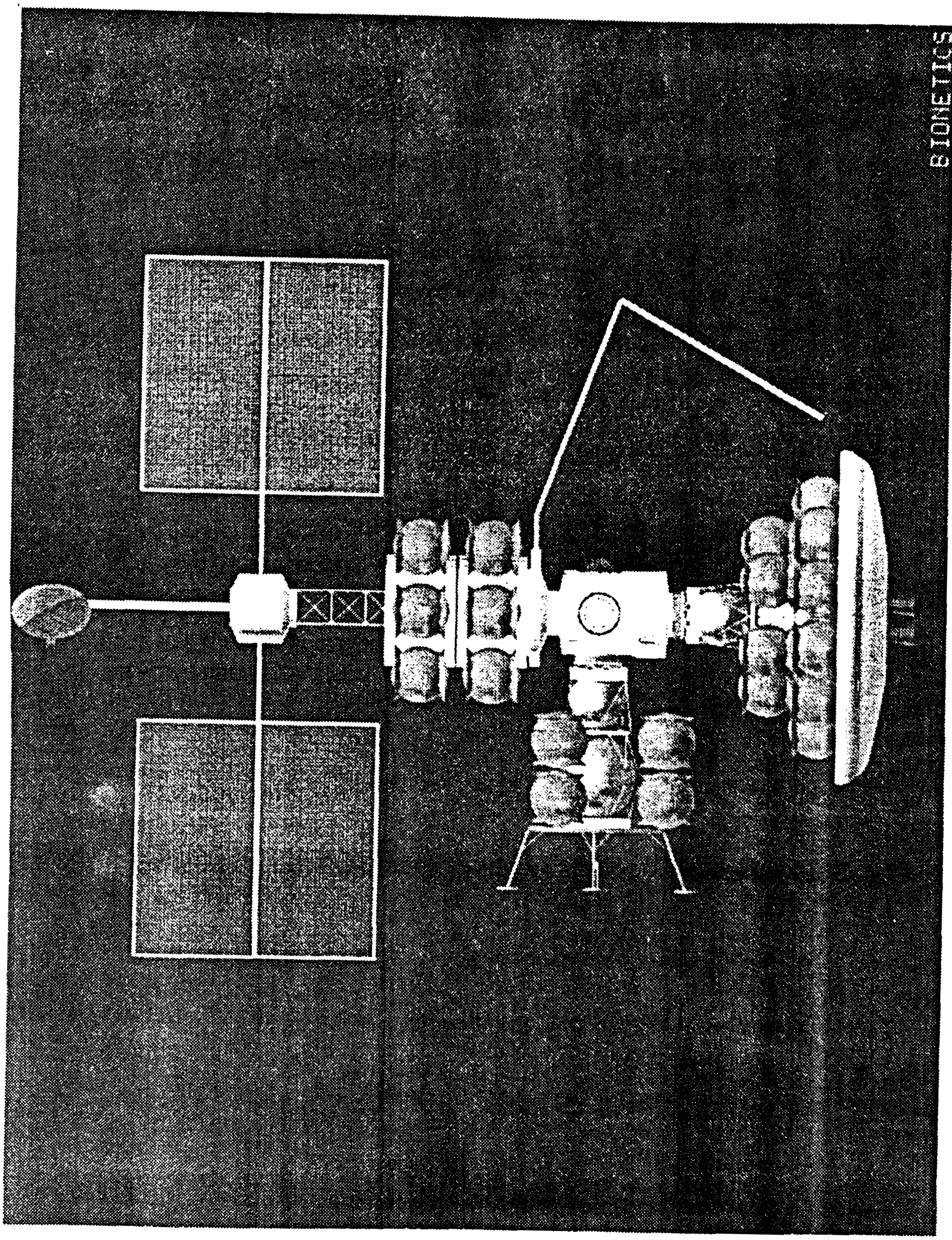

Figure 5-4 Alternate Concept for the Man Tended Lunar Node Docked to the OTV and LADV

The tracking system operates line-of-sight to the horizon for operation with the LADV, and line-of-sight as available for the OTV. Both the communication and tracking functions require antenna position controls.

D. Cryo-Maintenance The attachment mechanism for securing the tanks includes a collection manifold linked to the tank vent-and-fill lines. The OTV flight schedule determines the maximum on-board storage time for any tank. A nominal reliquification capability is included to preserve the O_2 and H_2. The option remains to retain the boil-off, using the gases for attitude stabilization or potentially in fuel cells for energy during dark periods.

E. Power System The power system has been estimated on the basis of 15 kW to operate the node plus 2.5 kW for each person aboard. The photovoltaic power system has been sized for 20 kW or 35 kW, depending upon the safe haven capacity. Regenerative fuel cells are the preferred power storage option.

F. Habitat Module Full Crew Safe Haven (Figure 5-1 and 5-2) This option has the capability to sustain the expanded crew for a period of 110 days. The structure and accommodations are based upon the IOC Space Station habitat module.

G. Minimum Crew Accommodation The minimum crew option utilized the Resource Node as the principal manned operations area, with the necessary ancillaries attached. An open beam structure supports and provides the retention mechanism for the tankage and cargo. The structural element concept resembles that used for the Long Duration Exposure Facility.

6.0 Lunar Node Mass Schedule

The Lunar Node, as described in the preceding sections, was evaluated to obtain estimates of subsystem masses. Two options were addressed, (1) the safe-haven habitat option providing 1500 man-days of emergency occupancy and (2) the man-tended option which offers only operational life support. In both cases, the Lunar Node exhibits one mass total when loaded with LLOX preparing for a transfer vehicle arrival from LEO, and another mass total following transfer of LLOX to the transfer vehicle and LH_2 and cargo from the transfer vehicle. These estimates for the two options and two conditions of loading (4 cases) are presented in Table 6-1. The mass budget does not include propellant for Lunar Node station keeping or for rendezvous/docking maneuvers.

7.0 Topics for Future Study

This cursory study of a Lunar Node must be examined in more detail to better define the actual requirements in any follow-on study. The mission mode described in Reference 2 eliminated the Lunar Node but retained the functions of material transfer in LLO. The Lunar Node could provide a focal point and buffer storage for crew, cargo, and propellant. Some features of the Lunar Node application to Reference 2 requirements are presented in Figure 7-1.

A mission analysis should be conducted to trade-off the need for a Lunar Node and to define the crew safe-haven capability that covers the majority of contingencies.

LUNAR NODE: EVOLUTIONARY EXPANSION

TABLE 6-1 SUMMARY OF LUNAR NODE MASSES

MASS ELEMENT	NODE MASSES WITH O_2 (MT)		NODE MASSES WITH H_2 (MT)	
	Habitat	Tended	Habitat	Tended
Fuel and Tanks Transferred	102.3	102.3	6.4	6.4
Cargo (per trip)			3.3	3.3
Resource Node and Docking Adapter	5.2	5.2	5.2	5.2
Robotics and Transfer	7.5	7.5	7.5	7.5
Communications, Tracking G, N and C	3.5	3.5	3.5	3.5
CRYO Maintenance	2.5	2.5	2.5	2.5
Space Habitat	34.0		34.0	
On Board Supplies and Consumables	33.0	3.0	33.0	3.0
Man-Tended Truss and Structure		12.0		12.0
Power Supply	3.3	2.4	3.3	2.4
Total Node	191.3	139.3	98.7	46.7

6-14-88 TBC #17

LUNAR NODE: EVOLUTIONARY EXPANSION

Potential of Lunar Node for Support of Mars Outpost *

- Common Rendezvous Site for LLO Operations
- Staging Point for Lunar/Mars Crews and Supplies
- Buffer/Storage for Inbound/Outbound Flight Schedule Slip
- Assurance of LLOX in LLO for Mars before LEO Departure
- Smaller More Frequent Lunar LLOX to LLO Cargo Flights
- Safe Haven Potential at All Times in LLO

* As described in SRD June 2, 1988 Version 1.0

Figure 7-1 Lunar Node Support for Mars Missions

6-13-88 TBC #18

Sensitivity studies should be conducted for a range of propellant mixture ratios and specific impulse values to size the LLOX requirements. For this study, a fixed pair of values were used, namely 7:1 oxygen-hydrogen mass ratio and a 4310 N-sec/kg specific impulse.

Orbital analysis is required to define the Lunar Node orbital properties such as time in sunlight, station-keeping requirement, and the velocity change requirement for the rendezvous with transfer vehicles from LEO and the lunar surface. These results are required to determine solar panel and energy storage capacities, viewing periods to various communication sites, and rendezvous opportunities.

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3. Hoy, T. D. et al.: Conceptual Analysis of a Lunar Base Transportation System, George Washington University - NASA Langley Research Center.

Input to Volume III
Special Trade Studies, Reports, and Systems Assesments

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TABLE OF CONTENTS

i. Introduction

The following 312 pages is a compilation of various study results, performed by Martin Marietta for the Transportation Agent (NASA/MSFC), which were the culmination of the four case studies presented in the OEXP Exploration Studies Technical Report (FY 1988 Status). By agreement with the MASE, only certain portions of Case study 4 were examined in any detail, because of changing requirements for this Moon/Mars Evolution pathway. Following up the detailed study reports, this collection is intended to provide supplementary information as well as more complete details generated during the study effort.

Although many of the ground rules and even the trajectory data base are expected to be altered for future studies, this material can be useful to provide insight into trade-off decisions that have been made and to bring to light various options which could only be given cursory treatment in the previous submittals.

Case Study 1

Human Expedition to Phobos

1.0 Transportation Systems Definition

The transportation for Case Study 1 consists of the vehicle necessary to preplace cargo into Mars orbit, the vehicle to transport the astronauts to Mars, an optional vehicle for flights from the mother spaceship in Mars orbit to Phobos and return, a propulsion system for return of the piloted vehicle to Earth, and a capsule to permit direct descent of the crew to the Earth's surface. These vehicles are termed the Mars Cargo Vehicle (MCV), the Mars Transfer Vehicle (MTV), the Phobos Excursion Vehicle (PhEV), the Trans-Earth Injection System (TEIS), and the Earth Crew Capture Vehicle (ECCV), respectively.The mission phases are listed in Table 1.0-1. This case study is baselined as all-propulsive, with only the ECCV utilizing aeroassist.

1.1 Elements and Systems Description

1.1.1 Transportation Requirements/Assumptions

This mission consists of the MCV launch in April 2001, followed by the human mission launch in August 2002. A nominal mission time of 30 days in Mars orbit includes 20 days of human exploration of Phobos. Fortuitously, this launch opportunity allows a swingby of Venus, greatly reducing the need for propellants by eliminating an outbound Deep Space Maneuver (DSM). Performing this mission at either the previous or next later launch opportunity greatly increases the propulsive mass requirements.

Other requirements and assumptions made for purposes of conducting the reference mission transportation analysis are given in Table 1.1.1-1. These assumptions are necessary in order to achieve a point design for the reference mission. An analysis shows that initial mass in low Earth orbit (IMLEO) can be significantly reduced (IMLEO reduction of 59%) through use of the relatively lightweight PhEV rather than requiring the entire manned spaceship to transfer to Phobos orbit. This is because Phobos lies in a near-equatorial, circular orbit about Mars, Figure 1.1.1-1, necessitating major plane changes of the spacecraft's orbit. Also, the relatively low velocity in Phobos orbit does not favor efficient escape from Mars' gravitational attraction for the return flight to Earth. Therefore, it is more effective to place the Mars Orbiting Vehicles (MOVs) into high elliptical orbits of 250 km periapsis by 33,840 km apoapsis. Then, to minimize total delta-V, the orbits are circularized at 33,840 km altitude and then reinstated by an additional propulsive maneuver at the appropriate time to achieve apsidal rotation. This special maneuver or some equivalent is required whenever short staytimes occur at Mars, such as for the sprint and opposition class trajectories, but can be accommodated during the longer conjunction class missions by orbital management strategies and much less expenditure of propulsion energy. Future studies will address alternative strategies for delta-V reduction for short staytime missions.

From all of the options available in this study, a baseline reference mission was chosen and is outlined in Figure 1.1.1-2. The baseline mission is called TIC-1R for technical implementation concept 1-reference. For the present study, the assumed sequence of events is: initial insertion into a 250 x 33,840 km elliptical orbit about Mars; circularization at 33,840 km altitude; rendezvous with the cargo vehicle in the same orbit; transfer of the PhEV and TEIS to the manned vehicle; and return to a 250 x 33,840 km orbit, but with the line of apsides appropriately chosen for optimum TEI. The scenario for PhEV rendezvous with Phobos then proceeds along the sequence indicated in Figure 1.1.1-3. Following deployment in orbit 3, the PhEV lowers its apoapsis to Phobos altitude (orbit 4), then raises periapsis to circularize (orbit 5). Upon rendezvous, a series of sorties to near the surface of Phobos allows EVA and exploration by one astronaut with the aide of an MMU (manned maneuvering unit). The other astronaut, fully suited, tends the depressurized PhEV and provides assistance, if needed, to the first astronaut. To return to the mother ship, the sequence of orbital maneuvers is accomplished in reverse. The trans-Earth injection burn is accomplished from elliptical orbit.

1.1.2 Reference System Description

1.1.2.1 Configuration and Mass Allocations

Cryopropellants (liquid hydrogen and liquid oxygen) are carried in standardized tanks designed to match the assumed ETO vehicle lift capability of 91 t to LEO. Tanks are 7.6 m diameter by 11.4 m long (25 ft dia x 37.5 ft), each holding 69.3 t of cryopropellant with an H/O ratio of 1:6. A pair of tanks is lifted each launch, as shown in Figure 1.1.2.1-1. These "Siamese twin" tanks are not exactly identical but hold the same quantity of propellant and are connected by propellant transfer lines. The "wet" tank is ruggedized for the launch vibration, acceleration, and acoustic loads on a full tank, and equipped with foam insulation to store cryopropellant under atmospheric conditions prior to and during launch. Upon achieving orbit, propellant is transferred into the "dry" tank via an automated transfer process. The dry tank is of lighter construction, with a 15% tankage factor (where tankage factor is the ratio of tank dry mass/propellant mass), and thermally protected for long-term storage of cryogens in space. This includes multilayer insulation blankets and vapor-cooled shields (VCS). This tank is to be used for all cryopropulsion stages, and achieves the low boiloff rates as specified in the assumptions (Table 1.1.2.1-1). The wet tanks are discarded after propellant transfer. One or more tanks are pre-outfitted with an engine and propulsion avionics. To assemble a complete propulsion system, several tanks are docked together with their propellant fill ports connected. These ports are envisioned to be of technology derived from the STS 17-inch disconnects. During propulsive burn, all interconnected tanks drain propellant into the engine's primary tank.

The habitability module is portrayed in Fig. 1.1.2.1-2a and 1.1.2.1-2b. It is an "H-module" configuration, consisting of two space station derivative modules with a single tunnel between them. At the mid-point of this tunnel, the ECCV is mounted. Because the ECCV ingress portal is in the nose, its interior is available to the crew at all times for habitation volume and access for continued training. It is necessary that the ECCV travel with the piloted vehicle so that it is available in the case that the optional Mars fly-around abort mode is selected in lieu of proceeding with Mars orbital capture and rendezvous with the cargo vehicle. No airlocks are provided, but EVA is made possible by venting the tunnel (which has an egress port opposite the ECCV) or one entire module.

The cylindrical habitats employ the pressure vessel and much support structure derived from Space Station modules. However, the massive internal experiment rack hardware is mostly replaced, both to provide a lighter-weight mounting structure appropriate to the small TMI propulsive loads and to allow more usable living volume for the astronauts. The interplanetary mission modules (IMM), including their electrical and communications support services, are sized at a mass of 44.3 t.

A solar flare radiation storm shelter is provided at the end of one of the modules (Fig. 1.1.2.1-3a). It consists of an approximately cubicle volume designed to hold 4 persons (functional reach envelope for torso-restrained, unsuited 95th percentile male, as in Fig. 1.1.2.1-3b[1]). A minimum of 20 g/cm^2 shielding is provided in the walls of this shelter through the use of judicious equipment installations, stowage of consumables and waste products, and added shielding and structure of 2.0 t. The shelter includes equipment for command and control of the spacecraft.

A life support system (LSS) with fourfold redundancy is provided at a mass cost of 2.8 t. The LSS provides recycling with respect to oxygen, carbon dioxide, and water. A total of 10.2 t of consumables (including 3.3 t food) is allocated to support the four astronauts for the duration of the mission, with 20% margin. Power is provided by two solar cell array wings of 100 m^2 each, deployed from the sides of the individual cylindrical modules.

The Phobos Excursion Vehicle (PhEV), Figure 1.1.2.1-4, is similar to a Gemini capsule and holds a crew of two. Notable differences include capacity for a 700 kg Phobos science payload (or, radiation shielding) and two MMU units, as well as a major propulsion system. The propulsion employs the space-proven Delta engine and the associated storable bipropellants sized to accomplish a total round-trip ΔV of 3327 m/s, as stipulated by Table 1.1.2.1-2. The PhEV gross mass is just under 10 t, and the complete mass breakdown is given in Table 1.1.2.1-3.

The Trans-Earth Injection System (TEIS) for Mars orbit escape consists of a single standard tank loaded with 51.9 t of cryopropellant. Three RL10B-2 engines rated at a specific impulse of 460 s are provided. These engines are not used prior to the TEI burn. The engines are mounted in a close-packed triangular cluster, as indicated in Figure 1.1.2.1-5, and the system has one engine-out capability. Acceleration at TEI initiation is 0.18 gee, rising to a peak acceleration of 0.32 gee.

The Earth Crew Capture Vehicle (ECCV) can be likened to the Apollo Command Module. It enters the Earth's atmosphere at less than 12.2 km/s (40,000 fps) with the aid of an aerobrake and parachutes for terminal splashdown. The mass of the ECCV is 6.9 t, including the four crewmembers. No propulsion is required for the ECCV to accomplish its mission, except for a small propulsive system which accomplishes final targeting (the main vehicle is targeted slightly off Earth intercept) and to provide roll and attitude control.

The piloted vehicle stack in LEO is shown in Figure 1.1.2.1-6a, (expanded view in Fig. 1.1.2.1-6b), where the lower two tiers of tanks are associated with the Trans-Mars Injection System (TMIS). The propellant load of this system is 811.5 t, stored in twelve standard tanks. An SSME-derivative engine provides the thrust necessary for escape from LEO onto the interplanetary trajectory to Mars (departure energetics are detailed in Table 1.1.2.1-4). The SSME-derivative employs an enlarged nozzle with 1000:1 expansion ratio and specific impulse of 480 s. The exit diameter of the bell is 8.2 m (27 ft). With its associated hardware, the nozzle + engine head + loaded tank + siamese twin tank (wet tank), the stack at launch is 28 m (92 ft) which may be accommodated depending upon the HLLV available. Alternatively, the nozzle could be segmented and then extended after launch, saving 8 m (26 ft) in stack length. The Mars Orbital Capture System (MOCS) is also a cryogenic propellant system, which in this case also provides Mars Orbital Operations (MOO) propellant sufficient for 1338 m/s of capability for apsidal rotation. It consists of four standard tanks, each with an RL-10B-2 engine. Engine-out capability is

[1] Manned System Integration Studies, NASA-STD-3000, Vol. 1, (March 1987), p.8.6-4.

provided. Initial deceleration at the beginning of the MOC burn is 0.13 gee. The encounter energetics for Mars orbital capture are given in Table 1.1.2.1-5.

The Trans-Mars Injection System (TMIS) of the cargo vehicle, Figure 1.1.2.1-7a, (expanded view in Fig. 1.1.2.1-7b), consists of four cryopropellant tanks and one SSME-derivative engine. The MOCS/MOOS propulsion system is a single tank with a triangular-cluster of RL-10B-2 engines. The cargo vehicle carries not only the TEIS and PhEV, but also a Relay Communication Satellite and two Mars Rover/Sample Return (MRSR) modules. The mass allocation for these additional payloads is 9 t, including all propellant loads necessary for these systems. An additional 0.45 t of instrument payload serves to provide solar flare monitoring capability and Mars orbital science.

1.1.2.2 Features of the System

This spacecraft is designed as a minimum system for accommodation of 4 astronauts for a deep space mission. It provides somewhat more living volume per person than Space Station will provide to its occupants by virtue of the fact that not as much equipment will be installed. Neither airlocks nor nodes are provided. A cupola could be added to the end of one module at a small mass penalty. There is very little margin for error in rendezvous with the TEIS in Mars orbit. In the event that orbit insertion errors were large, the MCV could jettison the PhEV to lowers its mass and hence provide some additional orbit modification capability in an effort to transfer to the orbit the astronaut vehicle had reached.

1.1.2.3 ETO, On-orbit Assembly, and Servicing Needs

With an assumed HLLV capability of 91 t per launch, the ETO manifest shows that a minimum of 7 launches will be required to deploy the cargo vehicle and 18 launches for the piloted spacecraft and its propulsion systems (Table 1.1.2.3-1). Of these, all but two will be dedicated solely to launch of propellant and propulsion system hardware. If the launch capacity is increased to 200 t by using the "Very Large HLV", on-orbit assembly decreases dramatically, as shown in Table 1.1.2.3-2. In order to launch the entire craft already assembled the "Magnum HLV" which has an assumed capacity of 1360 t must be used. A summary launch profile is given in Table 1.1.2.3-3.

On-orbit assembly will be primarily by automated and teleoperated control from Earth. Propellant tank assembly into the necessary propulsion system clusters will be accomplished via docking maneuvers and plug-in propellant lines. An OMV, OMV/FTS, and/or smart HLLV upper stage will be required as infrastructure to support this assembly. The IMM could be launched as a complete unit, including ECCV, if a 42 ft diameter HLLV payload shroud were made available.

No in-space propellant transfer is needed for this case study scenario. All tanks are sized to allow for on-orbit propellant boiloff losses. It is currently assumed that the MLI blankets and tankage walls will adequately minimize the probability of a leak induced by orbital debris and micrometeoroid impacts. Therefore, no special protection blankets need be removed prior to TMI.

Other servicing requirements are also minimal and no STS visits are required for the cargo vehicle, although they may be desired for inspection of the assembled system. On-orbit operation and checkout of the IMM prior to final mating with the piloted TMIS will be required to develop the prerequisite baseline for mission assurance. A minimum of three STS launches is estimated to be required to support this mission.

1.1.3 Transportation Program Development Schedule

A schedule for development, proof-flight testing and man-rating of transportation hardware and propulsion systems is shown in Figure 1.1.3-1. It must be stressed that the development of flight hardware must include time for prior development and man-rating of key elements in the mission. For example, the HLLV, PhEV, and TEIS characteristics will seriously affect design of both the cargo and piloted vehicles and therefore should be developed as early as possible. Demonstration of reliability can be made by precursor missions of various types, including Earth orbital and interplanetary unmanned launches. One method of verification would be manned operation of the PhEV on a LEO mission which exercises the near-Phobos operational capabilities, followed by simulated rendezvous with the mother spacecraft. Additionally, the PhEV also could be operated unmanned to perform major orbital sequences at high altitudes to simulate the accuracy of transfers to and from Phobos. Both the HLLV and PhEV should be well into in-space testing by the mid-C/D phase of Mars vehicle developments, and preferably much earlier. Prototype habitability modules and other key component of the system could be pretested on Shuttle or Space Station flights. Long-term testing of life support system and microgravity countermeasures must be accomplished before the deadline for alteration of interior equipment complements for the piloted spacecraft. Long-term storage and successful operation of a TEI system must be tested *in space* prior to launch of a crew to Mars. If a Mars aerobrake is to be employed (see Alternative, section 1.3), it should be assembled, on-orbit launched, and entry-tested in the Earth's upper atmosphere prior to beginning the C/D phase of the piloted system. The need for additional aerobrake performance verification at Mars is under assessment.

1.1.4 Trades/Options

Several options, listed in Table 1.1.4-1, have been studied for their effect on IMLEO of the total mass of cargo plus piloted vehicles, which is 1778.3 t for the baseline design. Use of conservative, high-boiloff tanks incurs a very significant additional mass penalty of 488 t. Conversely, if boiloff could be reduced to zero, some 152 t could be saved (less than a 9% reduction). Utilization of a two-stage TMIS would save even less off the baseline design. Therefore, the more complicated system that would be required for successful staging is not adopted.

Advanced propulsion engines, with specific impulse capabilities of 485 s (TMIS) and 470 s for cryopropellant, and 340 for stored bipropellant, result in a savings of only 6% in IMLEO. In view of the development time lags associated with these advanced systems, they were not assumed for this case. Conservative tankage factors of 20.6% (compared to the 15% assumed) combined with high boiloff resulted in a more than a doubling of IMLEO. Conversely, if the tankage factor could be reduced to 7.5% through use of advanced materials and technologies, over a 500 t reduction in IMLEO could be realized. A summary of these results is given in Table 1.1.4-2, as well as being depicted graphically in Figure 1.1.4-1. Comparisons between the launch manifests of missions with different trajectories and propulsion systems are made in Table 1.1.4-3

Two options also exist for Phobos exploration, as shown in Table 1.1.4-4 and in Figure 1.1.4-2. The first, called "Sub-scenario A", differs from the reference mission in that the cargo vehicle is captured into a parking orbit ellipse of 250 km x Phobos, orbit 1 in the figure, instead of the 250 km x 1 sol of the reference mission. The cargo MOV then circularizes into orbit 2 where it meets with the crew vehicle for TEIS and PhEV transfer. After the exploration of Phobos the return vehicle leaves via trajectory 3. The second option, "Sub-scenario-C", differs from the above in that once the cargo vehicle has achieved orbit 1, the transfer of the TEIS and PhEV takes place. After this has been accomplished, the vehicle circularizes to orbit 2 and the rest of the mission proceeds as above. The trade results are summarized in Table 1.1.4-5.

Major alternative approaches such as use of aerobraking at Mars and more advanced propulsion systems are considered in section 1.3.

1.2 Enabling Technology Needs

Many technology needs are evident in missions of this class. First and foremost are development of an HLLV and of the Space Station. A larger lift capacity of the HLLV will significantly reduce the number of launches and amount of on-orbit assembly (see Table 1.1.2.3-3). The shroud diameter is also of considerable importance. The Space Station is required for studying the effects and countermeasures against three major potential problems in long-duration spaceflight: deleterious adaptations to microgravity; diagnosis and treatment of complex medical problems; and psychosocial adjustment to the isolated and confined environment. Radiation hazards must also be understood and appropriate shielding provided. Other major developments include propulsion and storage of cryopropellants.

1.2.1 Propulsion engines

Space-operated qualification of the SSME-derivative engine will be required. Increased performance of the SSME and RL10 engines must be verified. Techniques for long-term in-space storage of the RL-10s must also be developed and tested.

1.2.2 Cryopropellant tankage

It is quite obvious from the discussion in section 1.1.4 that every effort should be made for advancements in cryopropellant storage and for minimization of the tankage mass fraction relative to propellant (the "tankage factor"). This includes consideration of advanced composites, removeable structures and shields, efficiency of large multi-layer insulation blankets, use of vapor cooled shields, and other options.

1.2.3 Precursor Missions

Selected missions will be needed to provide spaceborne demonstration/verification of the PhEV, ECCV, IMM, and propulsion systems.

1.3 System Alternatives and Opportunities

The two alternatives to our reference mission are listed in Table 1.3-1 The first involves use of an aerobrake to achieve Mars Orbital Capture. This allows elimination of the very large propulsion system otherwise required for the same purpose. For a Mars aerobrake (MAb) mass fraction of 10%, the reduction in total system mass is quite dramatic (Table 1.3-2). IMLEO drops by a factor of nearly 2.4, to a value of 764.6 t. With this change, tankage and boiloff factors become relatively less important. For example, use of 7.5% tankage factor results in only an additional 20% IMLEO reduction. Even with high boiloff assumptions, and the adoption of a more storable TEI propellant such as hydrocarbon/LOX, the IMLEO increase is only about 10 t. This allows consideration of a more reliable TEIS, eliminating the difficulty of storage of liquid hydrogen for very long time periods.

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The second alternative proposes the use of a Nuclear Thermal Rocket (NTR), which also results in major savings, based upon the demonstrated NERVA technology which allows a specific impulse of 850 s. Without the aerobrake, the IMLEO reduction is nearly 40%. Combined with aerobraking, the IMLEO becomes 510.1 t, almost 3.5 times less than that required for the all-chemical, all-propulsive baseline. The resulting mass changes from both alternatives are summarized in Table 1.3-3.

Figure 1.0-1 Mission Phases

Phobos Expedition Missions

A.	Pre-launch	
B.	Earth orbital	
C.	Earth escape	(TMI)
D.	Transfer to Mars	
E.	Mars orbital capture	(MOC)
F.	Mars orbital/landed	
	Subphase 1 - Cargo/MSS Rendezvous and Transfers	
	Subphase 2 - Phobos Exploration by PhEV	
	Subphase 3 - Return to MOV, Rendezvous, and Docking	(ARD)
G.	Mars escape	(TEI)
H.	Transfer to Earth	
I.	Earth capture/recovery (direct entry)	(EOC)
J.	Post-Landing on Earth	

Notes:

A. Exec. authoriz.; Program office levels I, II, III; Contractor phases A/B, C, D; Crew selection/training
B. On-orbit assembly, checkout, fuel-up
H. Orbital capture; direct entry or rendezvous and transfer
J. Hardware inspections and test; Astronaut de-briefings, re-assignment to future programs or other activities

Table 1.1.1-1. Transportation Requirements and Assumptions

Requirements from the SRD

- Man-rate transportation hardware 3 yr before launch. Four 6-hr EVAs. (paragraph 3.1.1 of SRD)
- Completion of microgravity countermeasures research (p. 21)
- Minimize on-orbit assembly and SS support. Split:sprint/conj. Fly-around aborts.
- 1-2 yr in-LEO demo/verif of process requirements (3.3.2)
- No radiation shielding for PhEV (from summary sheet), but required in MOV
- 20-30 days at Phobos, close proximity, "but not land per se".
- Dock with a previously planted anchor. "Crew stability/mobility aids" for EVA work on Phobos
- Direct entry (per FAX)

Assumptions for Reference Mission

- All-propulsive; ECCV for crew recovery at Earth; no recovery of ETV
- Excursion vehicle (PhEV; 9794 kg) to Phobos
- Crew contact with Phobos via EVA flight with MMU. PhEV does not contact Phobos surface.
- PhEV station-keeps at 100 km from Phobos, with four 6-hr sorties to the surface
- Single TMIS stage, non-recoverable
- Engine performance: I_{sp}=480 for TMI, 460 for other cryo; 320 for storable biprop
- Propulsion: Cryo for TMI, TEI, DSM, MOC, MOO. Biprop for PhEV, RCS
- TMI Engine: Single SSME (emergency use of MOCS for flyback in case of engine-out)
- Propulsion tankage factor: Nominal (cryo: 0.15; storable: 0.058)
- Boiloff. low: 0.15 %/mo. LEO, 0.3%/mo. interplanetary (sprint), 0.065%/mo. at Mars
  high: 0.55 %/mo. LEO, 1.0 %/mo. interplanetary (sprint), 0.33 %/mo at Mars
- Propellant margins: 1% each for ΔV, I_{sp}, and bulk (use sum of margins)
  2% mass margin on TEIS and ECCV retro-propulsion (if required)
- No Venus probes.
- Phobos science payload: 1.2 t, 200 W. Two each 3.5 t MRSR packages.
- Hab modules: two SS-derived modules ("H" configuration)
- PVPA for spaceborne power, 200 m^2
- Spaceborne ECLSS: closed for all, except food
- ΔV: 100 m/s for MOO rendezvous, each vehicle, plus 619 m/s maneuvers to high circular orbit
- MCV drops MRSR, RelayComSats from HEO-1 (prior to circ); PhEV from HEO-2.

Figure 1.1.1-1
Post Aerobrake Orbit and Orbits of Phobos and Deimos

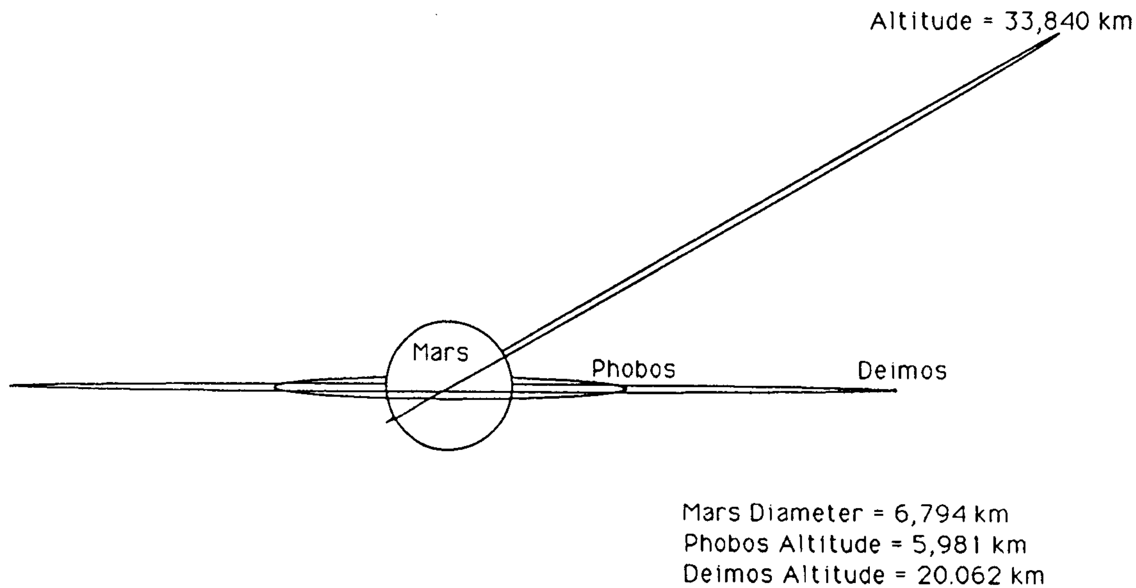

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Figure 1.1.1-2 Options Selected for Case Study 1

Scenario TIC -1R

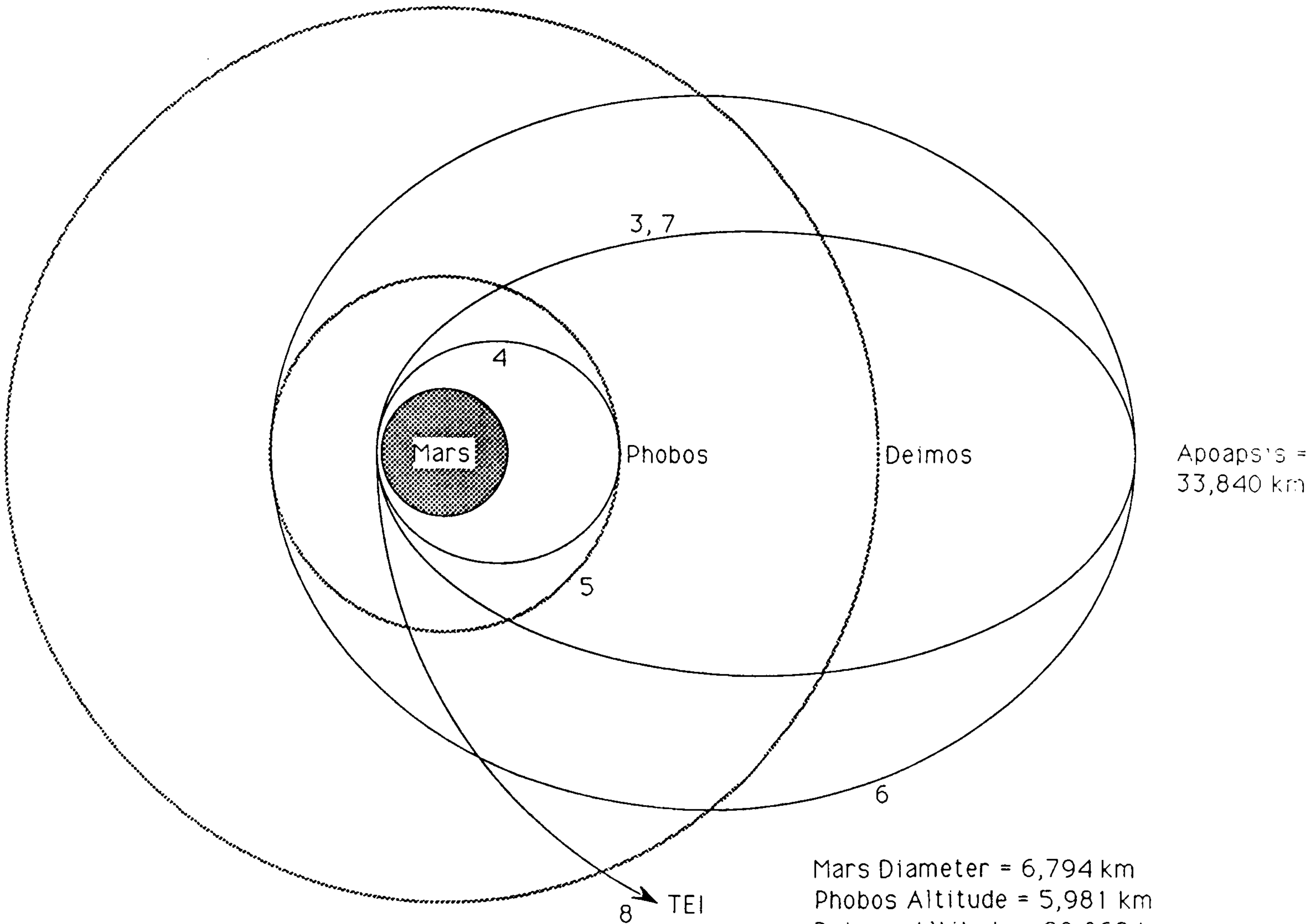
	OPTIONS — Options Selected: (Date & Your Name) — proven, or under development .... unproven, or must be developed / analyzed							
Earth departure location	**LEO**	HEO	GEO	L1,L2				
On-orbit assembly	SS, attached	SS, free-flyer	**Min. SS**	no SS req'd.				
Hardware staging	integrated	split, MDV	**split, MRSR**	**split, TEIS**				
Trajectory type	flyby	conjunction	opposition	**sprint**	low thrust	cycler		
Launch dates	1990's	**2000's**	2010's	2020's	2030's			
Crew size, total	3	**4**	5	6	7	8	9-11	12-24
Cabin pressure	4.3 psi	**14.7 psi**	10.2 psi					
Gravity environment	**microgravity**	artificial, .38 g	artificial, 0.6 g	artificial, 1 g	hybrid			
Rotation rate	**0 rpm**	1 rpm	2 rpm	4 rpm	6 rpm			
Radiation protection	none	**one storm shelter**	two storm shelters	GCR shield				
Hab/Lab modules	**SS modules (15' dia.)**	ET derived (25' dia.)	large dia. (31' dia.)	inflatables				
Science equipment	**interplanetary**	**Mars orbit**	**Mars surface**	**Phobos**	Deimos			
ECLSS, spaceborne	consumables	**SS ECLSS**	**water recycling**	low mass/power	CELSS			
TMI launch propellant	hybrid chemical	**LH2/LOX**	Hc/LOX	waste/LOX	SEP	NTR	NEP	solar sail
engines, cryo.	RL-10 growth	F1 derived	**SSME derived**	advanced cryo				
growth	max.-sized tank	stretch	**cluster**					
reusability	**non-recoverable**	engines, avionics	all recoverable					
recovery method	**none**	turn-around	re-encounter					
Cryoprop storage	**passive**	active, refrigeration	active, reliquifaction	from H2O				
Power, spaceborne	**PVPA**	**fuel cells**	RTGs	nucl. reactor	DIPS	solar th.-dy.		
TM abort capabilities	**Mars swingby**	propulsive abort						
Mars orbit	LMO	8.2 hr. ellip.	**24.6 hr. ellip.**	Phobos	Deimos	GMO		
Mars orbit capture	**propulsive braking**	prop/aero hybrid	aerobraking					
Satellites Relay Com.	none	LMO	**24.6 hr**	12.3 hr	Molniya	GMO		
Mars Science Orbiters	**none**	polar	circ.	elliptical				
Unmanned Landers	none	penetrator(s)	rover(s)	**sample return**				
Phobos/Deimos	none	Phobos Teleop	Deimos Teleop	**PhEV**	DeEV			

Figure 1.1.1-2 Options Selected for Case Study 1 (cont.)

	OPTIONS (cont.)							
	proven, or under development	new developments						unproven, or must be developed / analyzed
Mars Lander								
Number of MDV's	none	one	two					
Time on surface	0	1 wk.	3 wk.	6wk.	6 mo.	1 yr.	PMP	
Crew size, landed	0	2	3	4	5	6-20		
Propellants, MDV	LH2/LOX	Biprop	H2O2	CH4/LOX				
MELS	de-orbit prop.	parachutes	aerobraking	terminal prop.	hover/translate	airbags		
Landing hazard	large, safe areas	pinpoint landing	terminal H.A.					
Power generation	RTG	fuel cells	PVPA	DIPS	nucl. reactor	solar th.-dy.		
Power storage	batteries, Ni-H	Regen. FC	HEDRB					
ECLSS, Mars landed	consumables	SS ECLSS	low mass/power	ISCP	CELSS			
MLSE	RVR	analyt. eq.	geopys. pkg.	meteorol. pkg.	biol. eq.	drill rig		
MLOE mass	none	0.5 t	1 t	2 t	4 t	8 t	16 t	
RVR, manned	none	unpressurized	press., 5 sol	press., 20 sol				
ISPU	none	CO2	CO	H2O2	LOX	CH4/LOX		
ISRP demos	water	food	H2O2	buffer gas	GOX	fertilizer	MLOX	
Propellants, MAV	UDMH/N2O4	solids	ISPP					
Earth Capture								
TEI launch propellant	hybrid chemical	LH2/LOX	Hc/LOX	waste/LOX	SEP	NTR	NEP	solar sail
engines, cryo.	RL-10 growth	F1 derived	SSME derived	advanced cryo				
growth	max.-sized tank	stretch	cluster					
reusability	non-recoverable	engines, avionics	all recoverable					
recovery method	none	turn-around	re-encounter					
Recovery ETV	not recovered	Ab EOC	propulsive capture	prop/Ab hybrid				
ECCV	none	Ab EOC	prop. capture	prop/Ab hybrid	direct entry			
Orbital retrieval	STS to SS	STV to SS	STS to Earth					

Form Revised by: M. Thornton 7-8-88

Fig. 1.1.1-3 Phobos Excursion Maneuvers and Mars Escape



Mars Diameter = 6,794 km
Phobos Altitude = 5,981 km
Deimos Altitude = 20,062 km

Figure 1.1.2.1-1 Siamese-Twin Tank Concept

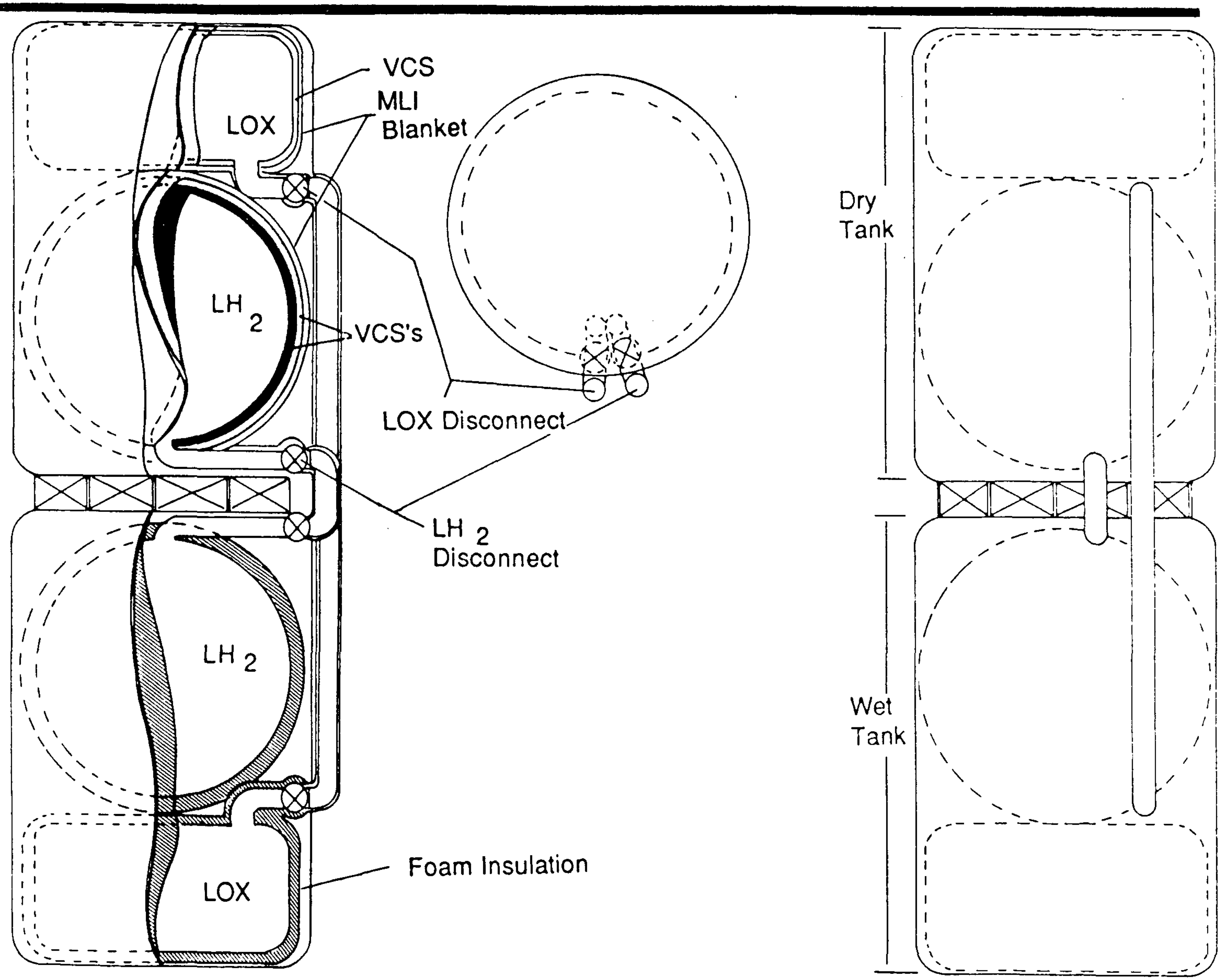


Table 1.1.2.1-1 Cryoprop Boiloff Assumptions and Guidelines

Boiloff Rates*

	Boiloff (%/mo.) wet-launched tanks	Boiloff (%/mo.) dry-launched tanks
LEO	0.55	0.15
Interplanetary		
Sprint	1.0	0.3
Conjunction	0.33	0.09
Mars orbit	0.3	0.065

* Source: Analysis of data provided by N. Brown, S. Tucker, and "Long Term Cryogenic Storage Facility (LTCSF) Systems S
36612.

Fig. 1.1.2.1-2a Interplanetary Mission Modules (IMM) with docked ECCV (4 Crewmembers)

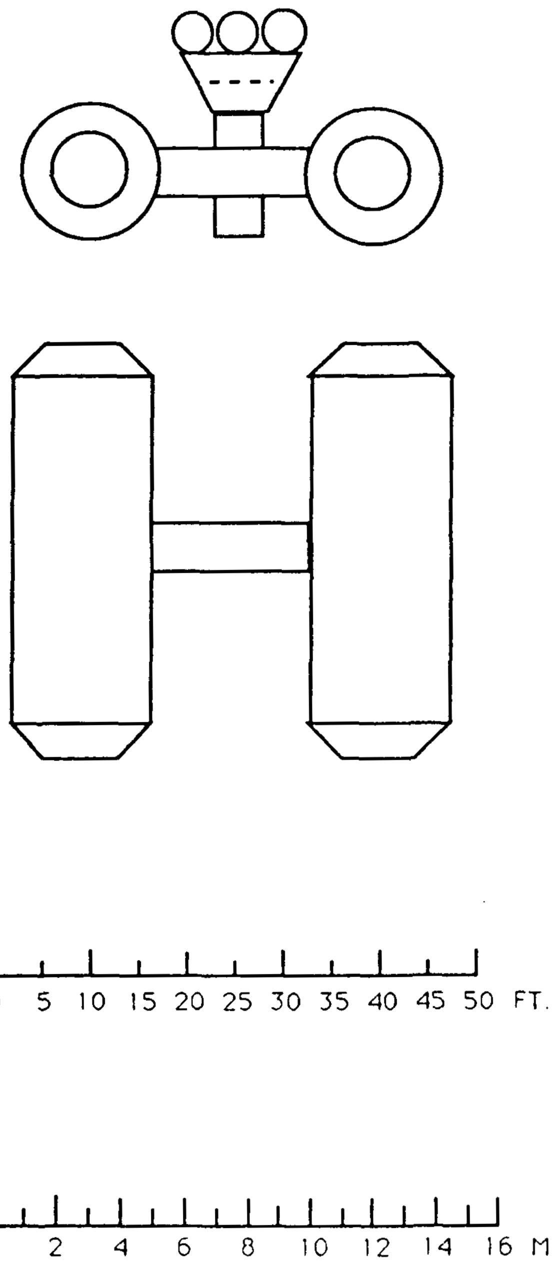


Figure 1.1.2.1-2b Artist's Rendition of Interplanetary Mission Modules (IMM) with Docked ECCV

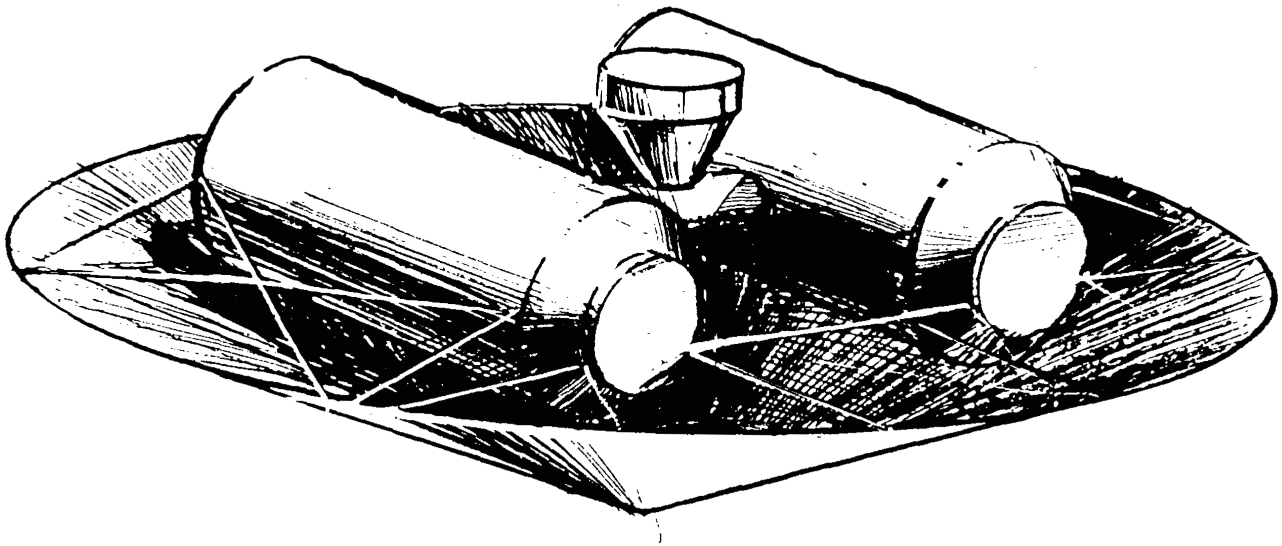


Figure 1.1.2.1-3a Radiation Shelter Situated at Habitat Module End

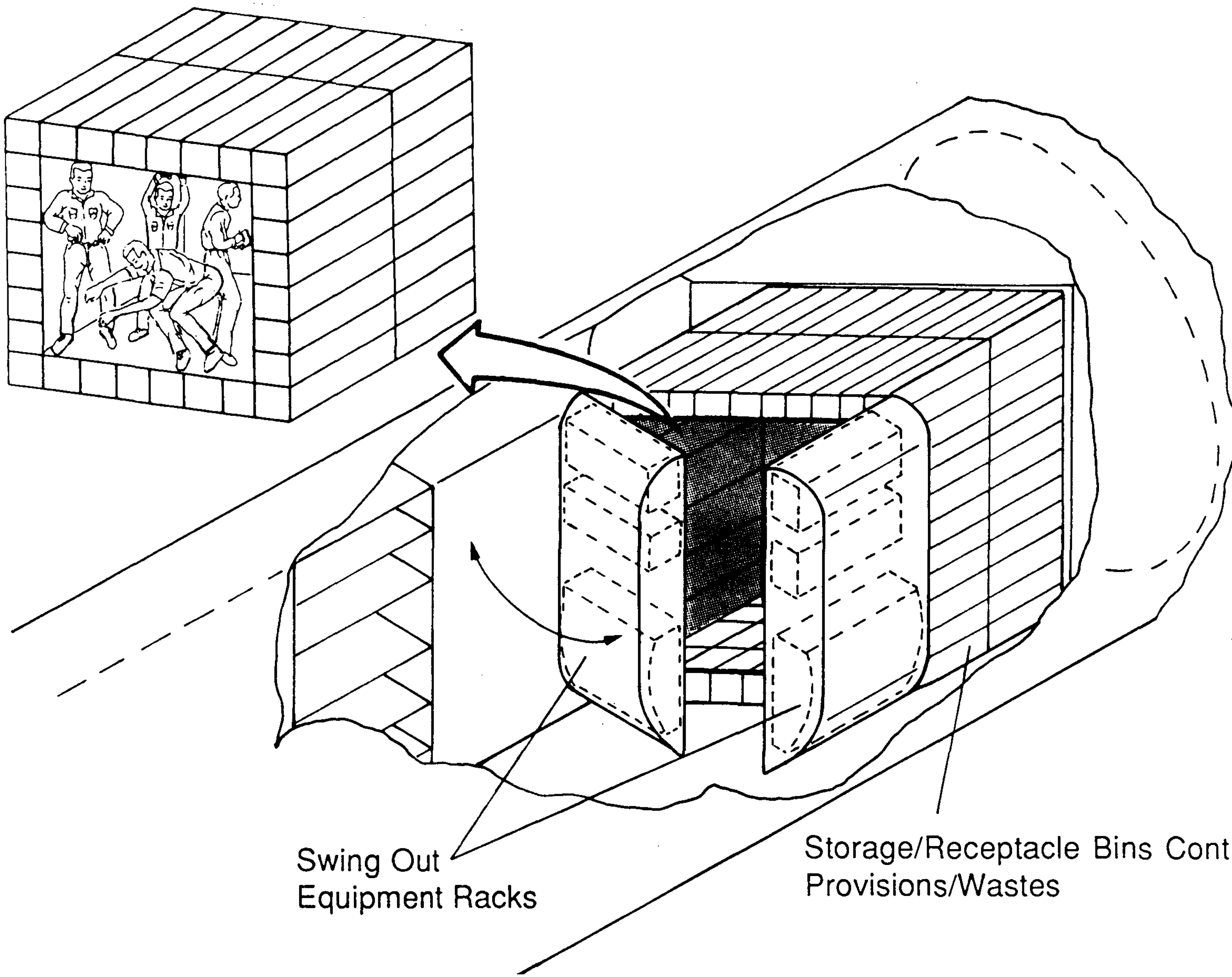


Figure 1.1.2.1-3b Envelope Geometry Design Considerations

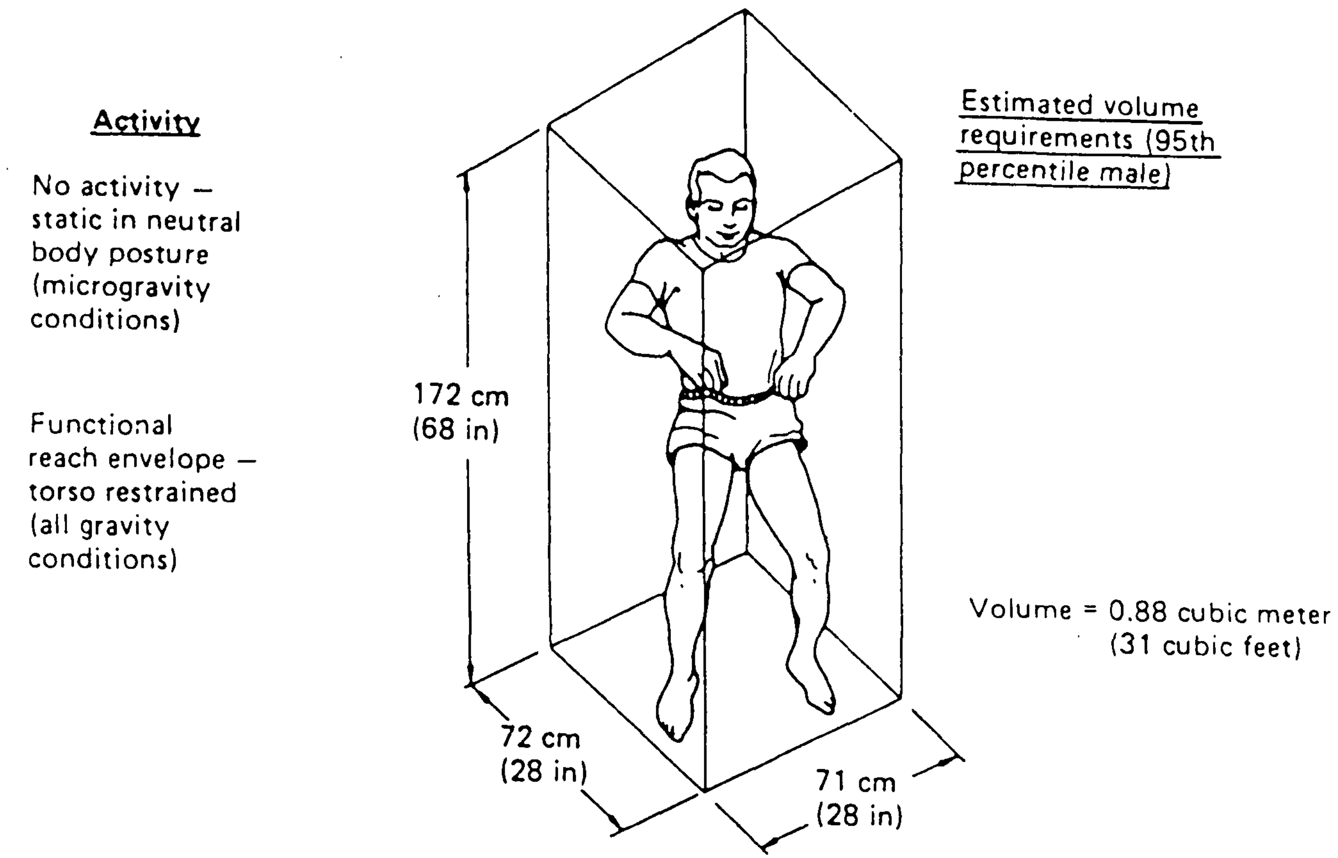


Fig. 1.1.2.1-4 Phobos Excursion Vehicle (PhEV)

(transfer and return from 250 x 1 sol orbit, 20 sol staytime, 2 crew)

Dry Mass	
(includes payload)	4,228kg
Payload Mass	
(user payload & crew)	1,670 kg
Payload Volume	
PhEV	
(cone - 1.8m rad., 2.3m ht.)	8 m^3
Propulsion System	
Propellant Type	MMH/NTO
Engines	
Number	1
Type	Delta
Mass (ea.)	100 kg
Thrust (total)(10.0 klb_f)	44.5 kN
I_{sp} (320 sec)	3.14 kN-s/kg
Propellant Mass	5,566 kg
Tank Mass	556 kg
Initial T/W	1.24
Mass Fraction	
Transfer to Phobos	0.7394
Phobos Sorties	0.7365
Transfer to 250 x 1 sol orbit	0.6920
Total Mass	9,794 kg

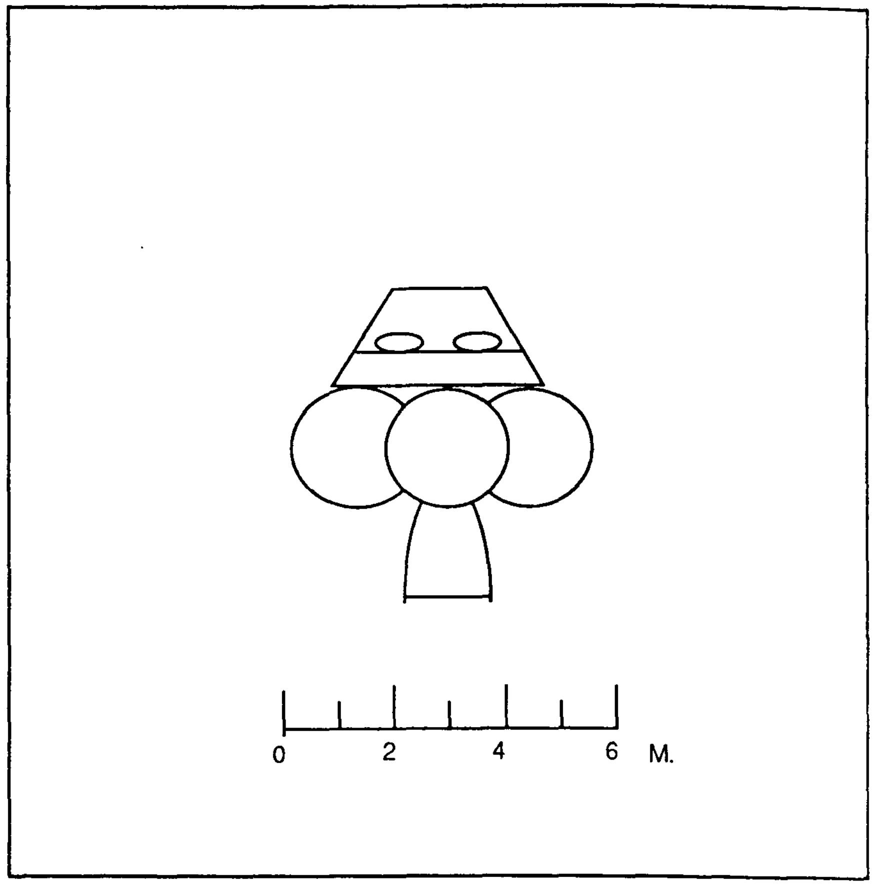


Table 1.1.2.1-2 ΔV Requirements for 2002 Phobos Excursion Vehicle

(from MOV orbit of 250 km x 1 sol, i = 24.5°)

	ΔV (m/s)
Plane change to equatorial orbit	264
Raise periapsis to 5981 km	
Lower apoapsis to 5981 km	
Rendezvous with Phobos	100
Total, MOV to Phobos	946
Phobos sorties	
(four 6-hr hover periods)	825
(20 days at 100 km distance)	134
Total Phobos sorties	959
Raise apoapsis to 33840 km	564
Lower periapsis to 250 km	227
Plane change to -24.5° orbit	264
Rendezvous and dock with MOV	100
Total, Phobos to MOV	1154
Round trip total	3059

Table 1.1.2.1-3 PhEV Mass Allocations Summary

Mass Allocations Summary Sheet (MASS) file name: PhEV. Scen #1.NTO

Scenario # 1 - Phobos Excursion Vehicle (PhEV)
Transfer from 250 x 1 sol orbit; 20 day staytime

Basis: E = Estimated (contingency = 10%)
C = Calculated from Drawing (contingency = 5%)
S = Manufacturer's Spec. (contingency = 1%)
A = Actual Mass (contingency = 1%)

Rev A 06/29/88
Rev B 07/09/88

Item Name	Basis	Rev	Mass, ea (kg)	Qty.		Mass Totals (kg)	
A. Payload							1070
1. Crew	E	A	75	2		150	
2. Spacesuit & MMU	E	A	100	2		200	
3. Couch	E	A	10	2		20	
4. User Payload (and/or radiation shield option)						700	
a. Left on Phobos	E	B	700		700		
b. Returned Payload (200 kg)			0.00		0		
B. LSS							460
1. Oxygen	E	A	1.00	40		40	
2. Water	E	A	4.00	40		160	
3. Food	E	A	2.00	40		80	
4. Air Purification	E	A	30			30	
5. Thermal Control	E	A	100			100	
6. Waste Management	E	A	50			50	
C. Structure							570
1. Outer Shell	E	A	300			300	
2. Storage Structure	E	A	150			150	
3. Insulation	E	A	60			60	
4. Window	E	A	10	2		20	
5. Hatch	E	A	20			20	
6. Panels and Supports	E	A	20			20	
D. Propulsion							6841
1. Orbit Change						6741	
a. Engine	E	A	100		100		
b. Fuel Tanks	E	B	604		604		
c. Propellant (NTO/MMH)	E	B	6037		6037		
2. Fueled ACS	E	A	100			100	
E. Power							548
1. Batteries	E	B	548			548	
F. Equipment							305
1. Controls and Displays	E	A	60			60	
2. Communication	E	A	50			50	
3. Guidance and Navigation	E	A	100			100	
4. Docking Provisions	E	A	60			60	
5. Other Instrumentation	E	A	35			35	
						Total Mass (kg) =	9794

Fig. 1.1.2.1-5 Earth Transfer Vehicle with TEIS

(sprint class mission)

Dry Mass	
(includes payload)	67,960 kg
Payload Mass	
(ETV)	59,070 kg
Propulsion System	
Propellant Type	LOX/LH_2
Engines	
Number	3
Type	RL10B-2
Mass (ea.)	191 kg
Thrust (total)(66.1 klbf)	294 kN
I_{sp} (460 sec)	4.51 kN-s/kg
Propellant Mass	51,890 kg
Initial T/W	0.67
Mass Fraction	
Earth return	0.57
Total Mass	119,850 kg

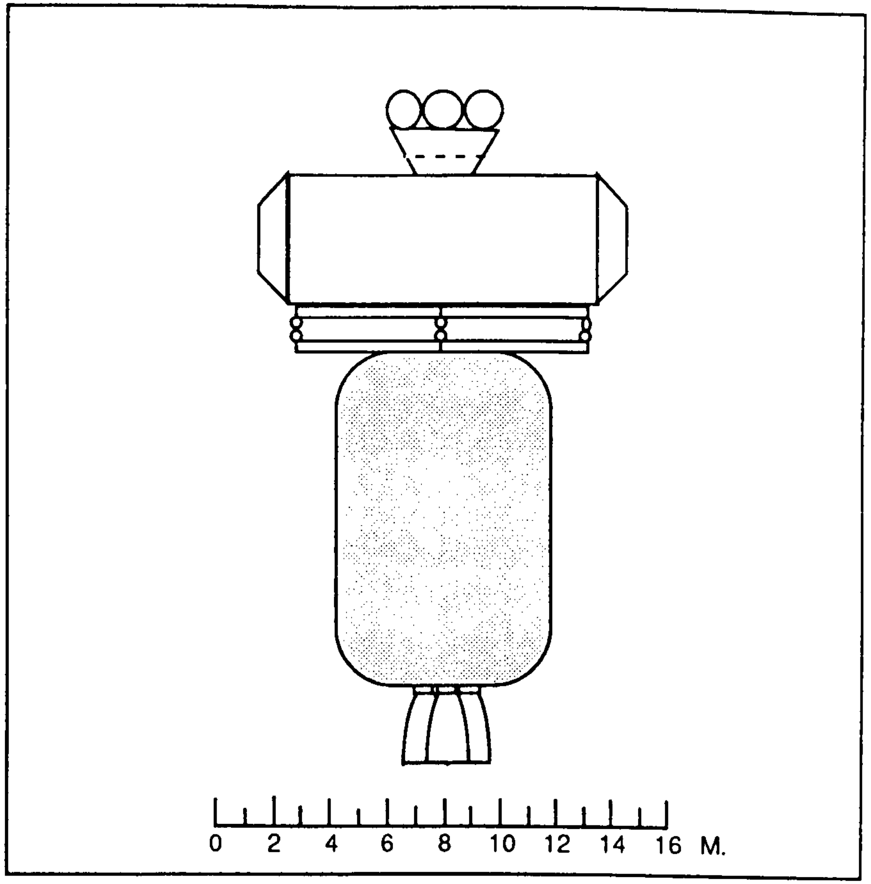


Fig. 1.1.2.1-6a Trans-Mars Injection System (TMIS)

Dry Mass	
(includes payload)	499,850 kg
Payload Mass	
(MTV)	377,030 kg
Propulsion System	
Propellant Type	LOX/LH_2
Engines	
Number	1
Type	SSME-derivative
Mass (ea.)	3,175 kg
Thrust (total)(543 klb_f)	2,415 kN
I_{sp} (480 sec)	4.71 kN-s/kg
Propellant Mass	811,490 kg
Initial T/W	0.19
Mass Fraction	
Total	0.38
Total Mass	1,311,340 kg

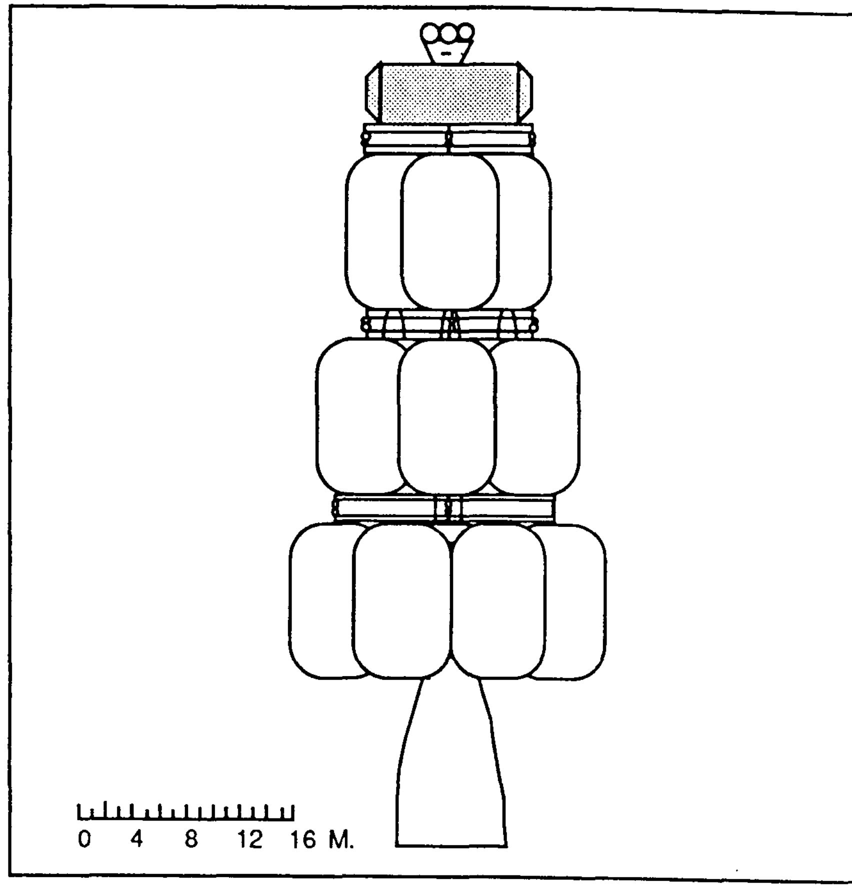


Fig. 1.1.2.1-6b
Expanded View of (MTV) with TMIS

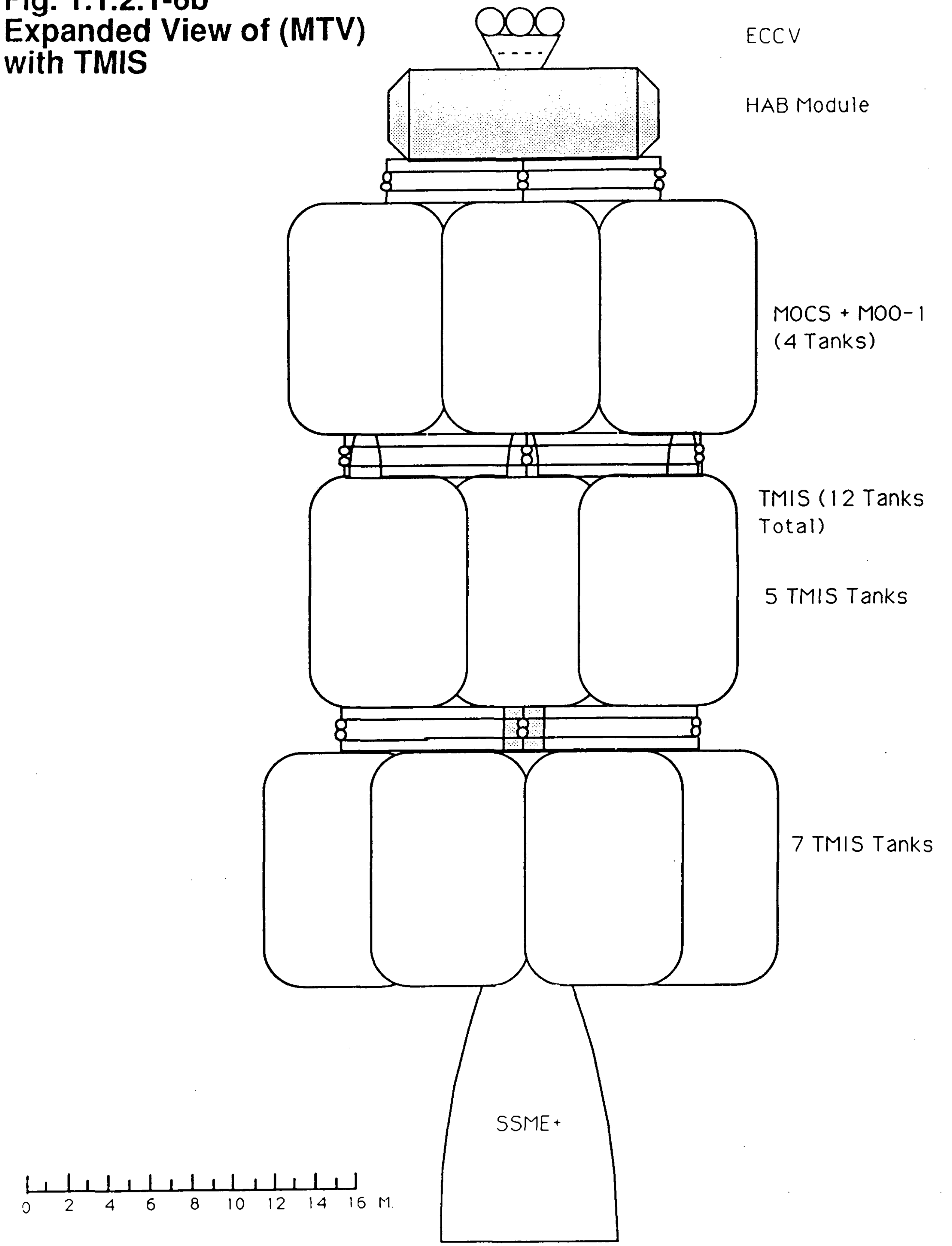


Table 1.1.2.1-4 Departure Energetics—Case Study 1 Astrodynamics

	TMI Dates		C_3 (km^2/s^2)			ΔV (km/s)		
			Earth Departure		Mars	Earth Departure		Mars
	Cargo	Piloted	Cargo*	Piloted	Departure	Cargo*	Piloted	Departure
Nominal	2-01	8-02	7.9	27.2***	25.0**	3.52	4.35	3.71**
Earlier	2-99	5-00	8.4	37.2	18.1**	3.54	4.76	3.08**

* With A/C at Mars
** From Phobos orbit
*** Includes Venus swingby

Table 1.1.2.1-5 Encounter Energetics—Case Study 1

	Launch Dates		C_3 (km^2/s^2)		
			Mars Arrival		Earth
	Cargo	Piloted	Cargo	Piloted	Arrival
Nominal	2-01	8-02	15.1	49.8	15.5
Earlier	2-99	5-00	29.8	50.0	25.0
Fallback	6-03	10-04	7.3	43.2	25.0

Figure 1.1.2.1-7a. Cargo Vehicle (with TMIS and MOCS)

Dry Mass (includes payload)	204,140 kg
Payload Mass (MTV)	163,610 kg
Propulsion System	
Propellant Type	LOX/LH_2
Engines	
Number	1
Type	SSME-derivative
Mass (ea.)	3,175 kg
Thrust (total)(543 klbf)	2,415 kN
I_{sp} (480 sec)	4.71 kN-s/kg
Propellant Mass	262,830 kg
Initial T/W	0.53
Mass Fraction	
Total	0.56
Total Mass	466,970 kg

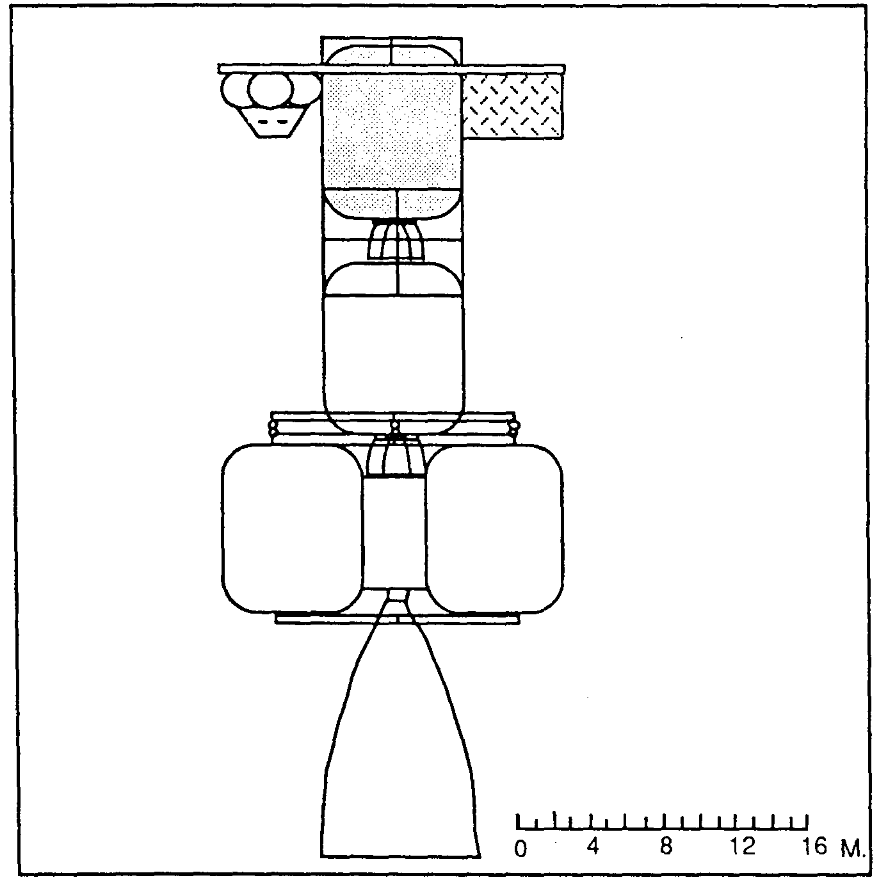


Fig. 1.1.2.1-7b Expanded View of MCV with MOCS and TMIS

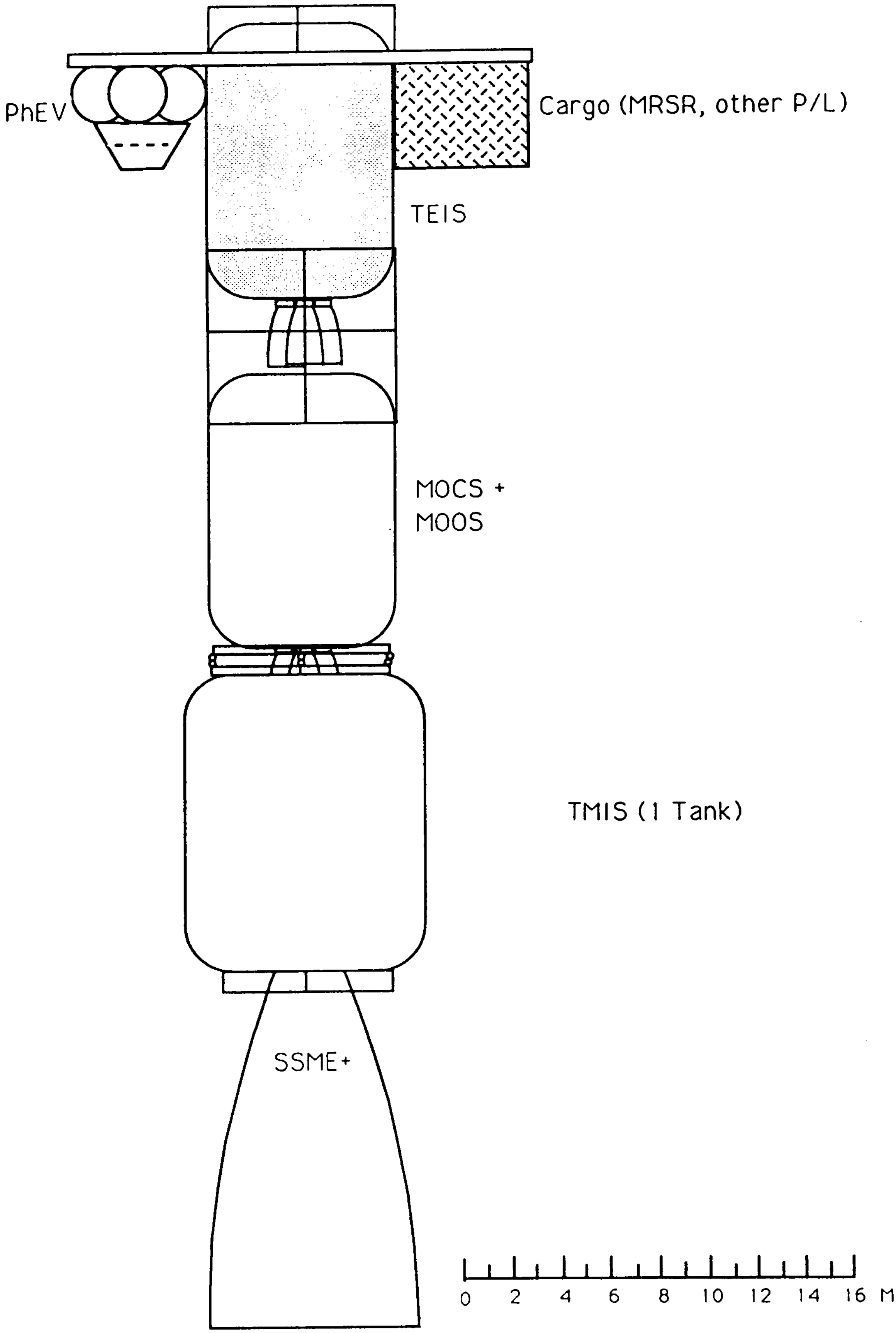


Table 1.1.2.3-1
ETO Manifest -- Case Study 1 [TIC-1R m-FD] (Baseline all-prop.)

Mars Cargo Vehicle (MCV) Components Launches for Assembly in LEO

Launch	Date	Item	Mass
1. HLLV-1	July '00	Mars Transfer Vehicle (MTV), PhEV	35.2 t
2. HLLV-2	Aug '00	MOCS & MOOS	73.5 t
3. HLLV-3	Sept '00	TMIS tank #1	91.0 t
4. HLLV-4	Oct '00	TMIS tank #2	91.0 t
5. HLLV-5	Nov '00	TMIS tank #3	91.0 t
6. HLLV-6	Dec '00	TMIS tank #4 (partial tank plus TMIS engine, PA) (includes 10 t reserve for top-off prop.)	86.6 t
7. HLLV-7	Jan '01	TEIS (fully loaded)	72.1 t
8. STS-1	Dec '00	Crew for teleoperated assembly of * TMIS * integration of MTV with TMIS	

Teleoperated docking of TEIS and MCV from Earth. Launch Feb '01 via Earth command.

		Total	540.4 t

Table 1.1.2.3-1 (cont.)
ETO Manifest -- Case Study 1 [TIC-1R m-FD] (Baseline all-prop.)

Mars Spaceship (MSS) Components Launches for Assembly in LEO

Launch	Date	Item	Mass
9. HLLV-8	Mar '01	MTV, minus MOCS and MOO	78.2 t
10. HLLV-9	Apr '01	MOCS + MOO tank #1 plus engine	91.0 t
11. HLLV-10	May '01	MOCS + MOO tank #2 plus engine	91.0 t
12. STS-2	May '01	Crew #1; initial boarding and checkout	

Crew telerobotically assembles structure, attachment of MTV to MAb
Crew inhabits MTV (4 months)

Launch	Date	Item	Mass
13. HLLV-11	Jun '01	MOCS + MOO tank #3	91.0 t
14. HLLV-12	Jun '01	MOCS + MOO tank #4 plus engine	72.0 t

Table 1.1.2.3-1 (cont.)
ETO Manifest -- Case Study 1 [TIC-1R m-FD] (Baseline all-prop.)

Mars Spaceship (MSS) Components Launches for Assembly in LEO

Launch	Date	Item	Mass
15. HLLV-13	July '01	TMIS tank #1	91.0 t
thru Monthly thru Apr, thru tank #11			91.0 t for 11
27. HLLV-25	Apr '02	TMIS tank #12 plus engine (includes 10 t for top-off of TMIS, MOCS)	81.0 t
		Total	1505.2 t
28. STS-3	Sept '01	Changeout to crew #2	
29. STS-4	Jan '02	Deliver final assembly hardware and crew #3 Return crew #2 to ground	

Specialized crew assembly and checkout of TMIS. Dock ECCV. Top-off tanks.

Launch	Date	Item	Mass
30. STS-5	Jun '02	Crew changeout to flight crew (#1, #2, or #3) Bring up any special final assembly hardware and replacement items	

Flight crew integrates TMIS with MSS. Final checkout. Launch Aug '02 by flight crew.

Table 1.1.2.3-2 ETO for Large and Magnum HLV

Assumed capacity: 200 t to LEO

MCV

1. MTV	163.6 t
2. TMIS, partial	200.0
3. TMIS, complete	103.4
Launch	

Piloted Vehicle

4. MTV	124.4
5. MOCS, partial	200.0
6. MOCS complete plus partial TMIS	200.0
7 thru 10. TMIS bal.	200
Launch of piloted vehicle.	

Table 1.1.2.3-3. Minimum ETO Yearly Profile for Case Study 1 (not including STS launches).

	2000	2001	2002	Total
Number of Launches				
Baseline HLLV (91 t)	6	12	7	25
Very Large HLLV (200 t)		3	7	10
Magnum HLLV		1	1	2
Aerocapture (91 t HLLV)	4	5	3	12
Mass (t)				
Baseline HLLV (91 t)	468.3	1041.3	535.9	2045.5
Very Large HLLV (200 t)		591.4.	1200.0	1791.4
Magnum HLLV		467.0	1311.3	1778.3
Aerocapture (91 t HLLV)	300.1	462.6	125.8	888.5

Title: **Fig. 1.1.3-1. Transportation Program Development Schedule**

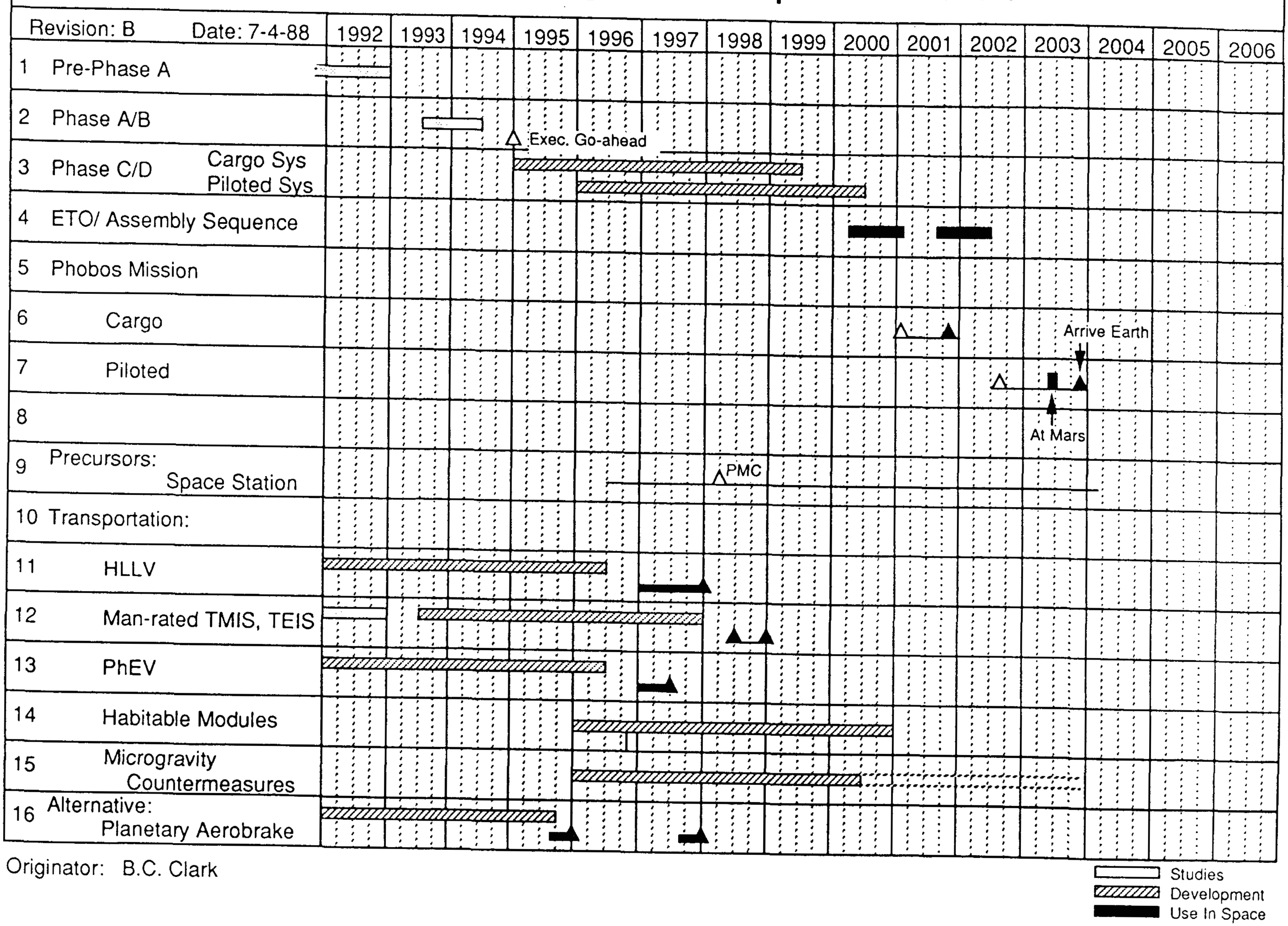


Table 1.1.4-1 Trades/Options—Case Study 1

Options

a	Next earlier opportunity (launch 2000 for piloted)
b	Next later opportunity (launch 2004 for piloted)
c	Conservative propulsion performance (e.g., 471/460/320)*
d	Advanced propulsion system performance (e.g., 485/470/340)
e	Smaller user accommodation (0.2 t Ph, 4.8 t MRSR)
f	Venus probes (3t)
g	v. conserv. LH_2/LOX propulsion (LTCSF wet-launched tanks, high boiloff)
h	nom. tankage, but high boiloff combination
i	Hydrocarbon/LOX for TEI
j	Both MOV to Phobos circular orbit (sub-scenario C)
k	Piloted MOV to Phobos circular, MCV in 250 x Phobos elliptical (sub-scenario A)
l	EOC of ETV
m	Two-stage TMIS
n	very conservative aerobrake (15%)
o	advanced aerobrake (5%)
p	TEI, MOC (if all-prop) using NTR
q	I_{sp} of 900 for NTR
r	ECCV retro-propulsion via reduction of C_3
s	Ph Teleoperator in lieu of PhEV (no humans *at* Phobos)
t	Without Mars high circular orbit maneuver
u	Advanced tankage technology (0.075)
w	miminum crew

* TMI/TEI , MOC, MOO/biprop

Table 1.1.4-2 Options Mass Summary—Case Study 1

TIC i.d. Concept	Trade/Option	IMLEO Hum	IMLEO Car	IMLEO Total	File Miss	File Ref
TIC-1R	Reference (all-prop.480/460/320; nom tank (0.15); low boil; separate PhEV, 9.8 t; 7 t MRSR; ECCV direct entry, 619m/s)	1311.3	467.0	1778.3	m-FD	m-DG
TIC-1R-h	high boiloff	1676.8	589.2	2266.0	m-FH	m-DH
TIC-1R-h2	no boiloff	1196.1	430.7	1626.8	m-FI	m-DG
TIC-1R-a	Earlier Mission (2000)	3888.3	487.4	4375.7	m-GE	m-FS
TIC-1R-b	Later Mission (2004) ("fallback" mission)	2047.6	316.5	2364.1	m-GF	m-FT
TIC-1R-m	2-stg TMIS	1200.8	467.0	1667.8	m-GC	m-FD
TIC-1R-d	adv. propulsion (485/470/340), 1-stg TMIS	1227.1	442.0	1669.1	m-FK	m-DI
-1R-dm	add 2-stg TMIS to preceding	1126.8	442.0	1568.8	m-FL	m-DK
-1R-hc	conservative propulsion (471/460/320); high boiloff	1727.1	603.3	2330.4	m-FO	m-DL
-1R-hdm	2-stg TMIS, adv prop., high boiloff	1411.3	554.3	1965.6	m-FP	m-DM
TIC-1R-hj	Both MOV's into Phobos circular orbit; high boiloff	5530.5	2032.0	7562.5	m-FQ	m-DN
TIC-1R-j	Both MOV's into Phobos circular orbit; low boiloff	3029.6	1294.7	4324.3	m-GH	m-FQ
TIC-1R-ha	Earlier Mission (2000), high boiloff	5242.7	624.2	5866.9	m-FS	m-DO
TIC-1R-hb	Later Mission (2004), high boiloff	2536.2	372.9	2909.1	m-FT	m-DP
TIC-1R-hg	very conservative tanks (0.206), high boiloff	3328.5	925.3	4253.8	m-FU	m-DQ
TIC-1R-d2	Adv. prop. (485/460/320)	1291.2	461.2	1752.4	m-FV	m-EH
TIC-1R-d3	Adv. prop. (480/470/320)	1252.8	449.7	1702.5	m-FW	m-EI
TIC-1R-u	Adv. technol. cryoprop (.075 tankage)	919.2	353.1	1272.3	m-FX	m-EJ
TIC-1R-u1	Adv. technol cryoprop (.075 tankage), no boiloff	852.0	328.2	1180.2	m-FY	m-EK

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Figure 1.1.4-1 Case Study 1

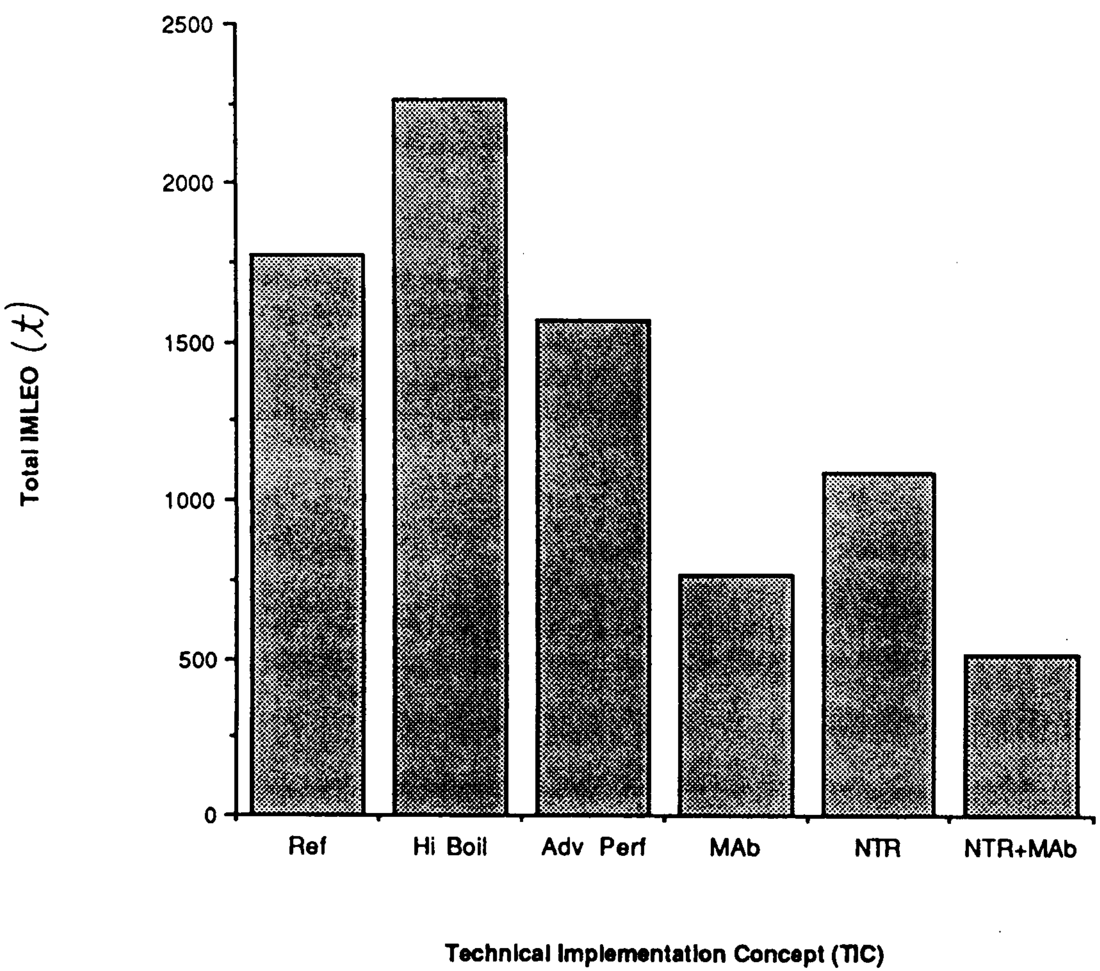


Table 1.1.4-3 Launch Manifest Comparisons for Case Study 1: Phobos

HLLV Cabability	Split Sprint '02 All Propulsive	Split Sprint '02 with Aerobrake	'02 Venus Swingby Sprint (440 day)	'04 Venus Swingby Opposition (660 day)
91 t	4 Dry (w/Assembly) + 20 Tankers	3 Dry (w/Assembly) + 9 Tankers	2 Dry (w/Assembly) + 8 Tankers	2 Dry (w/Assembly) + 4 Tankers
182 t	2 Dry (w/o Assembly) + 10 Tankers	2 Dry (w/o Assembly) + 5 Tankers	1 Dry (w/o Assembly) + 4 Tankers	1 Dry (w/o Assembly) + 3 Tankers

7/22/88

Table 1.1.4-4 Phobos Exploration Sub-Scenarios

Sub-Scenario A *(Option j)*

Cargo MOV captures into parking orbit ellipse of 250 km x Phobos
- Deploy MRSR+Network descent vehicles. Rotate plane as required.

Cargo MOV circularizes into Phobos rendezvous orbit
Crew MOV into Phobos circular orbit (via Phobos elliptical orbit)
Transfer of TEIS
Exploration of Phobos. (Teleoperator to Deimos)
MAV samples to Phobos orbit
TEI from Phobos orbit

Sub-Scenario B *(Reference Mission)*

Cargo MOV captures into min inclination HMO (250 km x 1 sol)
- Deploy MRSR+Network descent vehicles.

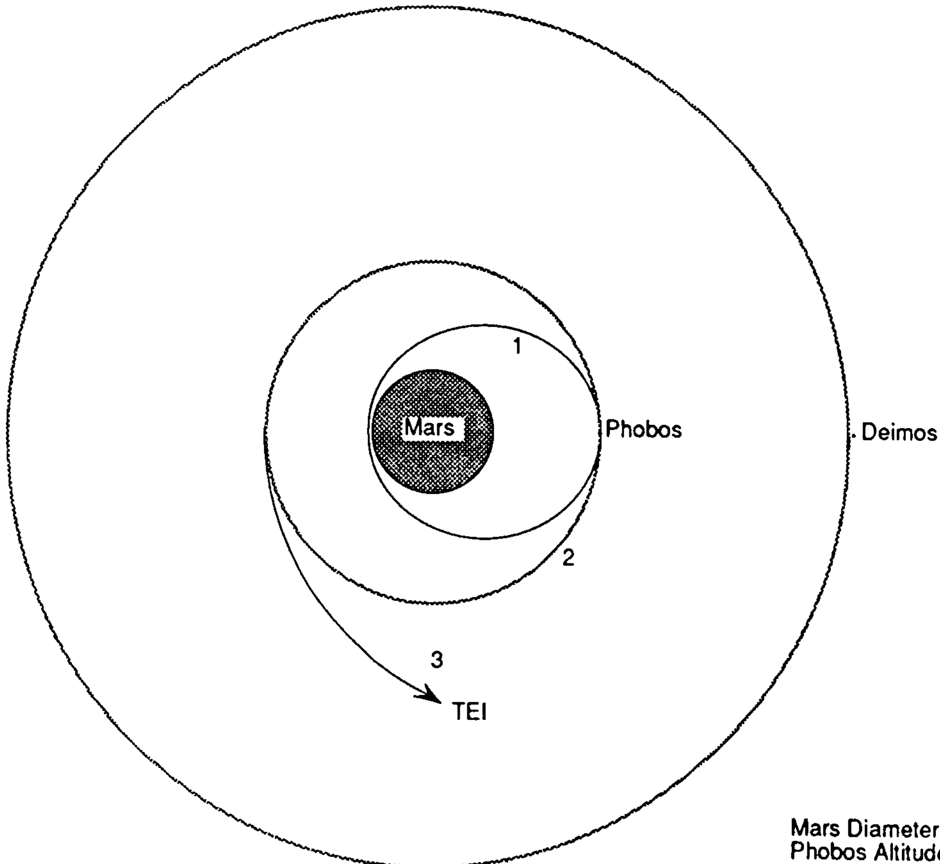
Crew MOV into same orbit
Transfer of TEIS. Transfer of Phobos Excursion Module (PhEV)
Manned excursion to Phobos and return to crew MOV. (Teleoperator to Deimos)
MAV samples to HMO
TEI from HMO

Sub-Scenario C *(Option k)*

Cargo MOV captures into parking orbit ellipse of 250 km x Phobos
- Deploy MRSR+Network descent vehicles. Rotate plane as required.

Crew MOV into same orbit
Transfer of TEIS
Crew vehicle circularizes into Phobos orbit
Manned exploration of Phobos. Teleoperator to Deimos.
MAV samples to Phobos orbit
TEI from Phobos orbit

Figure 1.1.4-2 Sub-Scenarios A and C*



Mars Diameter = 6,794 km
Phobos Altitude = 5,981 km
Deimos Altitude = 20,062 km

* Cargo vehicle (MCV) remains in orbit 1 in Sub-scenario C

Table 1.1.4-5 Phobos Parking Orbit—Trade Considerations

Criterion	Trade Results
IMLEO	Sub-scenario B is superior to A and C
Navigation	Phobos rendezvous point allows strong tie-point (optical-nav backup for long range navigation)
Hardware development	Sub-scenario B requires development of PhEV
Deimos exploration	PhEV can be used as a DeEV for manned exploration Deimos teleoperators possible for all sub-scenarios

Table 1.3-1 Alternative Missions—Case Study 1

R Reference mission
- Split/sprint. All-propulsive (480/460/320).
- Siamese tank sets
- User accommodation: 1.2 t to Phobos, 7 t MRSR.
- Separate vehicles for PhEV, ECCV.

A Mars Aerobrake (MAb) for MOC
- 10% aerobrake (nominal aerobrake technology)

B Nuclear Thermal Rocket (NTR)
- 850 I_{sp}; TMIs only

Table 1.3-2 Mars Aerobrake (MAb) Design—Case Study 1

Assumptions: $C_{3(\text{Mars arrival})} = 50\ km^2/s^2$; $M = 90\ t$

Aerocapture brake characteristics:

Diameter	23.8 m [78 ft]	(Area = 444 m^2 [4776 ft^2])
W/C_dA	124 kg/m^2 [25.4 lb_m/ft^2]	
Angle of attack (alpha)	10.25°	
L/D	0.165	
Peak deceleration	7.85 gee	

Mass Summary

RSI Core (33' dia, 0.73" thick)		349 kg
RSI Honeycomb substrate		1161
Interface ring		985
Radial Beams		477
Struts		1084
FSI Annulus (80' dia, 1.11" thick)		3053
	Sub-Total	7109
	Contingency (20%)	1422
	Total	8531 kg (9.5%)

Table 1.3.-3 Alternatives Mass Summary—Case Study 1

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TIC i.d. Concept	Alternatives	-------- IMLEO -------- Hum	Car	Total	----- File ----- Miss	Ref
TIC-1R	Reference (all-prop.480/460/320; nom tank (0.15); low boil; separate PhEV, 9.8 t; 7 t MRSR; ECCV direct entry, 619m/s)	1311.3	467.0	1778.3	m-FD	m-DG
TIC-1A	Mars aerobrakes (MAb's) for piloted and cargo	452.9	311.7	764.6	m-ES	m-DU
TIC-1A-b	Next later opportunity	817.5	275.1	1092.6	m-ET	m-DW
TIC-1A-f	Venus probes	463.7	311.7	775.4	m-GD	m-ES
TIC-1A-m	2-stg TMIS	416.0	311.6	727.6	m-EU	m-DX
-1A-bm	2-stg TMIS, next opportunity	737.9	275.1	1013.0	m-EV	m-DY
-1A-dm	adv. prop. (485/470/340), 2 stg	403.0	298.6	701.6	m-EW	m-DZ
-1A-fdm	Venus probes, plus preceding	412.7	298.6	711.3	m-EX	m-EA
-1A-gh	v. conserv. LH2/LOX propulsion (0.206 tanks), high boil	717.2	538.8	1256.0	m-EY	m-DU
-1A-hi	Hydrocarbon/LOX for TEI (395), high boil	454.9	320.0	774.9	m-EZ	m-DU
-1A-u	adv technol. cryoprop (tankage 0.075)	367.5	248.9	616.4	m-FZ	m-ES
-1A-ul	adv technol. cryoprop (tankage 0.075), no boiloff	357.5	234.8	592.3	m-GB	m-ES
TIC-1B	Nuclear Thermal Rocket (NTR) for TMI only	771.1	317.4	1088.5	m-FE	m-EB
TIC-1B-p	all propulsive, TMI for all (TMI, DSM, MOO, TEI)	538.2	314.0	852.2	m-FF	m-EC
TIC-1B-ph	same, but high boiloff	1004.9	555.8	1560.7	m-FG	m-ED
TIC-1AB	NTR+MAb	284.2	225.9	510.1	m-FA	m-EE
-1AB-h	NTR+MAb, high boiloff	321.4	276.3	597.7	m-FB	m-EF
-1AB-p	NTR for TEI also	281.9	214.8	496.7	m-FC	m-EG
1-AB-hu	NTR+MAb, adv cryoprop (tankage 0.075), high boiloff	302.6	244.6	547.2	m-GI	m-FB

Human Expedition to Mars

2.0 Transportation Systems Definition

The transportation for Case Study 2 consists of a series of three split missions to Mars, including Mars Cargo Vehicles, Mars Spaceships (MSS), Mars Descent Vehicles (MDV), PhEVs, and a Deimos Excursion Vehicle (DeEV). Each MDV includes within it a Mars Ascent Vehicle (MAV) for return of the landed astronauts to the Mars Orbiting Vehicle (MOV). Also utilized in this case study are TEIS and aerocapture ECCVs.

2.1 Elements and Systems Description

2.1.1 Transportation Requirements/Assumptions

The missions involve the launch of 8 astronauts on-board Mars Spaceships in November of 2004, December of 2006, and in February of 2009. Each manned launch is preceded by an appropriate cargo mission. Because they are split missions, only 30 days staytime at Mars is provided. On each mission, 4 crewmembers descend to the surface for up to 20 days of exploration. On the second mission, a Phobos exploration is also performed. On the third mission, both Phobos and Deimos are also explored. These three missions are futher detailed in Table 2.1.1-1 and Figure 2.1.1-1.

Other requirements and assumptions are given in Table 2.1.1-2 and Figure 2.1.1-2. Boiloff rates are assumed as given in Table 1.1.2.1-1. The ECCV in this case does not perform a direct entry to Earth, but rather is aerocaptured into Earth orbit, with subsequent recovery of the astronauts and transfer to the Space Station for isolation prior to return to Earth. The same orbital apsidal adjustments described in section 1.1.1 are incorporated into the mission profiles of Case Study 2.

2.1.2 Reference System Description

2.1.2.1 Configuration and Mass Allocations

Standardized tanks are the same as in Case Study 1. The IMM (habitability package) for the Humans to Mars scenario is selected as the "Hub-Triangle" configuration, Figures 2.1.2.1-1a and -1b, which is made up of three space station derivative modules with an additional central unit 7.6 m in diameter by 3.0 m tall (disk module, 25 ft dia x 10 ft). Three independent entry points are available to each of the four modules. Two separate airlocks are provided, one of which is rated as a hyperbaric lock. A docking port is located on top of the disk module. The ECCV is mounted at one of the intermodule connection tunnels for continuous access. The entire structure is arrayed in a planer configuration and supported by trusswork to the Mars aerobrake (MAb) and is called the MTV, or Mars Transfer Vehicle. Two solar flare radiation storm shelters, each accommodating 4 persons (as described in section 1.1.2.1) are provided, one in module 1 and the other in module 3. A closed-cycle life support system is provided. A total of 18.3 t of consumables supply the crew, of which 5.55 t is food. The IMM, including external services, is 65.9 t. Power is provided by three independent solar arrays, providing a total of 19.5 kW_e at Mars and higher power levels elsewhere. The Mars aerobrake is 27.4 m diameter (90 ft dia) and consists of a hard inner core (33 ft dia) based upon Shuttle tile technology, with an outer annulus of flexible thermal insulation (Nextel ceramic cloth). It is expected that the MAb will be initially launched in a folded configuration. The MAb characteristics and mass summary are given in Table 2.1.2.1-1.

The cargo vehicle (MCV), Fig. 2.1.2.1-2, is similar in appearance to the human-carrier vehicle because of its three cylindrical TEIS tanks arranged in a triangular configuration. Although the aerobrake needed for this vehicle is much smaller, the same size brake as for the manned vehicle is shown to provide for commonality in design.

After launch into LEO both vehicles are equipped with a TMIS to boost the craft towards Mars. The assembled spacecraft comprise the Mars Cargo Vehicle (MCV) and the Mars Spaceship (MSS). The departure energetics are given in Table 2.1.2.1-2. Figure 2.1.2.1-3a and -3b (expanded view) show the MCV with its attached TMIS, and figures 2.1.2.1-4a and -4b (expanded view) depict the MSS. During the journey, after the TMIS has been jettisoned, the MTV will extend solar arrays from each module to provide power (Fig. 2.1.2.1-5a and -5b).

Mars Orbital Operations begin after the aerocapture of the MTV into Mars orbit (encounter energetics are given in Table 2.1.2.1-3). First, rendezvous with the MCV occurs (Fig. 2.1.2.1-6), followed by the docking and transfer sequence (Figs. 2.1.2.1-7 and 2.1.2.1-8, respectively). During this sequence the MCV transfers the TEIS to the piloted vehicle (now termed the MOV for Mars Orbiting Vehicle) and four crewmembers transfer to the Mars Descent Vehicle (MDV), which nestles inside the TEIS triangle. The TEIS is described below and the MDV is portrayed in Figure 2.1.2.1-8a, -8b, and -8c. The MDV includes a disk module the same size on the MTV (25 ft dia) which serves as the landed habitat (Fig. 2.1.2.1-9a and -9b). Mars entry and landing is accomplished by aerobraking to a velocity of mach 2 or less (see Mars Aerocapture Parametric Data, Fig. 2.1.2.1-10), deployment of parachutes, and ignition of a bipropellant-based terminal descent propulsion system to provide for a soft-landing and up to 1.0 km of cross-range for terminal guidance. Three Delta engines are provided for terminal descent. Dual unpressurized rovers are included for surface exploration. Life support is based upon bulk supplies of water and oxygen (stored as hydrogen peroxide) and chemical removal of carbon dioxide. The ECLSS is summarized in Table 2.1.2.1-4. For the 20 day surface mission, a total LSS mass of 1.26 t is allocated. The MDV mass is 51.3 t, of which 23.1 t is the Mars Ascent Vehicle (MAV) (Figs. 2.1.2.1-11a and -11b). The MAV is a minimum-mass conical spacecraft which holds 4 persons, their spacesuits, and 100 kg of returned samples. Its propulsion system includes 19.4 t of storable bipropellant. Four pressure-fed Delta engines provide the thrust required for lift-off and burn to high elliptical orbit for rendezvous with the MOV (Fig. 2.1.2.1-12). The MAV is then discarded and the MOV is ready to prepare for the journey back to Earth (Figs. 2.1.2.1-13a and -13b).

The Trans-Earth Injection System (TEIS) for Mars orbit escape consists of three tanks with a total of 59.5 t of cryopropellant (Fig. 2.1.2.1-14). Six RL10-derivative engines are provided in a dual triangular array. Only one triad of engines need operate nominally to provide the requisite thrust and thrust-vector alignment during Mars orbital escape. The initial acceleration at Mars departure burn is 0.275 gee.

The human vehicle, the MTV, consists of a number of standard tanks with one or more advanced space engines having specific impulse performance of 485 s. Tank number varies with the launch opportunity. For the 2004 opportunity, ten tanks are required, while only four standard tanks are needed for the associated cargo vehicle launch. MCV payloads include not only the TEIS and MDV, but also satellites totalling 4 t, a 3.5 t MRSR, and 0.45 t of on-board solar monitoring and Mars science. Later missions also carry PhEVs and in the third mission also a DeEV.

The ECCV accommodates 8 persons. Its mass is estimated at 9.2 t (Figs. 2.1.2.1 -15a and -15b). It is similar to the Apollo system of a conical capsule and a cylindrical service module portion, but a separate aerobrake is added to facilitate aerocapture. Adequate propulsion to achieve a periapsis-raise after aerocapture is provided in the service module. The PhEV is as described in section 1.1.2.1. The DeEV is nearly identical except for minor modifications to handle the slightly different propulsion requirements.

2.1.2.2 Features of the System

The habitability modules and TEI propellant modules are all compatible with Shuttle-C 24-ft diameter payload envelopes. The disk module is compatible with Shuttle-C or ET Aft-cargo Carrier concepts. Both the MCV and MTV are based upon triangular structures, providing the strongest natural structural design. Hexagonal docking trusses allow double-redundant tripod connection points when the two vehicles dock. The TEIS is a fully contained propulsion system commanded by remote radio link. It is capable of being powered by fuel cells and/or solar array. Engines are fully redundant. Transfer of the TEIS from the MCV to the MTV requires only a mechanical docking; no plumbing or electrical connections need cross this mechanical interface.

The combination of cylindrical and disk modules allows the development of an interior architecture which provides a quality of living appropriate to long duration stays in deep space. Multiple entries enable sealing off any one module without restricting access to the other modules. These features are summarized in Table 2.1.2.2-1.

2.1.2.3 ETO, On-orbit Assembly, and Servicing Needs

The number of HLLV launches required depends not only on HLLV lift capability, Table 2.1.2.3-1, but also on the launch year because of the strong variations in astrodynamical factors from one opportunity to the next for sprint class missions. Total IMLEO for the three mission opportunities are 1627.8, 2511.5, and 2625.1 t (see ETO manifest , Table 2.1.2.3-2, and Fig. 2.1.2.3-1). All habitable modules, standard propellant tanks, and the TEIS propulsion units can be accommodated by 25-ft diameter payload shrouds. The system may be assembled as a stand-alone, or at the Space Station node. Early-on habitation is possible as soon as the aerobrake, the disk module, and one cylindrical module (and its associated photovoltaic power array) are placed into orbit and assembled. In-space propellant transfer is not necessarily required, but because of the large number of launches and the possible stretchouts in assembly time in LEO, top-off propellants from a propellant depot or additional standard tank is desirable. Orbital debris hazards may be mitigated by use of the aerobrakes as forward shields during vehicle buildup.

2.1.3 Transportation Program Development Schedule

The schedule for development, proof-flight testing and man-rating of transportation hardware and propulsion systems is shown in Fig. 2.1.3-1. As in the previous case study, a number of prior developments are key to the success of this program. It will be critically important to achieve early developmlent of the HLLV, TEIS, and MAV because the capabilities of these transportation systems will affect the derivation of requirements for all other transportation vehicles.

2.1.4 Trades/Options

Several options have been considered (Table 2.1.4-1) and their effect on the total IMLEO has been calculated (summarized in Table 2.1.4-2 and Fig. 2.1.4-1). For the first mission launch opportunity, the use of a more realizable TMI engine performance of 480 s (down from 485) causes only a 1.3% increase in IMLEO. Backing down the TEI and other non-TMI propulsion performance from 470 to 460 s results in an additional 2.4% mass penalty. Cryopropellant storage issues affect IMLEO more profoundly. Use of very conservative tankage factors (dry tank mass/propellant mass) and high boiloff rates would cause an increase in IMLEO of over 50%, whereas advanced tankage (7.5% factor) can allow an 18% reduction in initial mass to low Earth orbit. Deleting the manned excursions to Phobos and Deimos, but providing exploration spacecraft teleoperated by the orbiting crew saves less than 1% of IMLEO for the three missions (see TIC-2R-s of Table 2.1.4-2). However, manned visits to these moons will significantly complicate mission operations during the relatively short staytimes at Mars and also expose personnel to new hazards without the assurance of any more effective exploration than what could be accomplished with well-designed robotic freeflyers (Table 2.1.4-3).

Slipping all three missions to the next launch opportunity causes a very large increase in IMLEO requirements, from 6764.4 t for the baseline case to 9715.0 t for the slipped scenario. Substitution of a longer Opposition/Venus Swingby trajectory for the third mission changes this result, with a slight reduction (7%) in cumulative IMLEO for the three missions. Conjunction class trajectories are sufficiently equivalent that only minor changes in propellant loadings are necessary when launch opportunities are shifted.

2.2 Enabling Technology Needs

As with Case Study 1, both the HLLV and Space Station are required for enabling the transportation systems development. In this case, because of landing on the Martian surface, several additional technological developments are needed as well.

2.2.1 Propulsion engines

Development of an advanced cryogenic space engine or space-operated qualification of an SSME-derived engine will be required. Increased performance of the RL10 engines must be verified (the RL10B-2 may be an acceptable candidate). Techniques for long-term in-space storage of the RL-10s or equivalent must also be developed and tested.

2.2.2 Cryopropellant tankage

It is quite obvious from previous discussion that every effort should be made for advancements in cryopropellant storage and for minimizing the tankage mass fraction relative to propellant (the "tankage factor"). This includes consideration of advanced composites, removeable structures and shields, efficiency of large multi-layer insulation blankets, use of vapor cooled shields, and other options.

2.2.3 Mars Descent and Ascent Vehicles

These represent major developments. The MAV, in particular, should be developed and demonstrated very early to provide a solid basis for design of the MDV.

2.2.4 Mars Surface Power

Development of a deployable photovoltaic power array (PVPA) suitable for operation on the Martian surface is of high priority because life support and operations for 20 sols will require more power than can be supplied by storage. Issues include broad-range thermal cycling, dust interferences, and methods of deployment. This system will serve as prototype for longer surface stays.

2.2.5 Precursor Missions

Selected missions will be needed to provide spaceborne demonstration and verification of the MDV, MAV, ECCV, IMM, and propulsion systems. Sample return missions to Mars (MRSR) are a necessity because of concerns for toxicity (biological and/or chemical) and possible reactivity of Martian soil.

2.2.3 System Alternatives and Opportunities

Five alternatives to the reference mission are considered and are summarized in Table 2.2.3-1. The resulting mass changes are given in Table 2.2.3-2. For example, elimination of aerobraking for this mission results in very large mass increases -- doubling IMLEO for the first mission. Utilization of an NTR stage for TMI cuts the IMLEO by one-third. Replacing the cryochemical TEIS stage with a nuclear thermal stage results in only an additional 3% gain, however.

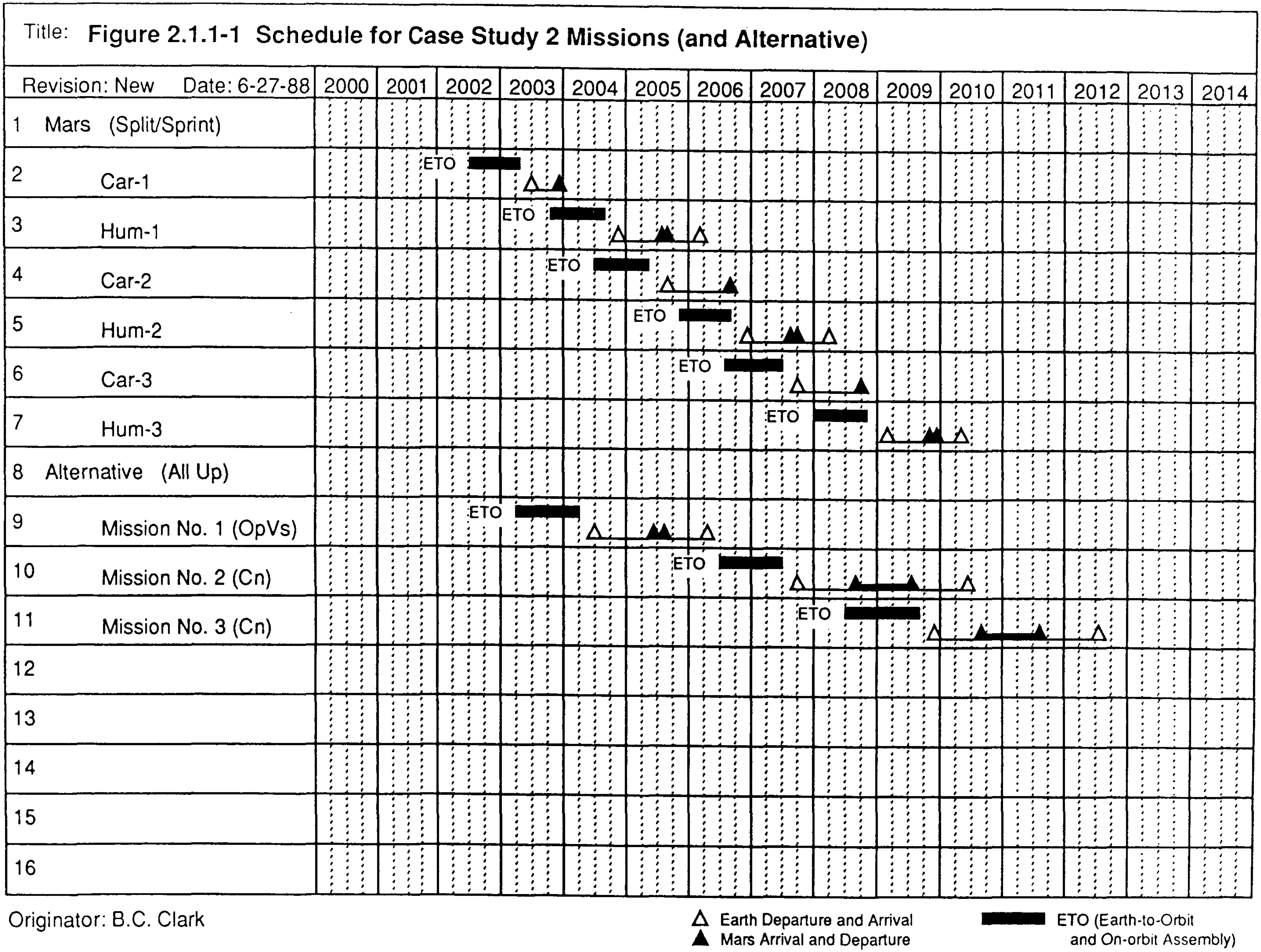
A major impact on IMLEO (up to 50%) has been the adoption of a strategy to correct the asymptote by circularization of the orbit at high altitude followed by re-establishment of the elliptical orbit. It is planned to examine alternate strategies for accomplishing these necessary adjustments without the mass propulsion penalties that are now being taken.

Conjunction class missions should be given strong consideration for this type of mission scenario because of their beneficial effect on lowering propulsion requirements as well as enormous increases (up to 25-fold) in time available for exploration at Mars. The effect on the total IMLEO is portrayed graphically in Figures 2.2.3-1 and 2.2.3-2. Also, these trajectories do not travel sunward in their initial stages as do opposition and sprint trajectories, thereby avoiding the higher thermal loads and increased solar flare radiation levels. The 2.2 to 2.5 times longer trip times of conjunction missions compared to sprints may call for a requirement for artificial gravity. Figures 2.2.3-3a through -3f show several views of one concept for a rotating spaceship. Four station-derivative modules are arrayed in the "Bent-I" configuration, connected by tunnels and with a 25-ft disk module at the center of rotation. For the 55-ft swing radius to the floors of the cylindrical modules, the acceleration is up to 0.675 gee (achieved at 6 revolutions per minute). Decreasing the rate to 4.5 rpm produces the Mars surface gravity of 0.38 gee, allowing adaptation by the astronauts to martian conditions prior to arrival at the planet (see Table 2.2.3-3).

Even allowing for the larger habitat and the larger diameter aerobrake (135 versus 90 ft), the savings in IMLEO are one-third for the first mission, and over 60% for subsequent missions. Yet pressurized habitation volume is increased from 737 up to 1271 m^3, Table 2.2.3-4, the astronauts live in a more Earth-like environment, more science payload is provided, and the solar cell array no longer has to be deployable/retractable but can be fix-mounted. The mission is "all-up", meaning no rendezvous in Mars orbit is required to obtain the return TEIS propulsion, since it is built into the spaceship. Humans arrive at Mars one and one-half years earlier, with virtually no change in programmatics, except a slightly earlier peak in funding. For a 20% increase in IMLEO, but still about 20% less than for the sprint case, this vehicle could carry *two* MDVs (Fig. 2.2.3-4). This would allow exploration of two different sites and nearly one year on the surface of Mars, while providing all crew members the opportunity to go to the surface and even permitting a rescue of the first landed crew by the second MDV, if it became necessary.

Table 2.1.1-1 Transportation Requirements -- Case Study 2

		--- Total ---		Launch		User Accommodation		---Surface ---		LEO
Mission		Crew	Time	Date	Destination	Mass (t)	Type(s)	Crew	Time	Mass (t)
CS-2	Car-1	--	790 d	6-03	Mars	15	MDV+G/M Lander(1)	--	--	<1150
	Hum-1	8	440 d	11-04	Mars	6	Orb Sci (Ph+De)	4	20 d	<1150
	Car-2	--	760 d	8-05	Mars	15	MDV	--	--	<1150
	Hum-2	8	440 d	12-06	Mars	12.5	MRSR(2)+GM(2)	4	15 d	<1150
							Phobos	2	5 d	
	Car-3	--	790 d	9-07	Mars	15	MDV	--	--	<1150
	Hum-3	8	440 d	2-09	Mars	12.5	MRSR(2)+GM(2)	4	10 d	<1150
							Phobos, Deimos	2	10 d	


Title: Figure 2.1.1-1 Schedule for Case Study 2 Missions (and Alternative)
Revision: New Date: 6-27-88
2000
2001
2002
2003
2004
2005
2006
2007
2008
2009
2010
2011
2012
2013
2014
1 Mars (Split/Sprint)
2 Car-1
3 Hum-1
4 Car-2
5 Hum-2
6 Car-3
7 Hum-3
8 Alternative (All Up)
9 Mission No. 1 (OpVs)
10 Mission No. 2 (Cn)
11 Mission No. 3 (Cn)
12
13
14
15
16
ETO
ETO
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ETO
ETO
ETO
Originator: B.C. Clark
Earth Departure and Arrival
Mars Arrival and Departure
ETO (Earth-to-Orbit and On-orbit Assembly)

Table 2.1.1-2. Transportation Requirements and Assumptions

SRD Requirements:

- *Split:sprint/conjunct* .
- On Mars surf, activate geophys and atmos long-term monitoring exper.
- SS or other LEO node used for assy. No nodes beyond LEO (2.2)
- 1 km landing accuracy (4.1.1)
- "Minimize the single major sys(s) that could cause to miss a launch period". EOC. (4.1.1)
- EVAs: four 6-hr at Phobos; 10 on Mars. Flyby aborts.
- User accom on flight veh: 100 kg, 1 kW. (4.1.1)
- Payloads (A.2): see User Accommodations above

Assumptions for Reference:

- 2-stage TMIS for piloted; 1-stage TMIS for cargo
- Propulsion: Cryo for TMI, TEI, DSM, MOO; biprop for MCC, MOC, RCS
  MAV is single-stage biprop.
- Engine performance: I_{sp}=485 TMI, 470 for other cryo; 320 for storable biprop
- Propellant margins: 1% each for ΔV, I_{sp}, and bulk (use sum of margins)
  3% ΔV margin on MAV; 2% bulk margin on TEI
- Hab modules: three SS-derived modules plus one disk module
  ("Hub-Triangle" configuration)
- PVPA for spaceborne power, 300 m^2
- Spaceborne ECLSS: closed for all, except food
- No Venus probes
- Mars aerobraking; ECCV for crew recovery at Earth
  Aerobrake technology: very conservative (15%) for piloted; nominal (10%) for cargo
- MOV Mars parking orbit: 250 km x 1 sol; Phobos excursion vehicle (PhEV)
- MDV entry and landing: biprop deorbit, terminal propulsion; aerobraking and parachutes
- MDV habitat: one 7.6 m [25'] diameter disk module
- Landed ECLSS: no O_2, CO_2 recycling; water recycling
- MAV direct to MOV parking orbit (ΔV = 5408 km/s)

Figure 2.1.1-2 Reference Mission Options

Scenario: CS-2 TIC-1R

Options Selected: (Date & Your Name)

OPTIONS: proven, or under development ---------- unproven, or must be developed / analyzed

Earth departure location	**LEO**	HEO	GEO	L1,L2				
On-orbit assembly	**SS, attached**	SS, free-flyer	Min. SS	no SS req'd.				
Hardware staging	integrated	**split, MDV**	split, MRSR	**split, TEIS**				
Trajectory type	flyby	conjunction	opposition	**sprint**	low thrust	cycler		
Launch dates	1990's	**2000's**	2010's	2020's	2030's			
Crew size, total	3	4	5	6	7	**8**	9-11	12-24
Cabin pressure	4.3 psi	**14.7 psi**	10.2 psi					
Gravity environment	**microgravity**	artificial, .38 g	artificial, 0.6 g	artificial, 1 g	hybrid			
Rotation rate	**0 rpm**	1 rpm	2 rpm	4 rpm	6 rpm			
Radiation protection	none	one storm shelter	**two storm shelters**	GCR shield				
Hab/Lab modules	**SS modules (15' dia.)**	**ET derived (25' dia.)**	large dia. (31' dia.)	inflatables				
Science equipment	**interplanetary**	**Mars orbit**	**Mars surface**	**Phobos**	**Deimos**			
ECLSS, spaceborne	consumables	**SS ECLSS**	**water recycling**	low mass/power	CELSS			
TMI launch propellant	hybrid chemical	**LH2/LOX**	Hc/LOX	waste/LOX	SEP	NTR	NEP	solar sail
engines, cryo.	RL-10 growth	F1 derived	**SSME derived**	advanced cryo				
growth	max.-sized tank	**stretch**	**cluster**					
reusability	**non-recoverable**	engines, avionics	all recoverable					
recovery method	**none**	turn-around	re-encounter					
Cryoprop storage	**passive**	active, refrigeration	active, reliquifaction	from H2O				
Power, spaceborne	**PVPA**	**fuel cells**	**RTGs**	nucl. reactor	DIPS	solar th.-dy.		
TM abort capabilities	**Mars swingby**	propulsive abort						
Mars orbit	LMO	8.2 hr. ellip.	**24.6 hr. ellip.**	Phobos	Deimos	GMO		
Mars orbit capture	propulsive braking	prop/aero hybrid	**aerobraking**					
Satellites Relay Com.	none	LMO	**24.6 hr**	12.3 hr	Molniya	GMO		
Mars Science Orbiters	none	polar	circ.	**elliptical**				
Unmanned Landers	none	penetrator(s)	rover(s)	sample return				
Ph/De teleoperators	none	Phobos	Deimos					

C-2

Figure 2.1.1-2 (cont.)

	OPTIONS (cont.)						
	proven, or under development	new developments					unproven, or must be developed / analyzed
Number of MDV's	none			one			two
Time on surface	0	1 wk.	3 wk.	6wk.	6 mo.	1 yr.	PMP
Crew size, landed	0	2	3	4	5		6-20
Propellants, MDV	LH2/LOX	Biprop		H2O2			CH4/LOX
MELS	de-orbit prop.	parachutes	aerobraking	terminal prop.	hover/translate		airbags
Landing hazard	large, safe areas			pinpoint landing			terminal H.A.
Power generation	RTG	fuel cells	PVPA	DIPS	nucl. reactor		solar th.-dy.
Power storage	batteries, Ni-H			Regen. FC			HEDRB
ECLSS, Mars landed	consumables	SS ECLSS		low mass/power	ISCP		CELSS
MLSE	RVR	analyt. eq.	geopys. pkg	meteorol. pkg.	biol. eq.		drill rig
MLOE mass	none	0.5 t	1 t	2 t	4 t	8 t	16 t
RVR, manned	none		unpressurized		press., 5 sol		press., 20 sol
ISPP	none	CO2	CO	H2O2	LOX		CH4/LOX
ISRU demos	water	food	H2O2	buffer gas	GOX	fertilizer	MLOX
Propellants, MAV	UDMH/N2O4	solids					ISPP
Recovery ETV	not recovered		Ab EOC		propulsive capture		prop/Ab hybrid
ECCV	none	Ab EOC	prop. capture		prop/Ab hybrid		direct entry
Orbital retrieval	STS to SS		STV to SS				STS to Earth

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Figure 2.1.2.1-1a **Mars Spaceship, Case Study 2**

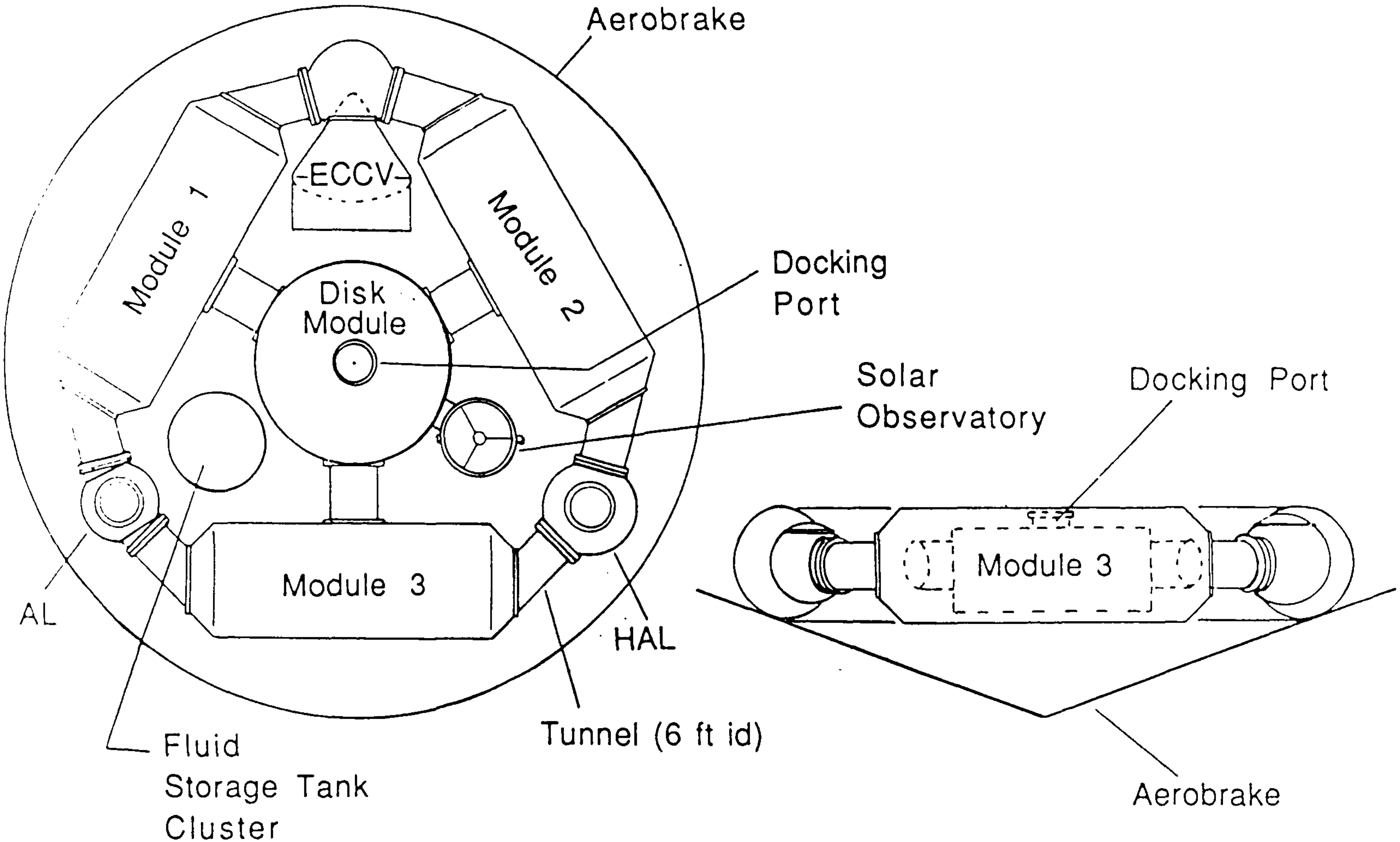

Figure 2.1.2.1.1b Artist's Depiction of the Mars Spaceship

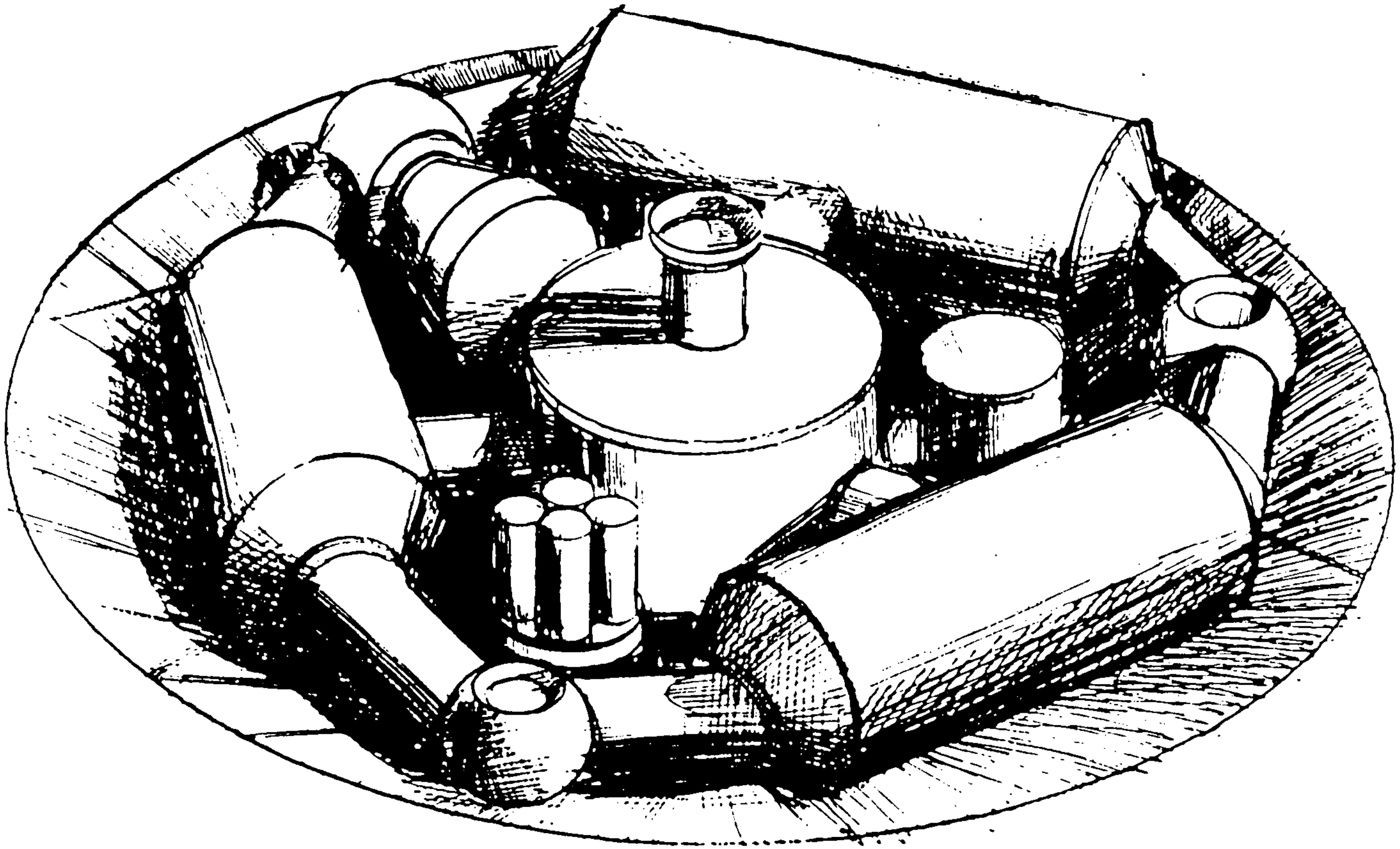

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Table 2.1.2.1-1. Mars Aerobrake Design

Assumptions: C_3 = 50 km^2/s^2; P/L mass = 130 t

Aerocapture brake characteristics:

Diameter	27.4 m [90 ft]	(Area = 591 m^2 [6361 ft^2])
M/C_dA	143 kg/m^2 [29.3 lb_m/ft^2]	
Angle of attack (alpha)	11.18°	
L/D	0.18	
Peak decelleration	8.08 gee	

Mass Summary

RSI Core (33' dia, 0.73" thick)	349 kg
RSI Honeycomb substrate	1261
Interface ring	1068
Radial Beams	692
Struts	1731
FSI Annulus (90' dia, 1.27" thick)	4590
Sub-Total	9691
Contingency (20%)	1938
Total	11,629 kg (8.9%)

Figure 2.1.2.1-2 Mars Cargo Vehicle (MCV)

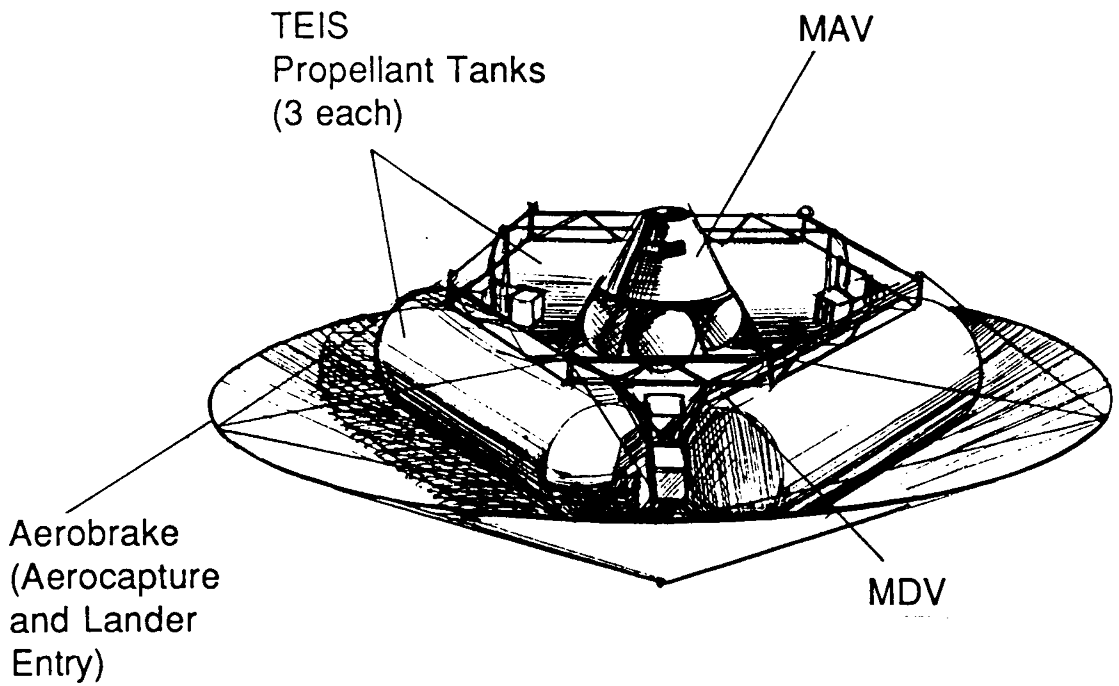


Table 2.1.2.1-2 Departure Energetics—Case Study 2

	TMI Dates		C_3 (km^2/s^2)			ΔV (km/s)		
			Earth Departure		Mars	Earth Departure		Mars
	Cargo	Piloted	Cargo*	Piloted	Departure	Cargo*	Piloted	Departure
TIC-1	6-03	10-04	8.8	33.0	21.0	3.56	4.59	2.04
TIC-2	7-05	12-06	15.4	70.9	29.5	3.85	6.06	2.65
TIC-3	9-07	2-09	12.7	61.1	46.9	3.73	5.69	3.76
Earlier	4-01	8-02	7.9	27.2***	25.0**	3.52	4.35	3.71**

* With A/C at Mars
** From Phobos orbit
*** Includes Venus swingby

Figure 2.1.2.1-3a Trans-Mars Injection System (TMIS), Cargo Vehicle

(sprint class mission, cargo, 2003)

Dry Mass	
(includes payload)	231,010 kg
Payload Mass	
(MTV)	189,070 kg
Propulsion System	
Propellant Type	LOX/LH_2
Engines	
Number	1
Type	Adv. Space Engine
Mass (ea.)	3,175 kg
Thrust (total)(543 klbf)	2,415 kN
I_{sp} (485 sec)	4.76 kN-s/kg
Propellant Mass	272,290 kg
Initial T/W	0.49
Mass Fraction	
Total	0.46
Total Mass	503,300 kg

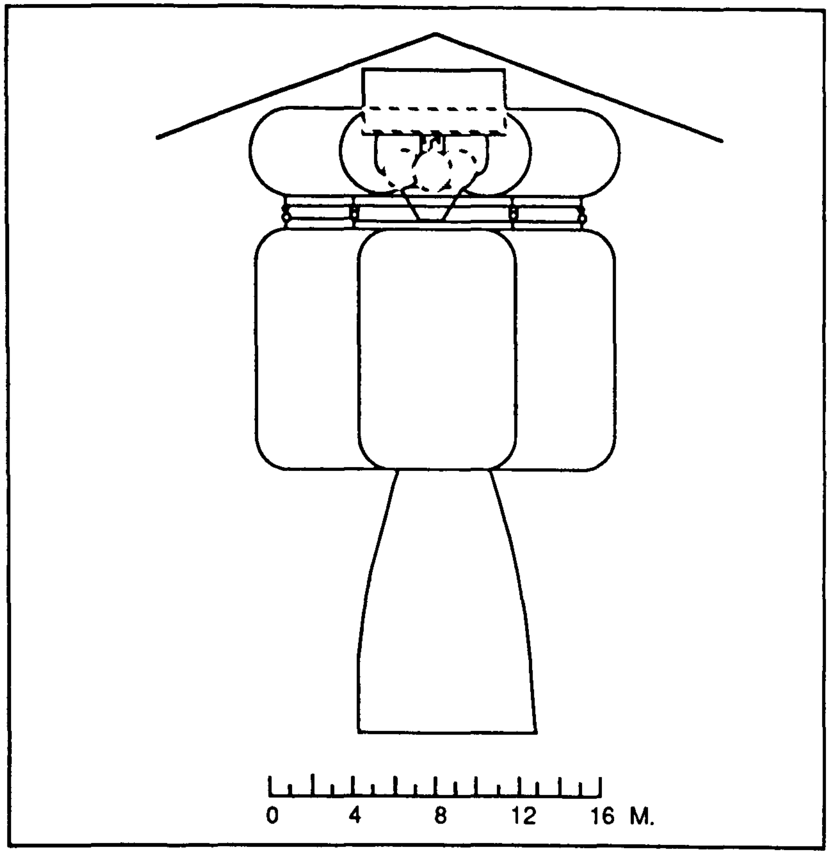


Figure 2.1.2.1-3b Expanded View of Cargo Vehicle with TMIS

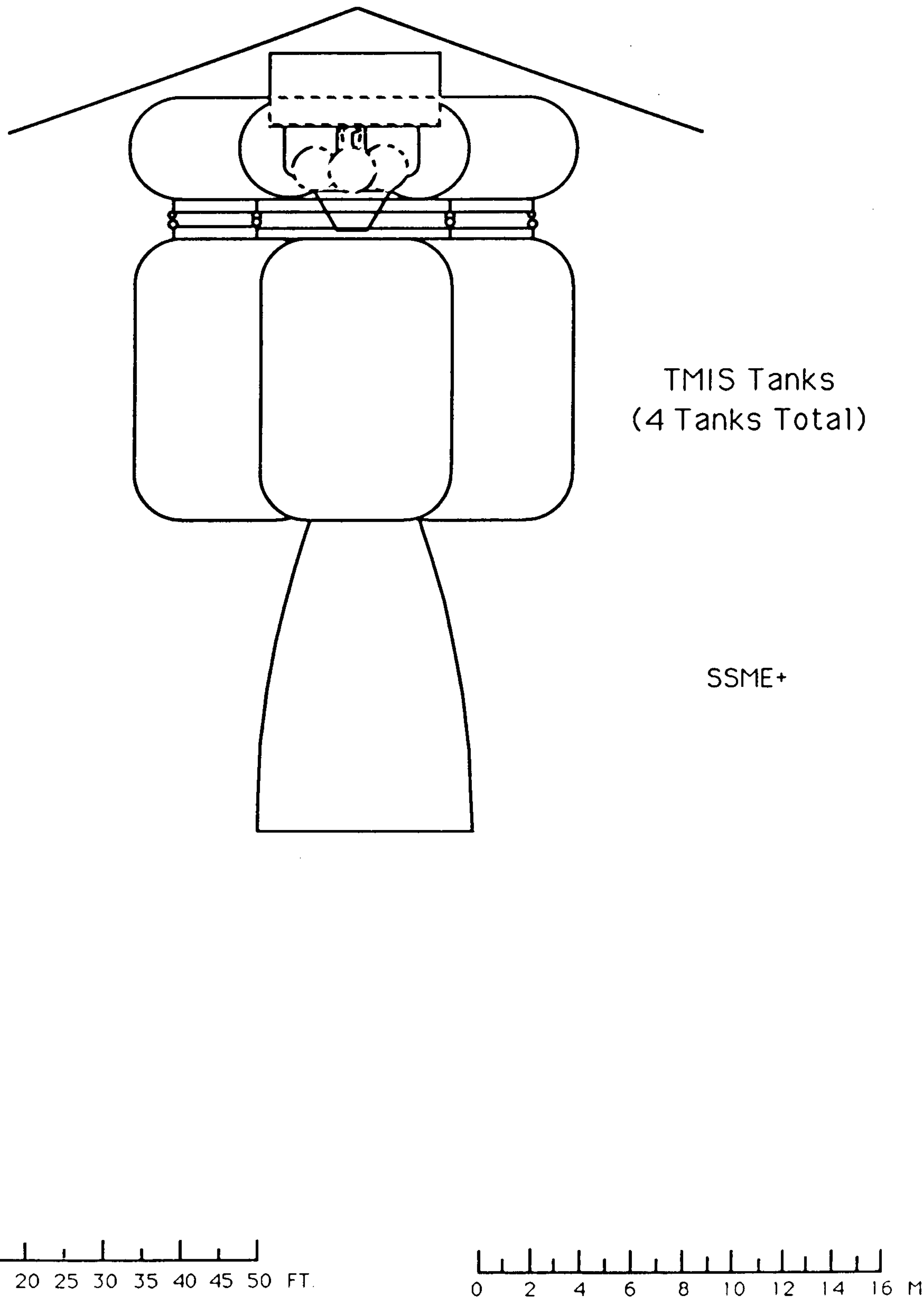


Fig 2.1.2.1-4a Trans-Mars Injection System (TMIS), Crew Vehicle

(sprint class mission, piloted, 2004)

Dry Mass	
(includes payload)	439,140 kg
Payload Mass	
(MTV)	334,130 kg
Propulsion System	
Propellant Type	LOX/LH_2
Engines	
Number	1
Type	Adv. Space Engine
Mass (ea.)	3,175 kg
Thrust (total)(543 klbf)	2,415 kN
I_{sp} (485 sec)	4.76 kN-s/kg
Propellant Mass	685,330 kg
Tank Mass	21,100 kg
Initial T/W	0.22
Mass Fraction	
Total	0.39
Total Mass	1,124,470 kg

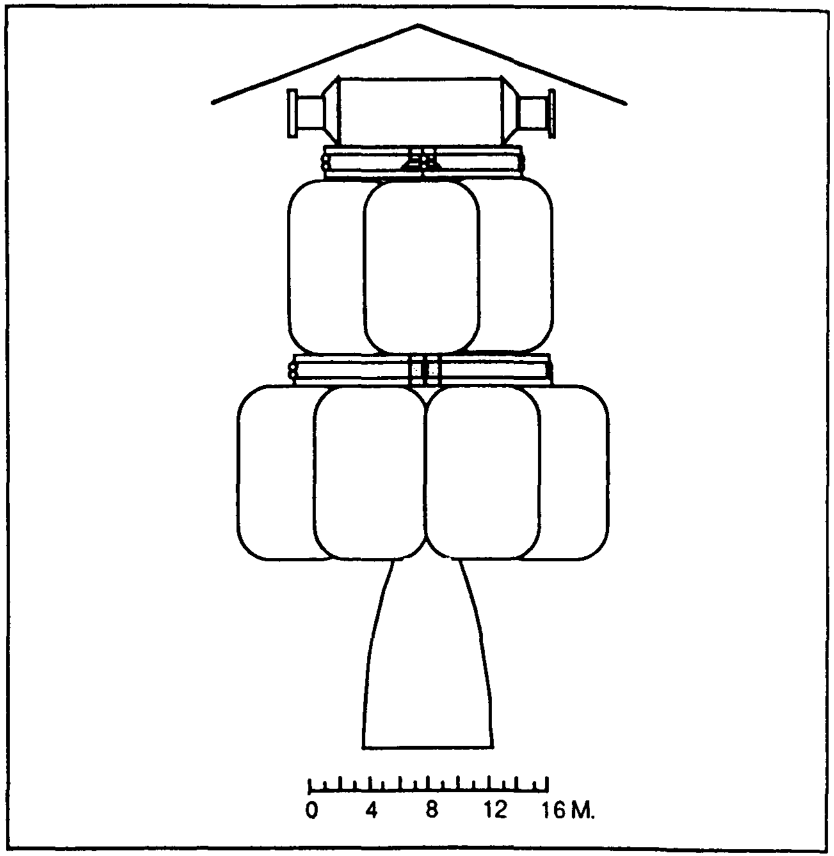


Figure 2.1.2.1-4b
Expanded View of Crew Vehicle with TMIS

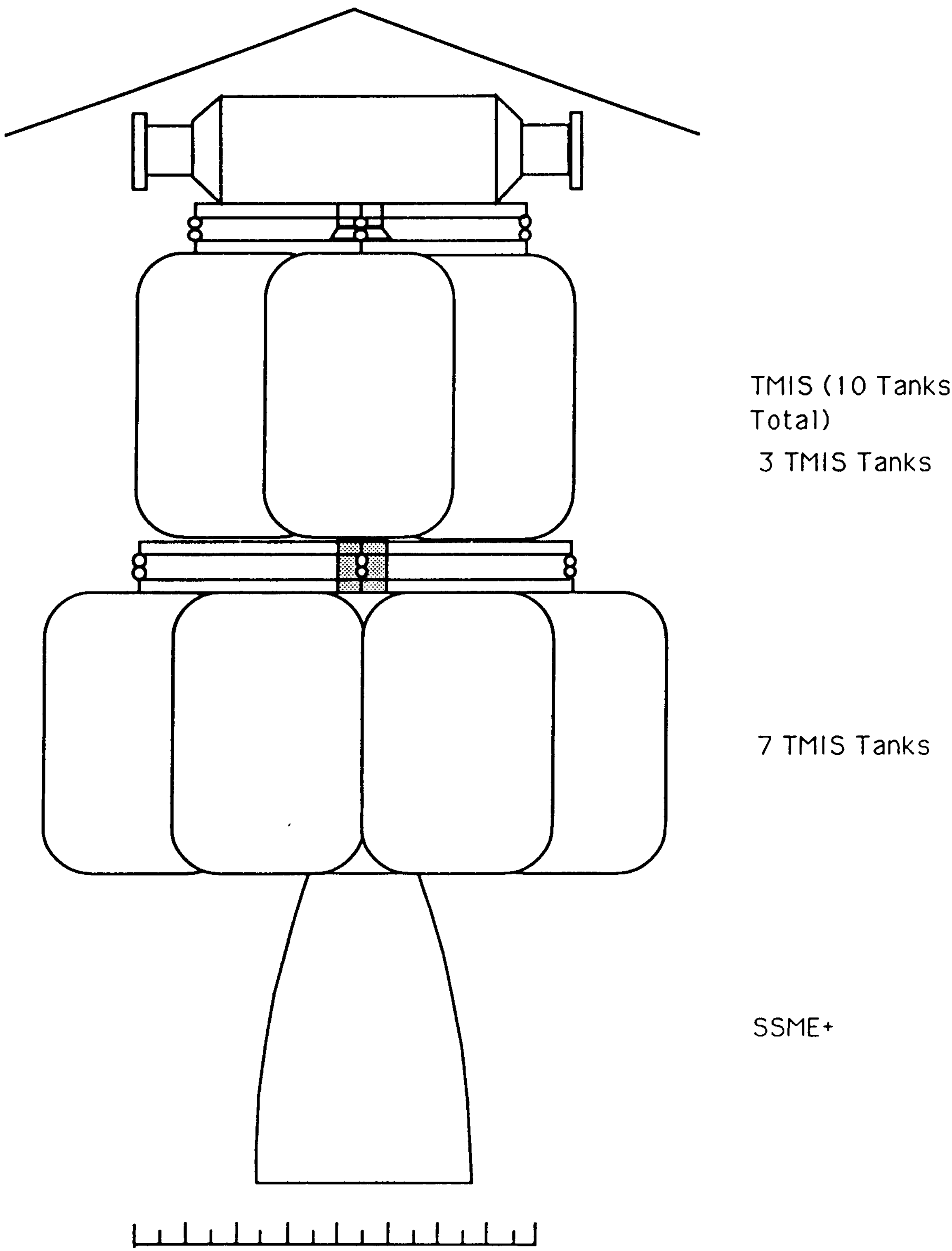


Figure 2.1.2.1-5a Mars Transfer Vehicle (MTV) (Piloted)

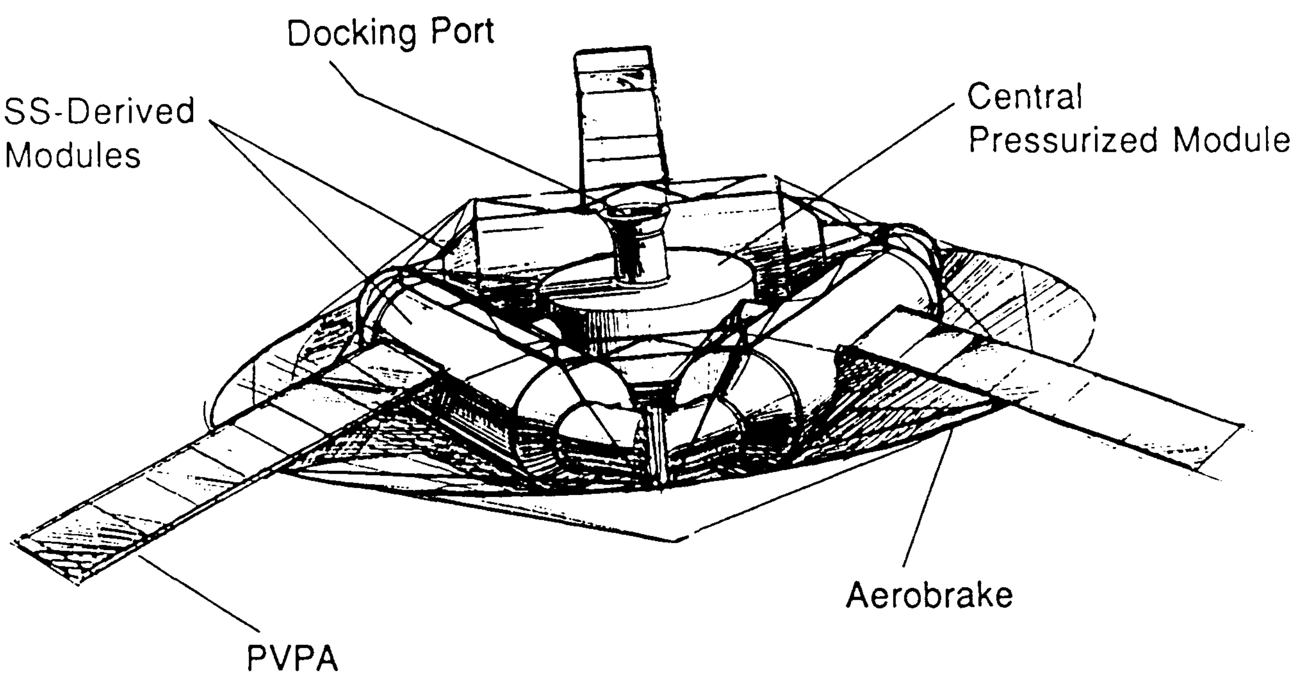


Figure 2.1.2.1-5b Mars Transfer Vehicle (MTV) with Photovoltaic Panels Deployed

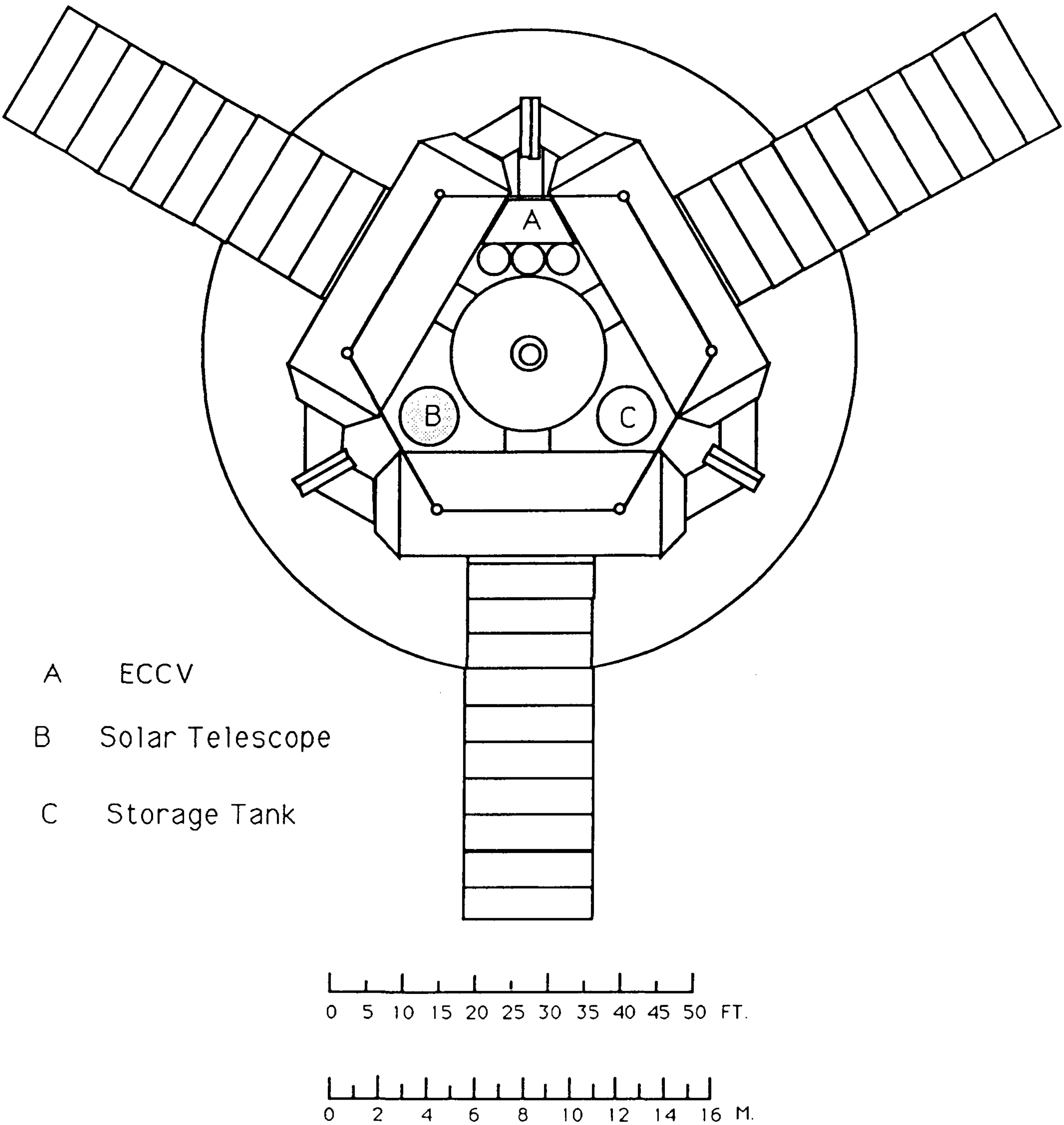


Table 2.1.2.1-3 Encounter Energetics—Case Study 2

	Launch Dates		C_3 (km^2/s^2) Mars Arrival		Earth Arrival
	Cargo	Piloted	Cargo*	Piloted	
TIC-1	6-03	10-04	7.3	43.2	25.0
TIC-2	9-05	12-06	12.3	46.5	12.1
TIC-3	9-07	2-09	8.0	48.6	18.7
Earlier	4-01	8-02	23.3	49.8	15.5

* With A/C at Mars

Figure 2.1.2.1-6
Mars Orbital Operations, Step 1 - Rendezvous

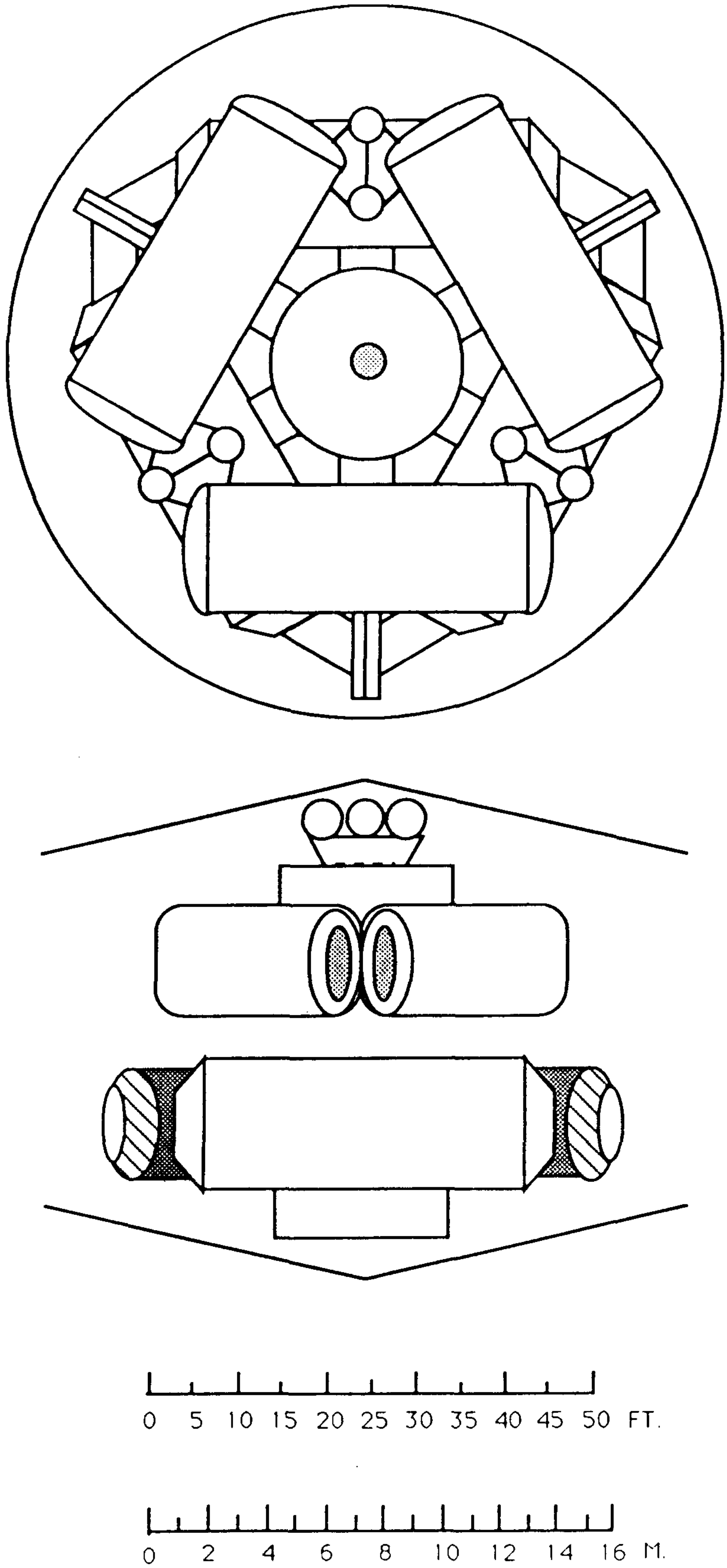


Fig. 2.1.2.1-7
Mars Orbital Operations, Step 2 - Docking of TEIS and MDV

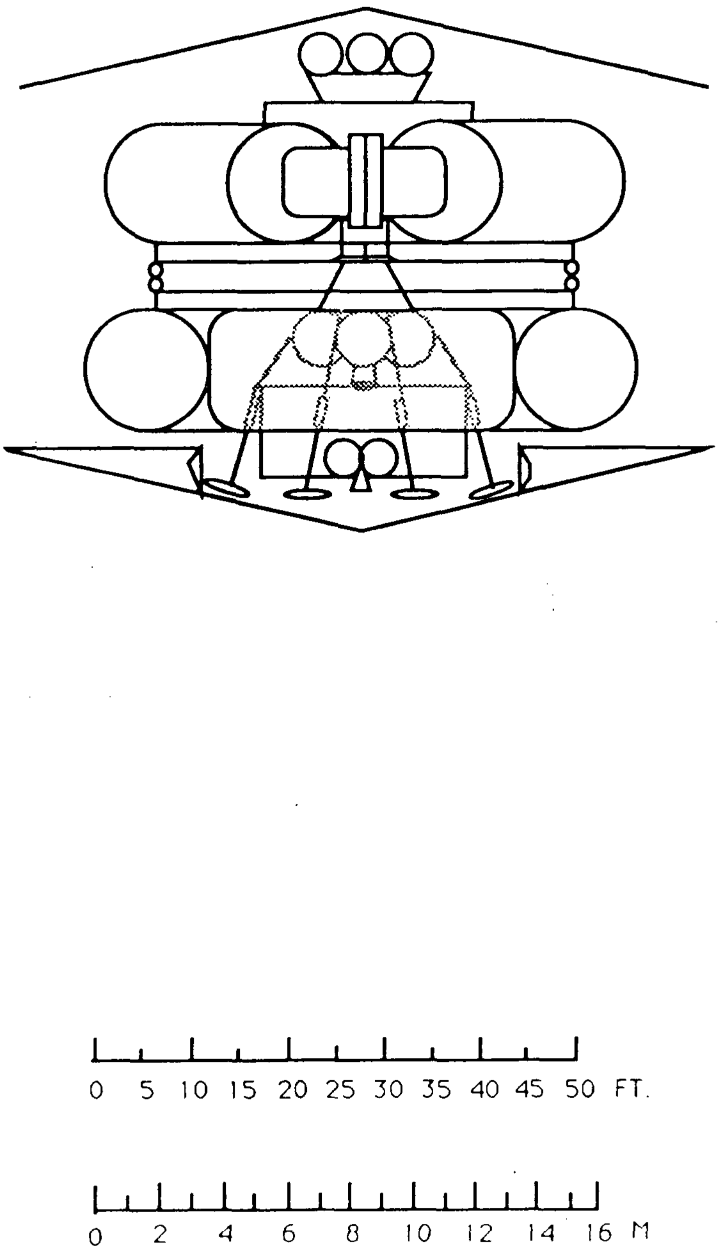
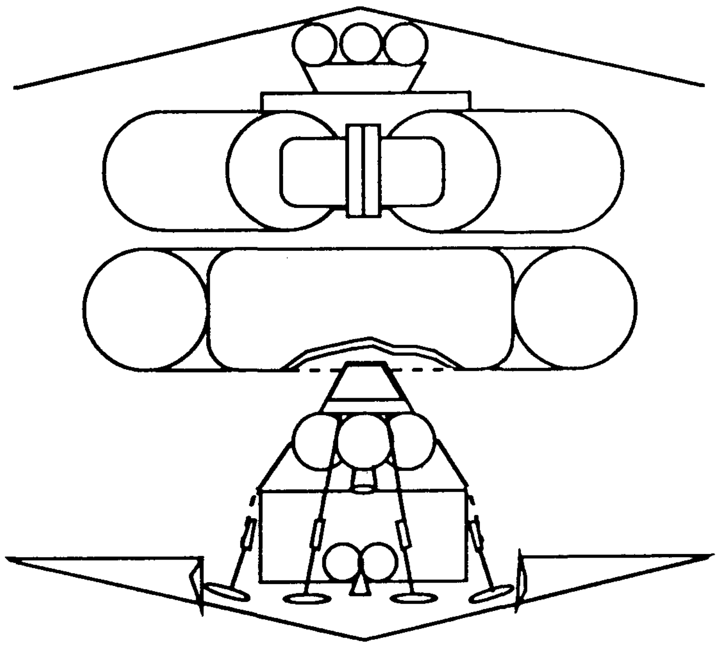


Figure 2.1.2.1-8
Mars Orbital Operations, Step 3 -
Release of MDV (after crew transfer) and Transfer of TEIS



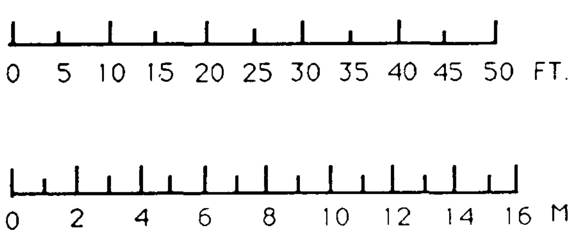


Figure 2.1.2.1-8a Mars Descent Vehicle (MDV)

(30 sol staytime, 4 crew)

Dry Mass (includes payload)	48,114 kg
Payload Mass (crew & user payload)	13,730 kg
Payload Volume	
MAV (cone - 1.8m rad., 2.3m ht.)	8 m^3
HAB Module (cyl.- 3.8m rad., 3m ht.)	136 m^3
Propulsion System	
Propellant Type	MMH/N_2O_4
Engines	
Number	3
Type	Delta
Mass (ea.)	100 kg
Thrust (total)	134 kN (30 klbf)
I_{sp} (320 sec)	3.14 kN-s/kg
Propellant Mass	3,406 kg
Tank Mass	341 kg
Initial T/W	0.71
Mass Fraction	
Terminal descent w/ parachute	0.93
Total Mass	51,520 kg

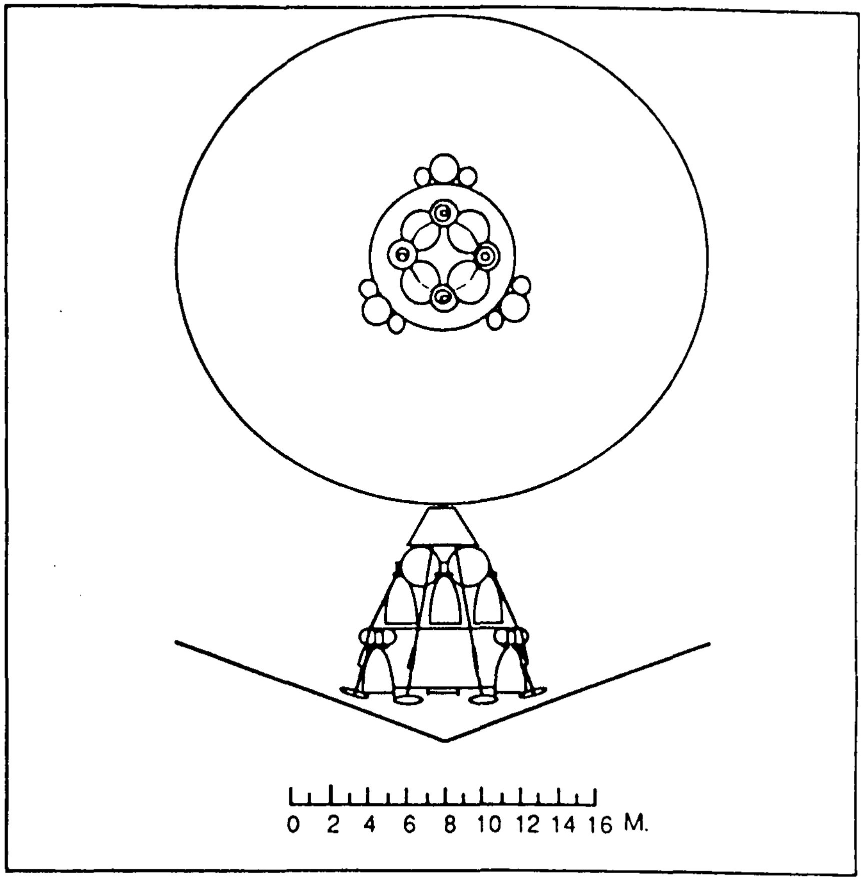


Figure 2.1.2.1-8b MDV

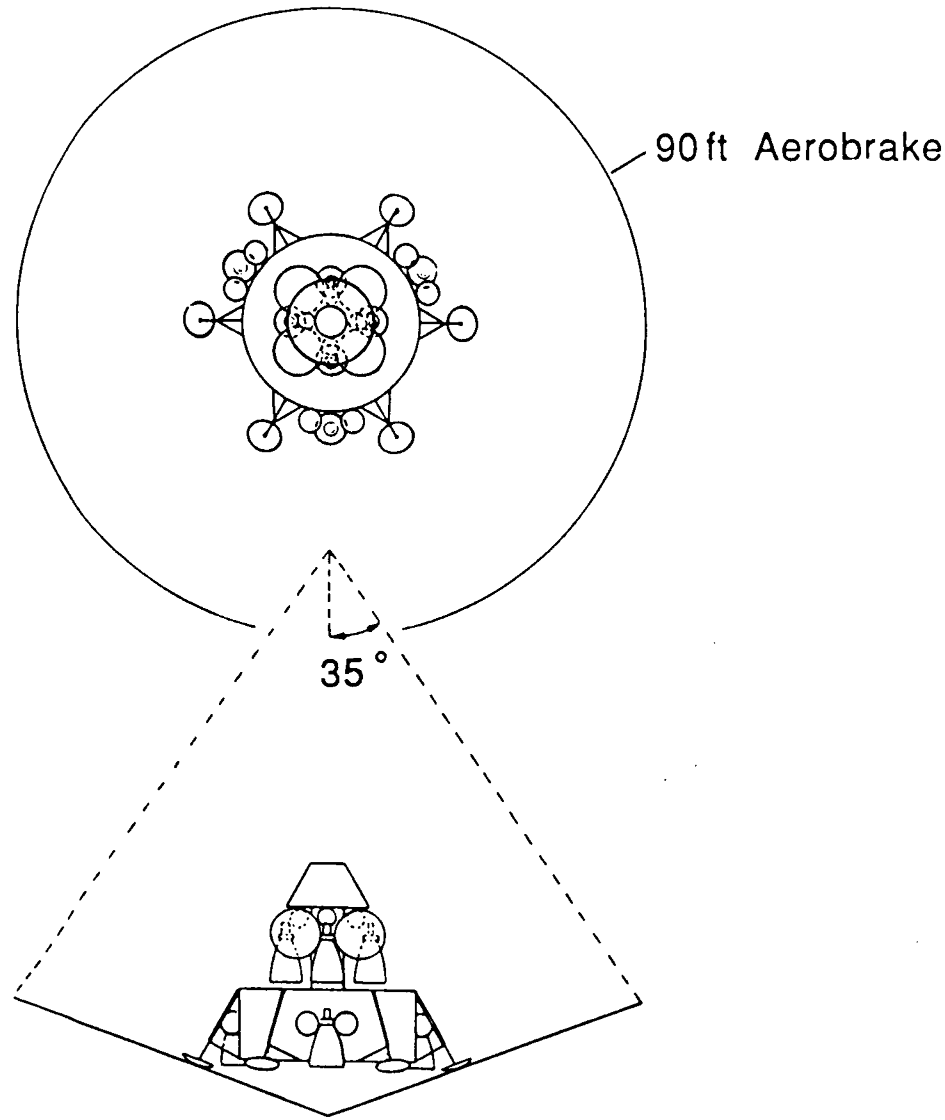
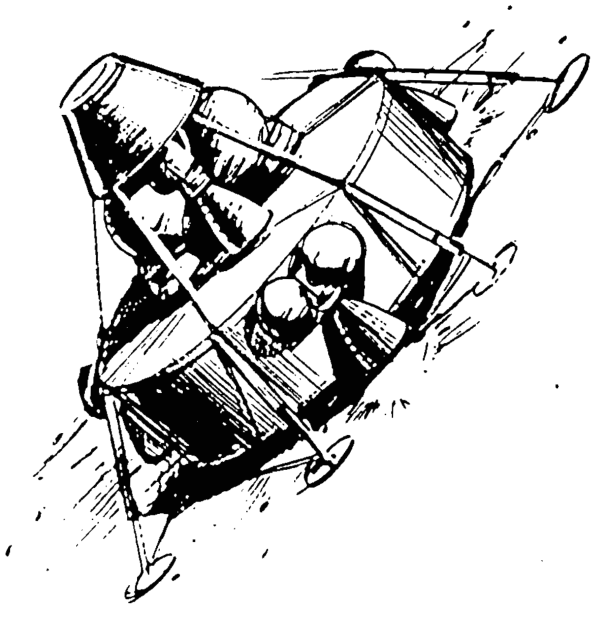


Figure 2.1.2.1-8c Artist's Depiction of the Mars Descent Vehicle



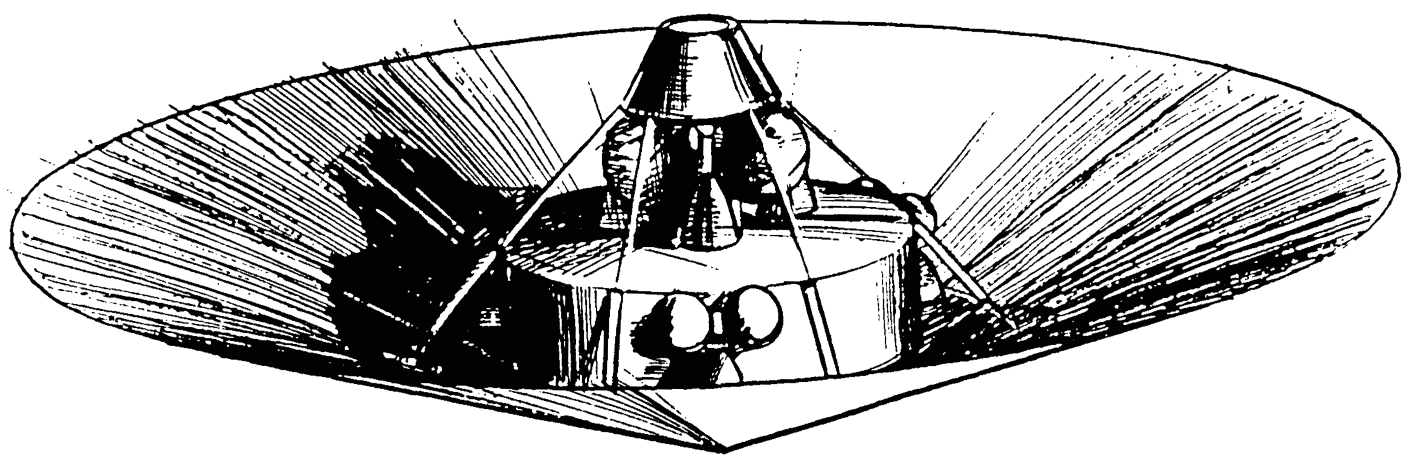


Figure 2.1.2.1-9a Mars Landed Mission Module (MLMM), Designed by Ethan Cliffton

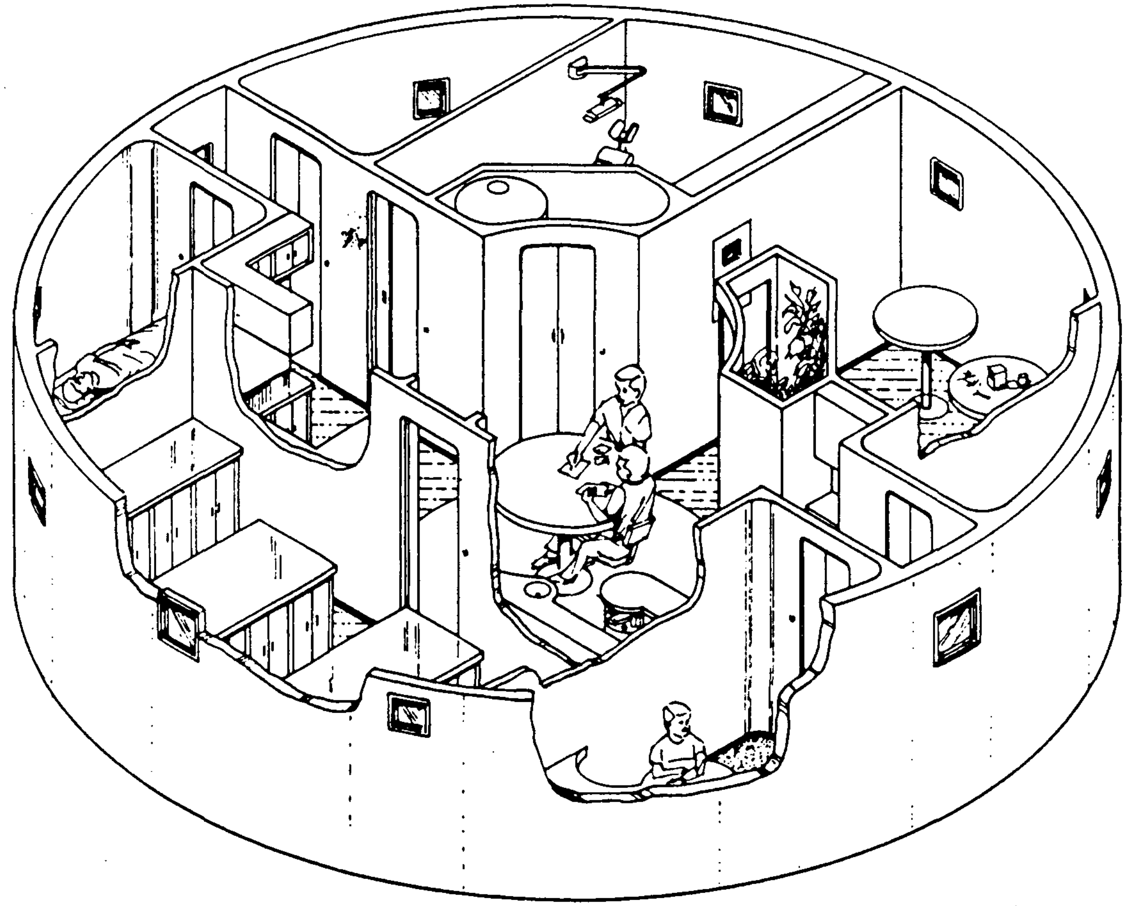


Figure 2.1.2.1-9b **Mars Landed Mission Module (MLMM), Designed by Ethan Cliffton**

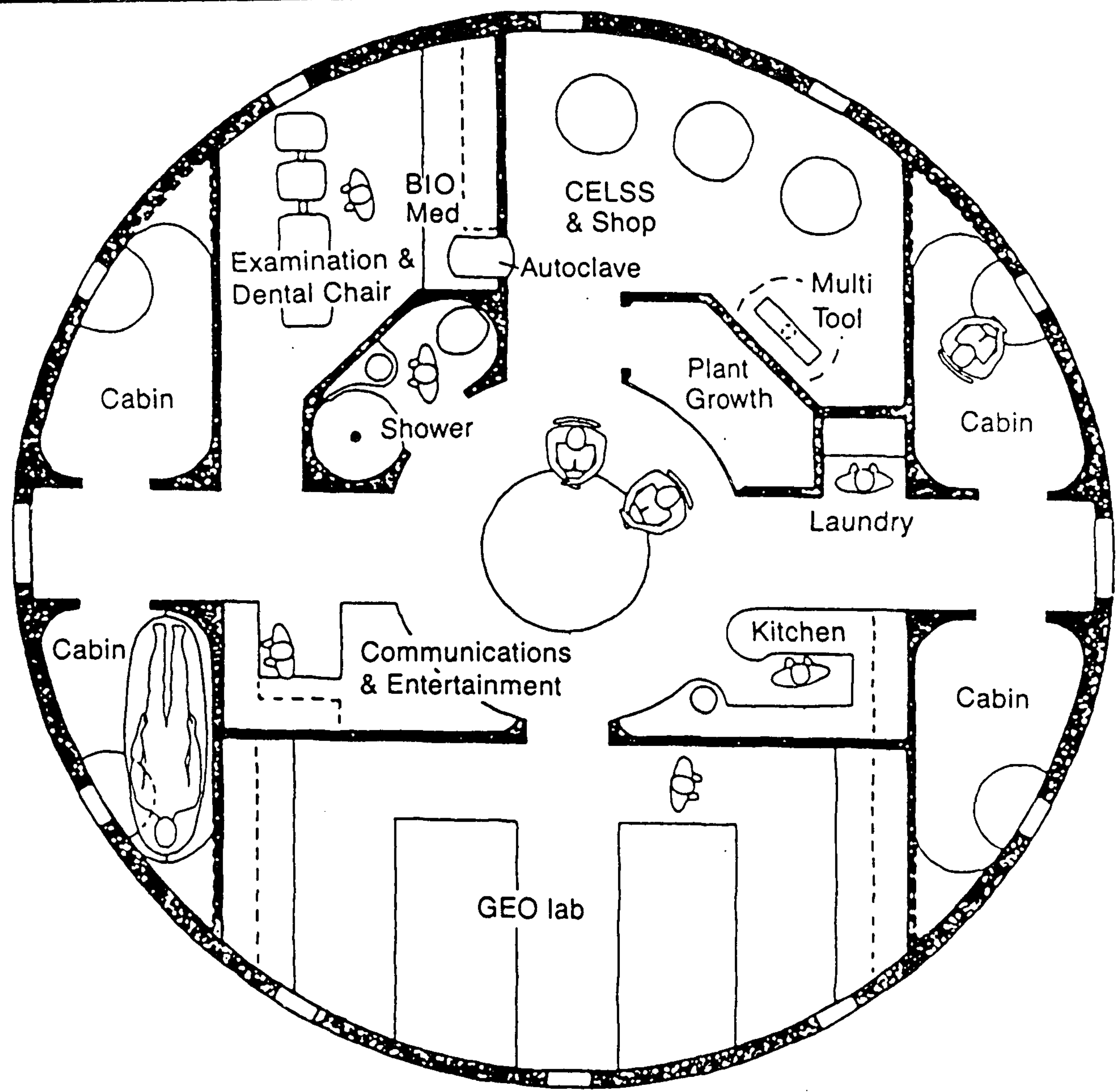


Figure 2.1.2.1-10 Mars Aerocapture Parametric Data

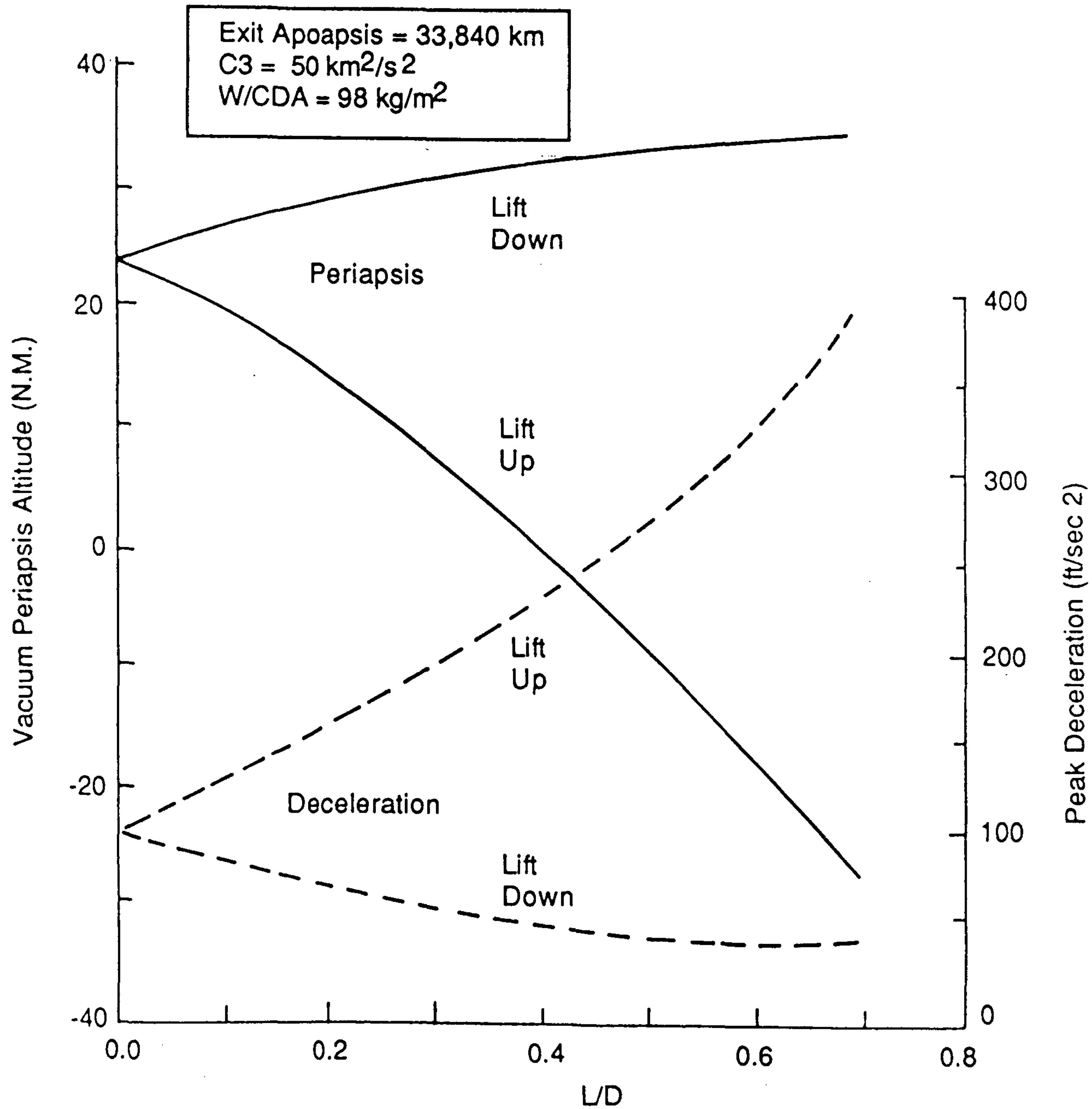


Table 2.1.2.1-4 ECLSS and Food for 20-Day, 4-Person Lander*

System	Mass (kg)	Power (W)
Hydrogen peroxide solution (85% H_2O_2) (supplies 100 kg O_2 and 150 kg H_2O)	250	
Tank and converters for hydrogen peroxide	30	20
Water recycling (VCD system, 2 ea.)	102	110
SFE (Static Feed Electrolyzer)**	46	(833)**
LiOH bed for CO_2 removal (incl. 10 kg hardware)	150	20
Trace Contaminant Control bed(TCCS)	49	113
Food (@ 1.5 kg/p-d)	150	
Food freezer/refrigerator	30	100
Make-up N_2 (incl. 20 kg tank)	45	
"Other" (@ 0.68 kg/p-d)***	68	
FDS (Fire Detection/Suppression System)	113	47
THC (Temperature/humidity Control)	153	600
ACS (Atmosphere Control System)	14	24
WM (Waste Management facilities)	109	30
Total	1259	1064

* with 25% contingency (note: habitat also includes approximately 36 kg O_2 in starting atmosphere)
** back-up for O_2 generation; also used as demonstrator for oxygen generation
*** includes TCSS, humidity removal, urine & fecal processing, trash collection, liquid supply, VCD

Fig. 2.1.2.1-11a Mars Ascent Vehicle (MAV)

Dry Mass	
(includes payload)	3,758 kg
Payload Mass	
(crew & supplies)	470 kg
Payload Volume	
MAV	
(cone - 1.8m rad., 2.3m ht.)	8 m^3
Propulsion System	
Propellant Type	MMH/N_2O_4
Engines	
Number	4
Type	Delta
Mass (ea.)	100 kg
Thrust (total)	178 kN (40 klb_f)
I_{sp} (320 sec)	3.14 kN-s/kg
Propellant Mass	19,422 kg
Tank Mass	1,942 kg
Initial T/W	2.55
Mass Fraction	
Ascent to 24.66 hr elliptical orbit	0.183
Total Mass	23,180 kg

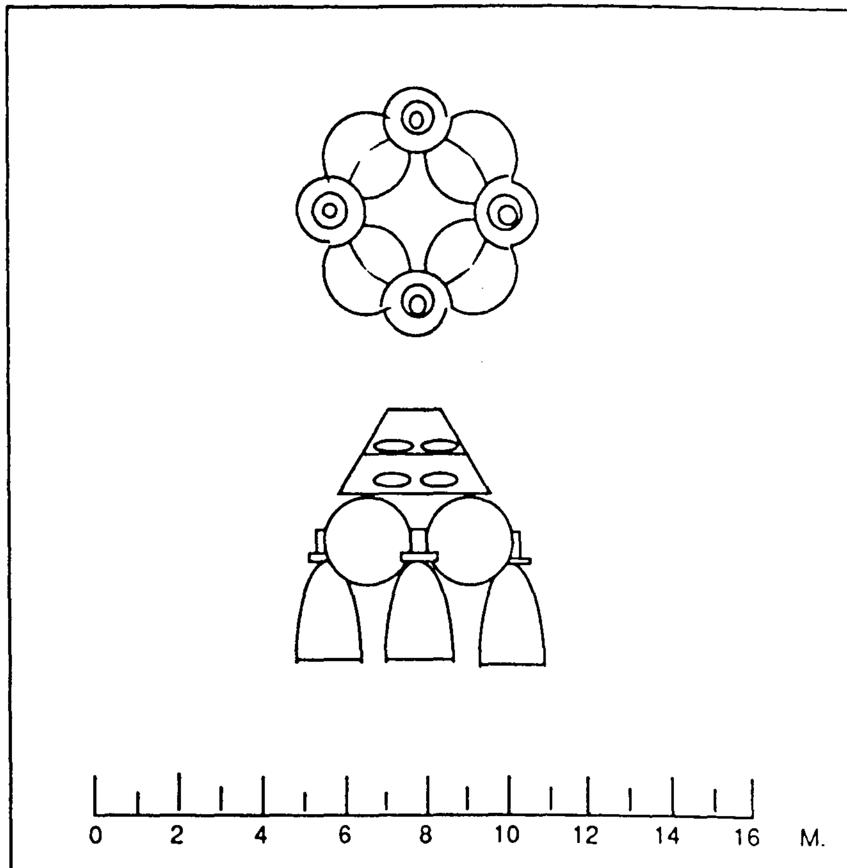


Figure 2.1.2.1-11b Artist's Depiction of the MAV

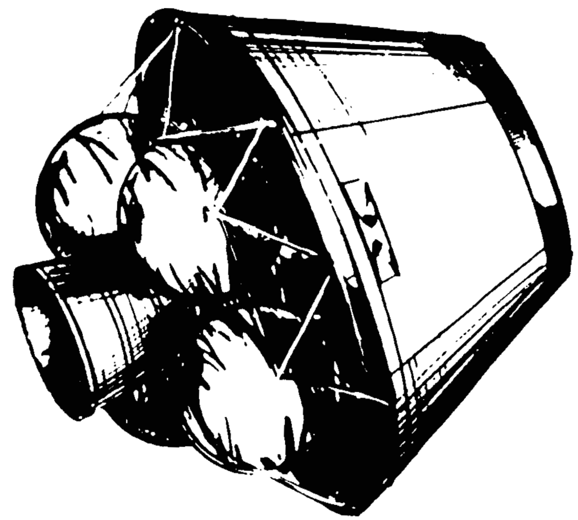


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Figure 2.1.2.1-12
Mars Orbital Operations, Step 4 - Docking of MAV

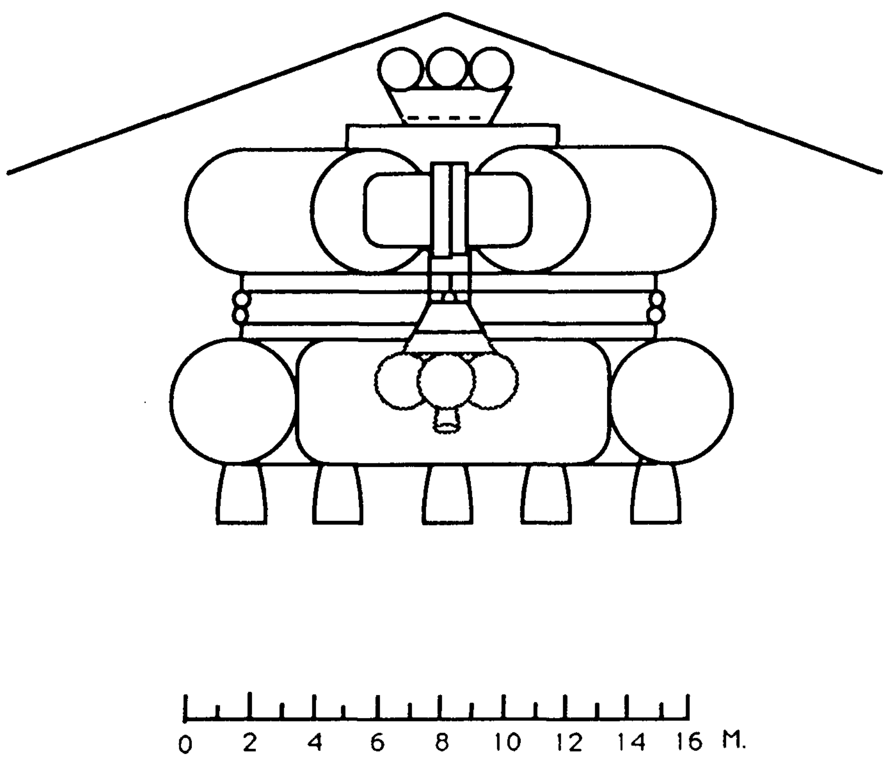


Figure 2.1.2.1-13a
Mars Orbital Operations, Step 5 - Jettison of MAV and Firing of TEIS

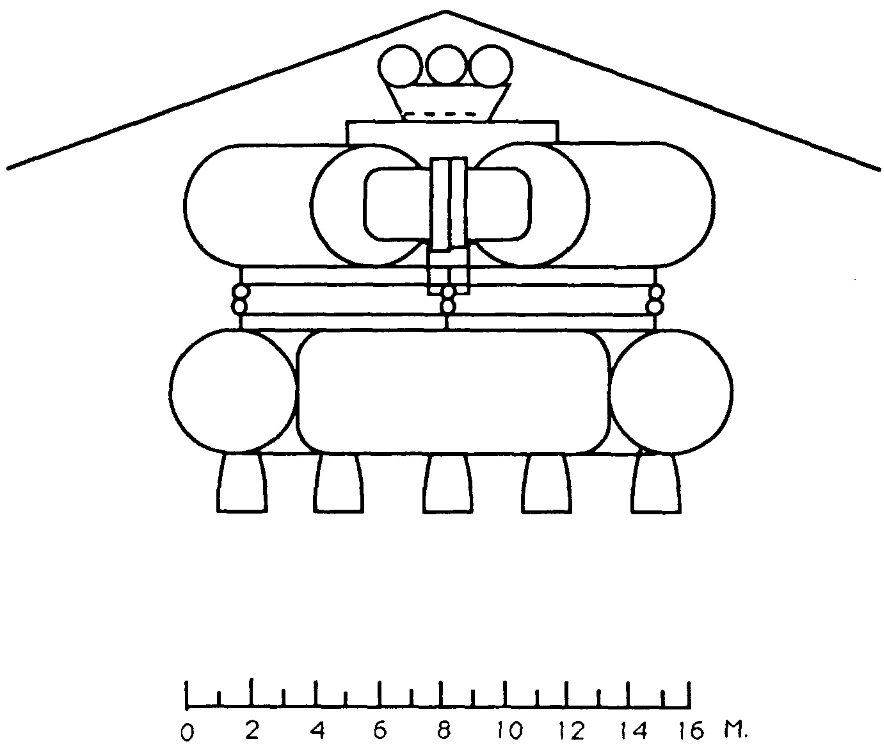


Figure 2.1.1-2 Reference Mission Options

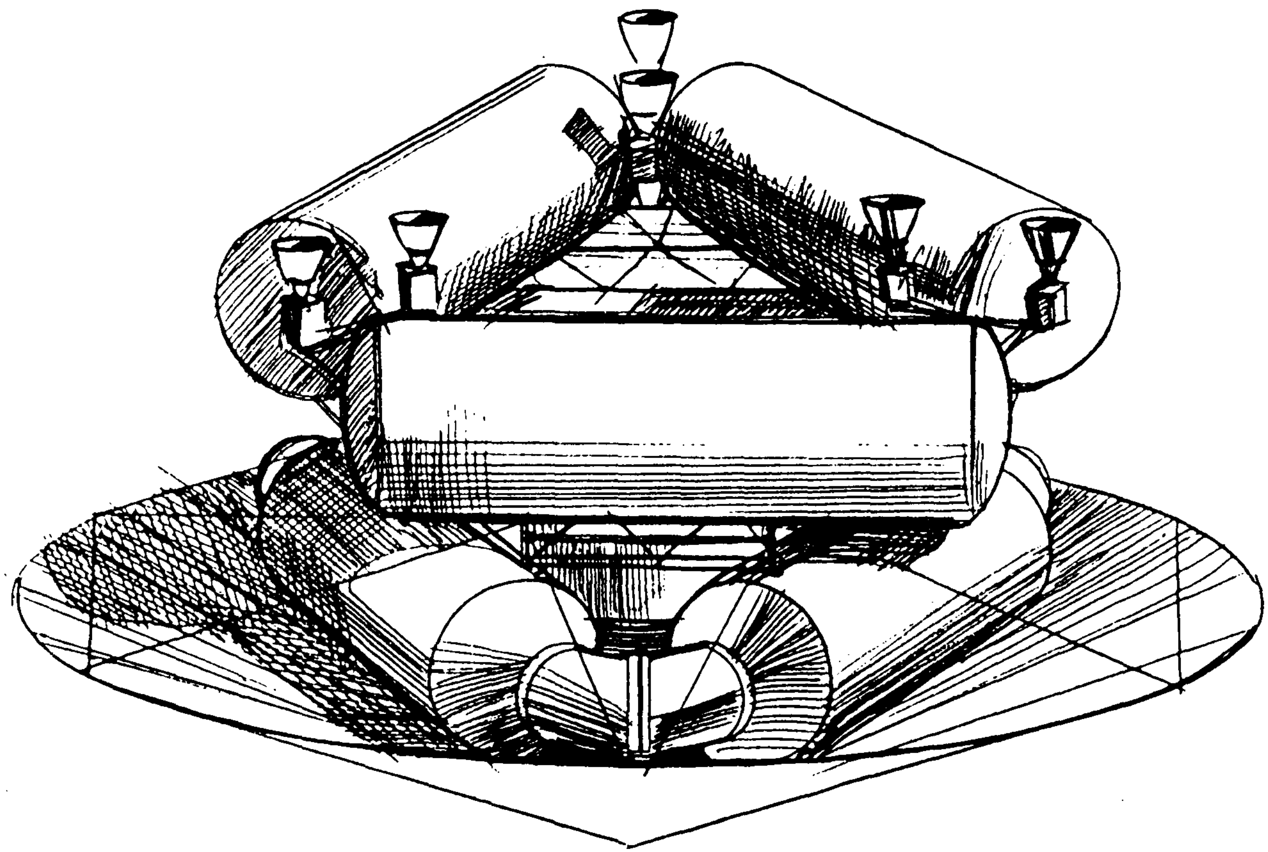
Scenario: CS-2 TIC-1R

OPTIONS — Options Selected: (Date & Your Name)

proven, or under development -- unproven, or must be developed / analyzed

Earth departure location	LEO	HEO	GEO	L1,L2				
On-orbit assembly	SS, attached	SS, free-flyer	Min. SS	no SS req'd.				
Hardware staging	integrated	split, MDV	split, MRSR	split, TEIS				
Trajectory type	flyby	conjunction	opposition	sprint	low thrust	cycler		
Launch dates	1990's	2000's	2010's	2020's	2030's			
Crew size, total	3	4	5	6	7	8	9-11	12-24
Cabin pressure	4.3 psi	14.7 psi	10.2 psi					
Gravity environment	microgravity	artificial, .38 g	artificial, 0.6 g	artificial, 1 g	hybrid			
Rotation rate	0 rpm	1 rpm	2 rpm	4 rpm	6 rpm			
Radiation protection	none	one storm shelter	two storm shelters	GCR shield				
Hab/Lab modules	SS modules (15' dia.)	ET derived (25' dia.)	large dia. (31' dia.)	inflatables				
Science equipment	interplanetary	Mars orbit	Mars surface	Phobos	Deimos			
ECLSS, spaceborne	consumables	SS ECLSS	water recycling	low mass/power	CELSS			
TMI launch propellant	hybrid chemical	LH2/LOX	Hc/LOX	waste/LOX	SEP	NTR	NEP	solar sail
engines, cryo.	RL-10 growth	F1 derived	SSME derived	advanced cryo				
growth	max.-sized tank	stretch	cluster					
reusability	non-recoverable	engines, avionics	all recoverable					
recovery method	none	turn-around	re-encounter					
Cryoprop storage	passive	active, refrigeration	active, reliquifaction	from H2O				
Power, spaceborne	PVPA	fuel cells	RTGs	nucl. reactor	DIPS	solar th.-dy.		
TM abort capabilities	Mars swingby	propulsive abort						
Mars orbit	LMO	8.2 hr. ellip.	24.6 hr. ellip.	Phobos	Deimos	GMO		
Mars orbit capture	propulsive braking	prop/aero hybrid	aerobraking					
Satellites Relay Com.	none	LMO	24.6 hr	12.3 hr	Molniya	GMO		
Mars Science Orbiters	none	polar	circ.	elliptical				
Unmanned Landers	none	penetrator(s)	rover(s)	sample return				
Ph/De teleoperators	none	Phobos	Deimos					

Figure 2.1.2.1-13b Mars Orbiting Vehicle (MOV)



(After TEIS Transfer)

Figure 2.1.2.1-14 Trans-Earth Injection System (TEIS)

(sprint class mission)

Dry Mass	
(includes payload)	96,350 kg
Payload Mass	
(ETV)	86,340 kg
Propulsion System	
Propellant Type	LOX/LH_2
Engines	
Number	6
Type	RL10-deriv.
Mass (ea.)	191 kg
Thrust (total)(132 klbf)	588 kN
I_{sp} (470 sec)	4.61 kN-s/kg
Propellant Mass	59,480 kg
Initial T/W	1.03
Mass Fraction	
Earth return	0.62
Total Mass	155,830 kg

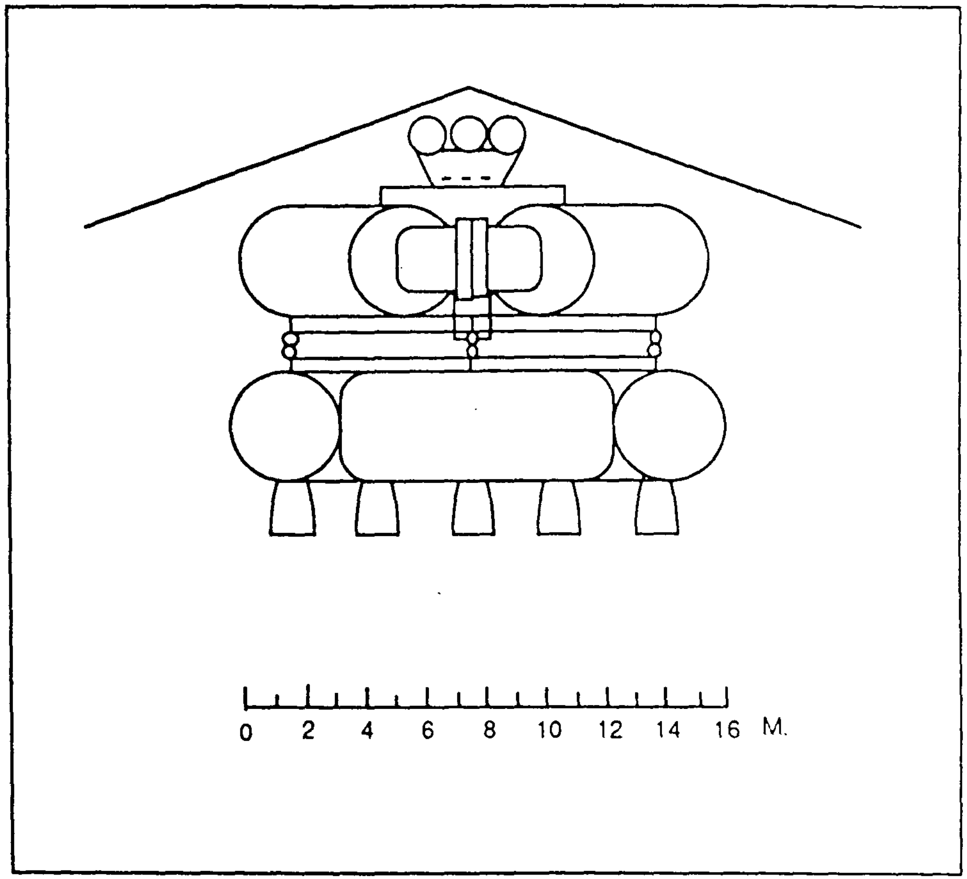


Figure 2.1.2.1-15a Earth Crew Capture Vehicle (ECCV)

(8 person crew)

Dry Mass	
(includes payload)	7840 kg
Payload Mass	
(Crew & Supplies)	1110 kg
Payload Volume	
(cone - 4.2m dia., 4m ht.)	18.5 m^3
Propulsion System	
ACS only	
Total Mass	7840 kg

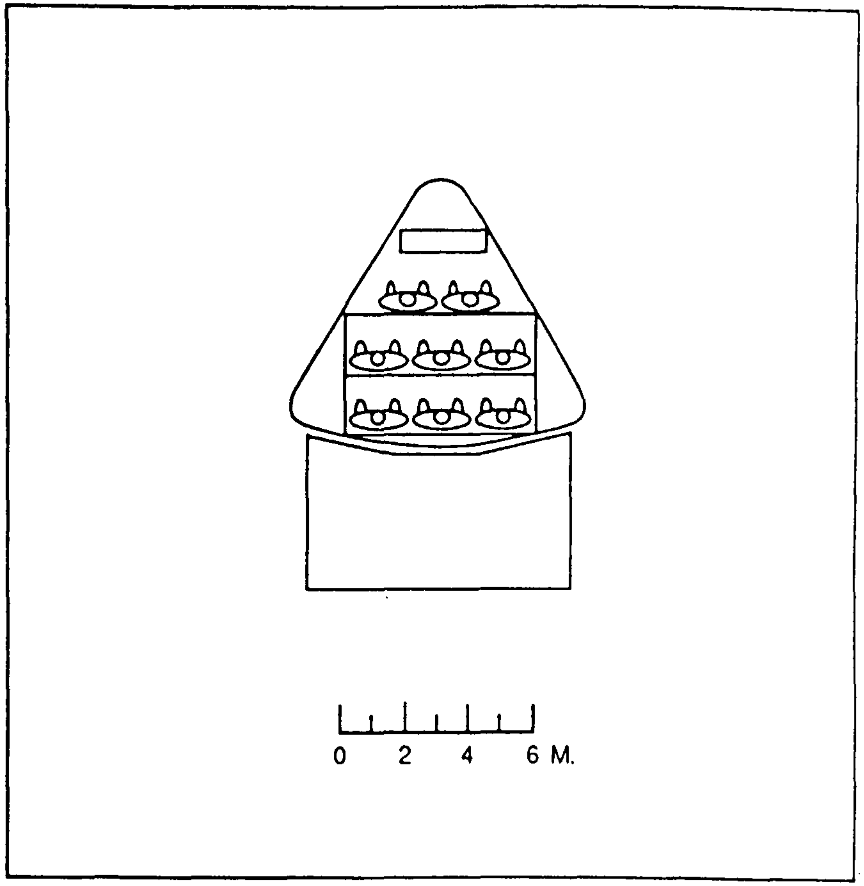


Figure 2.1.2.1-15b Artist's Depiction of the ECCV

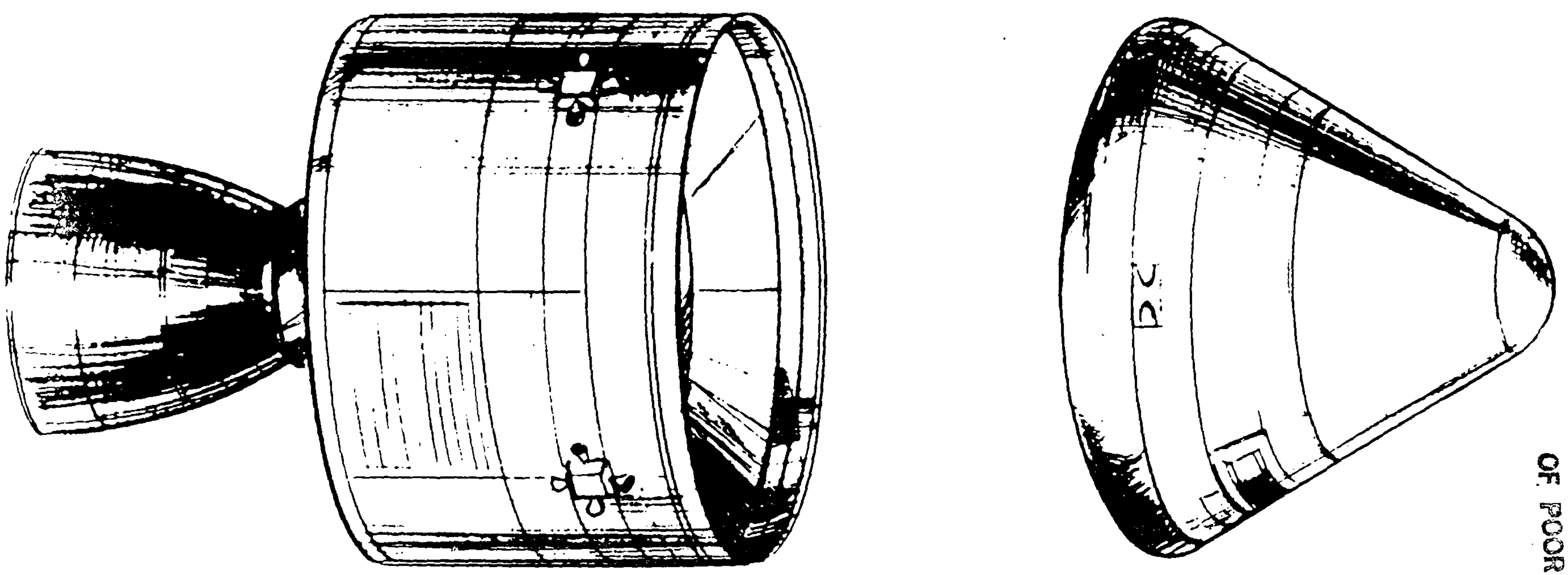


Table 2.1.2.2-1
CS-2 Design Features of Split Configuration

- **Extensive use of triangular structures**
  - Strongest natural structural basic design
  - Inverted triangle mating configuration
    - provides hexagonal docking points, with double-redundant tripod connects

- **Commonality in aerobrake designs** (27.4 meter [90'])
  - Both brakes are dual-purpose (four for the price of two)
    - cargo MOCS brake is re-used as piloted landing (MELS) brake
    - piloted MOCS brake is re-used as EOCS brake
  - Aerobrake size-driver is EOC
    - optimum MELS brake is 18 m [60']
    - optimum piloted MOCS brake is 27.4 m [90']
  - Core load-bearing brakes
    - 7.3 m [24'] diameter inner core
      (outer brake is one time locking self-deployment)

- **Commonality in configurations**
  - Triangular structures for both habitat modules and propellent tankage
  - Truss designs similar, but different load factors

- **On-orbit assembly**
  - Major use of pre-assembled, pretested elements
    - cylindrical modules *with* end-fittings fit in Shuttle-C 24-foot diameter envelope
    - disk modules compatible with Shuttle-C or ET Aft-cargo Carrier concepts
  - Early on-orbit habitation possible
    - one disk module coupled to one cylindrical module with PVPA is habitable

- **TEIS is stand-alone, strong-back, fully self-contained system**
  - Hard tie-points may be released and checked before piloted mission launch
    (requires only soft-points release for acquisition)
  - Electrical interconnects not required
    - radio link control from Command Module to TEIS
    - TEIS power derived, as required, from fuel cells
      use boil-off propellants in standby mode; direct consumption for firings

- **Fully redundant engine arrays**
  - inner triangle is *primary* engine set
    - either set is capable of fulfilling complete TEI propulsion needs
  - each engine in a pair is fed from a different tank

- **Command Module**
  - provides high-quality living space (max area:height ratio)
  - can serve as command post, radiation shelter, safe haven

- **Entry Points**
  - three independent entries to all modules
    - allows sealing of any faulty module(s) without preventing
      assess to all remaining functional modules

Table 2.1.2.2-1 (cont.) CS-2 Design Features of Split Configuration

- **External Services**
  - Power
    - triple, independent PVPA systems
      (300 m^2 total, 19.5 kWe at Mars)
      independent direct-connects to cylindrincal modules
    - retrievable stowage for TMI, MOC, TEI, EOC; jettison capability
  - Communication
    - multiple communication portals; steerable dishes connected to safe haven
  - Thermal control
    - triple systems, behind PVPAs

- **MAV docking and checkout**
  - No EVA required for occupation (shirt-sleeve passthrough tunnel)
  - Docking for intial entry serves as checkout of post-ascent docking
  - Permits rapid crew access to MDV system

- **Emergency rendezvous**
  - Cargo vehicle can make major maneuvers to reach MSS if necessary
    (jettison of aerobrake; use of MDV descent propulsion;
    jettison of all of MDV but MAV; use of MAV propulsion;
    jettison of MAV; use of some MCV RCS and MOO allocation)

Table 2.1.2.3-1 Earth-to-Orbit Sequence*

Year		'02	'03	'04	'05	'06	'07	'08	
Mission	1R	3	9	9					
	2R			8	12	14			
	3R					9	13	18	
Total Launches		3	9	17	12	23	13	18	(=95)

* 91 t HLLV; crew launches not included

Table 2.1.2.3-2
ETO Manifest -- Case Study 2

Mars Cargo Vehicle (MCV), Mission #1 Components Launches for Assembly in LEO

Launch	Date	Item	Mass (t)	L(m)	D(m)
1. HLLV-1	Oct '02	MDV	53.8	9.8	10.1
		TMIS Engine	3.4	9.1	8.2
2. HLLV-2	Nov '02	3 TEIS Flight Tanks	10.5	15.2	10.1
3. HLLV-3	Dec '02	Aerobrake	20.9	12.2	10.1
		MOO, MTX, Payload	50.3	12.2	10.1
4. HLLV-4	Jan '03	TEIS Fuel	72.5	12.2	10.1
5. STS-1	Jan '03	Crew for teleoperated assembly: - TMIS - integration of MTV with TMIS			
6. HLLV-5 to 8	Feb -Apr '03	TMIS Tanks #1-4	91 ea.	24.4	10.1

Teleoperated docking of TEIS and MCV from Earth . Launch June '03 via Earth command.

Total	575.4 t

Table 2.1.2.3-2 (cont.)
ETO Manifest -- Case Study 2

Mars Spaceship (MSS), Mission #1 Components Launches for Assembly in LEO

Launch	Date	Item	Mass (t)	L(m)	D(m)
1. HLLV-1	Oct '03	Aerobrake	28.6	12.2	10.1
		ECCV	9.7	6.1	4.6
		TMIS Engine	3.4	9.1	8.2
2. HLLV-2	Nov '03	3 SS Modules	53.6	15.2	10.1
		Disk module & supplies	14.5	3.1	7.7
3. HLLV-3	Dec '03	DSM, MOO, MOC	75.0	24.4	10.1
4. HLLV-4	Dec '03	DSM, MOO, MOC	75.0	24.4	10.1
5. HLLV-5	Jan '04	DSM,MOO, MOC	75.0	24.4	10.1
6. HLLV-6 to15	Feb - Aug '04	TMIS Tank #1-10	91 ea.	24.4	10.1

Flight crew integrates TMIS with MSS. Final checkout. Launch Oct '04 by flight crew.

Total	1244.8 t

Table 2.1.2.3-2 (cont.)
ETO Manifest -- Case Study 2

Mars Cargo Vehicle (MCV), **Mission #2** Components Launches for Assembly in LEO

Launch	Date	Item	Mass (t)	L(m)	D(m)
1. HLLV-1	Sept '04	MDV	53.7	9.8	10.1
		TMIS Engine	3.4	9.1	8.2
2. HLLV-2	Sept '04	TEIS Flight Tanks	15.6	15.2	10.1
3. HLLV-3	Oct '04	Aerobrake	27.2	12.2	10.1
		MOO, MTX, Payload	63.8	12.2	10.1
4. HLLV-4	Nov '04	TEIS Fuel #1, PhEV	66.4	18.3	10.1
5. HLLV-5	Nov '04	TEIS Fuel #2	58.7	12.2	10.1
6. STS -1	Dec '04	Crew for teleoperated assembly: - TMIS - integration of MTV with TMIS			
7. HLLV-6 to 12	Dec '04 -Apr '05	TMIS Tanks #1-7	91 ea.	24.4	10.1

Teleoperated docking of TEIS and MCV from Earth . Launch Sept '05 via Earth command.

Total 925.8 t

Table 2.1.2.3-2 (cont.)
ETO Manifest -- Case Study 2

Mars Spaceship (MSS), Mission #2 Components Launches for Assembly in LEO

Launch	Date	Item	Mass (t)	L(m)	D(m)
1. HLLV-1	May '05	Aerobrake	30.5	12.2	10.1
		ECCV	9.7	6.1	4.6
		TMIS Engine	3.4	9.1	8.2
2. HLLV-2	June '05	3 SS Modules	53.6	15.2	10.1
		Disk module & supplies	14.5	3.1	7.7
3. HLLV-3	July '05	DSM, MOC, MOO	76.7	24.4	10.1
4. HLLV-4	July '05	DSM, MOC, MOO	76.7	24.4	10.1
5. HLLV-5	Aug '05	DSM, MOC, MOO	76.7	24.4	10.1
6. HLLV-6 to24	Sept '05 -Aug '06	TMIS Tank #1-18	91 ea.	24.4	10.1

Flight crew integrates TMIS with MSS. Final checkout. Launch Dec '06 by flight crew.

Total	1979.8 t

Table 2.1.2.3-2 (cont.)
ETO Manifest -- Case Study 2

Mars Cargo Vehicle (MCV), **Mission #3** Components Launches for Assembly in LEO

Launch	Date	Item	Mass(t)	L(m)	D(m)
1. HLLV-1	Sept '06	MDV	53.9	9.8	10.1
		TMIS Engine	3.4	9.1	8.2
2. HLLV-2	Oct '06	3 TEIS Flight Tanks	10.5	15.2	10.1
3. HLLV-3	Nov '06	Aerobrake, MOCS	40.8	12.2	10.1
4. HLLV-4	Nov '06	MOO	89.4	24.4	10.1
5. HLLV-5	Dec '06	TEIS Fuel #1, DEV	77.5	18.2	10.1
6. HLLV-6	Jan '07	TEIS Fuel #2, PhEV	79.6	18.2	10.1
7. HLLV-7	Jan '07	TEIS Fuel #3	69.3	12.2	10.1
8. HLLV-8 to16	Feb '07 -July '07	TMIS Tanks #1-9	91 ea.	24.4	10.1

Teleoperated docking of TEIS and MCV from Earth . Launch Sept '07 via Earth command.

Total	1243.4 t

Table 2.1.2.3-2 (cont.)
ETO Manifest -- Case Study 2

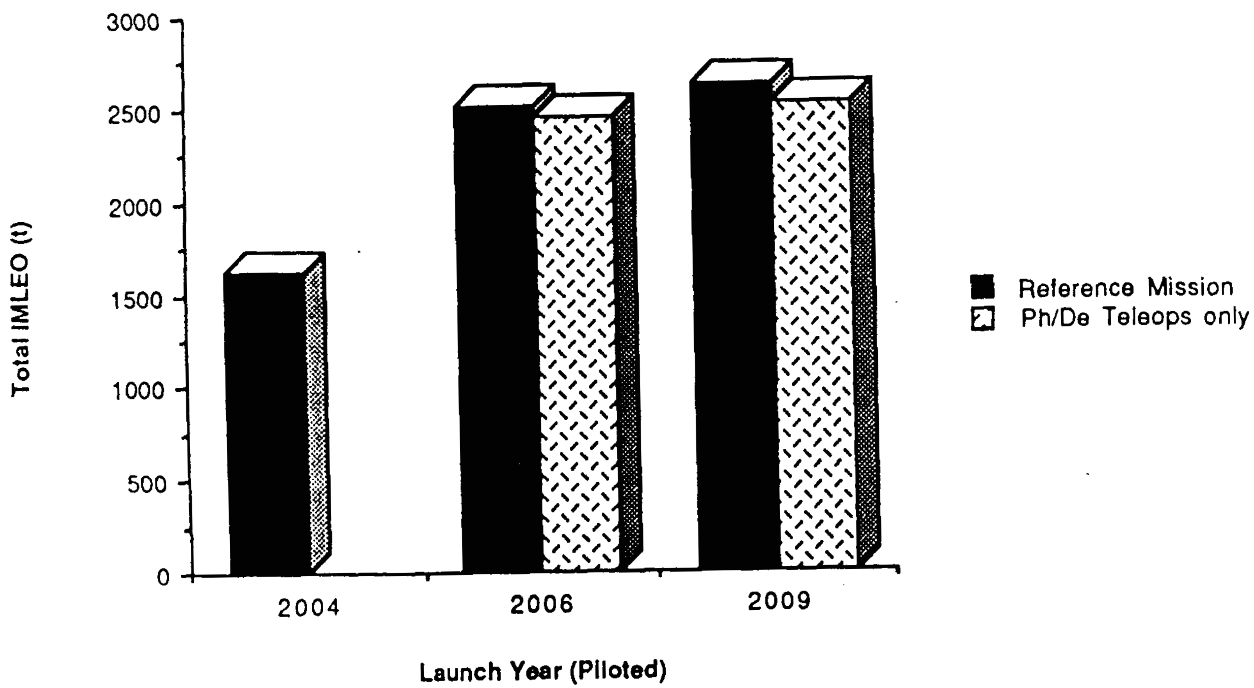
Mars Spaceship (MSS), Mission #3 Components Launches for Assembly in LEO

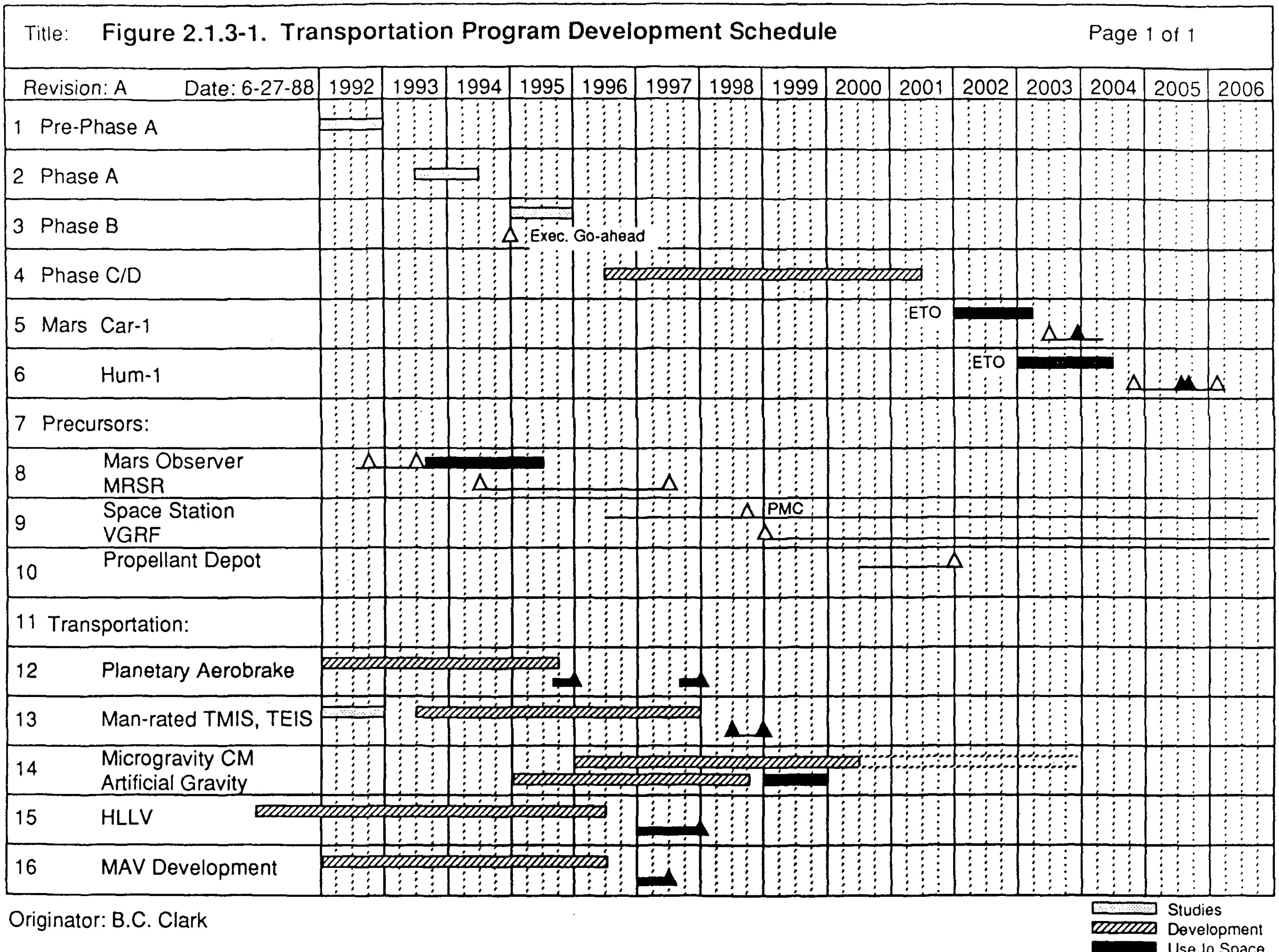
Launch	Date	Item	Mass (t)	L(m)	D(m)
1. HLLV-1	Nov '07	Aerobrake	34.0	12.2	10.1
		ECCV	9.7	6.1	4.6
		TMIS Engine	3.4	9.1	8.2
2. HLLV-2	Dec '07	3 SS Modules	53.6	15.2	10.1
		Disk module & supplies	14.5	3.1	7.7
3. HLLV-3	Jan '08	DSM, MOC, MOO	83.2	24.4	10.1
4. HLLV-4	Jan '08	DSM, MOC, MOO	83.2	24.4	10.1
5. HLLV-5	Feb '08	DSM, MOC, MOO	83.2	24.4	10.1
6. HLLV-6 to 20	Mar '08 -Dec '08	TMIS Tank #1-15	91 ea.	24.4	10.1

Flight crew integrates TMIS with MSS. Final checkout. Launch Feb '09 by flight crew.

Total	1729.8 t

Figure 2.1.2.3-1 Total IMLEO, CS-2, Reference Missions




Title: Figure 2.1.3-1. Transportation Program Development Schedule
Page 1 of 1
Revision: A
Date: 6-27-88
1992
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2006
1 Pre-Phase A
2 Phase A
3 Phase B
Exec. Go-ahead
4 Phase C/D
5 Mars Car-1
ETO
6 Hum-1
ETO
7 Precursors:
8 Mars Observer
MRSR
9 Space Station
VGRF
PMC
10 Propellant Depot
11 Transportation:
12 Planetary Aerobrake
13 Man-rated TMIS, TEIS
14 Microgravity CM
Artificial Gravity
15 HLLV
16 MAV Development
Originator: B.C. Clark
Studies
Development
Use In Space

Table 2.1.4-1 Trades/Options—Case Study 2

Options

a	Next earlier opportunity (launch 2002 for piloted)
b	Next later opportunity (launch 2006 for piloted)
c	Conservative propulsion performance (480/470/320)*
d	V. conservative propulsion system performance (480/460/340)
e	Smaller user accommodation (0.2 t Ph, 4.8 t MRSR)
f	Venus probes (3t)
g	v. conserv. LH2/LOX propulsion (LTCSF wet-launched tanks, high boiloff)
h	nom tankage, but high boiloff
i	Hydrocarbon/LOX for TEI
j	Both MOV to Phobos circular orbit (sub-scenario C)
k	Piloted MOV to Phobos circular, MCV in 250 x Phobos elliptical (sub-scenario A)
l	EOC of ETV
m	One-stage TMIS
n	nominal aerobrake (10%)
o	advanced aerobrake (5%)
p	TEI, MOC (if all-prop) using NTR
q	I_{sp} of 900 for NTR
r	ECCV retro-propulsion via reduction of C_3
s	Phobos/Deimos teleoperators in lieu of PhEV
t	
u	Advanced cryoprop (tankage factor of 0.075)
x	OpVs for slipped mission 3

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* TMI/TEI, MOC, MOO/biprop

Table 2.1.4-2 Options Mass Summary—Case Study 2

TIC i.d.		-------- IMLEO --------			----- File -----	
Concept	Trade/Option	Hum	Car	Total	Miss	Ref
TIC-1R	Reference (MAb, 2-TMIS, (485/470/320), nom. tank, low boiloff, PhTele, ECCV)	1124.5	503.3	1627.8	c-HG	c-HG
TIC-1R-a	Next earlier opportunity (2002)	641.1	552.6	1193.7	m-GJ	c-HG
TIC-1R-b	Next later opportunity (2006) ("fallback")	1747.1	708.1	2455.2	m-GK	c-HG
TIC-1R-c	Conservative propulsion performance (480/470/320)	1139.5	509.0	1648.5	m-GL	c-HG
TIC-1R-d	Very conservative performance (480/460/320)	1168.7	519.3	1688.0	m-GM	c-HG
TIC-1R-h	High boiloff	1226.3	568.6	1794.9	m-GN	c-HG
TIC-1R-g	V. conservative tankage (.256), low boiloff	1516.7	665.7	2182.4	m-GO	c-HG
TIC-1R-gh	V. conservative tankage, high boiloff	1682.6	768.3	2450.9	m-GP	c-HG
TIC-1R-u	Advanced tankage (0.075)	916.0	417.1	1333.1	m-GR	c-HG
TIC-1R-ul	Advanced tankage, no boiloff	889.3	402.3	1291.6	m-GS	c-HG
TIC-1R-i	Hydrocarbon/LOX for TEI	1132.9	525.1	1658.0	m-GT	c-HG
TIC-1R-m	1-stg TMIS	1242.7	503.2	1745.9	m-GU	c-HG
TIC-1R-n	Nominal MAb (10%)	1034.3	445.0	1479.3	m-GW	c-HG
TIC-2R	Reference for 2nd mission (Launch 2006)	1769.7	741.8	2511.5	c-HQ	c-HQ
TIC-3R	Reference for 3rd mission (Launch 2009)	1511.8	1113.3	2625.1	m-HH	m-HH
TIC-2R-s	no PhEV on mission 2, but Phobos teleoperator	1748.2	707.5	2455.7	m-HE	c-HQ
TIC-3R-s	no PhEV or DeEV on mission 3, but teleoperators	1482.4	1048.7	2531.1	m-HF	c-HP

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Figure 2.1.4-1 Total IMLEO, CS-2, Options

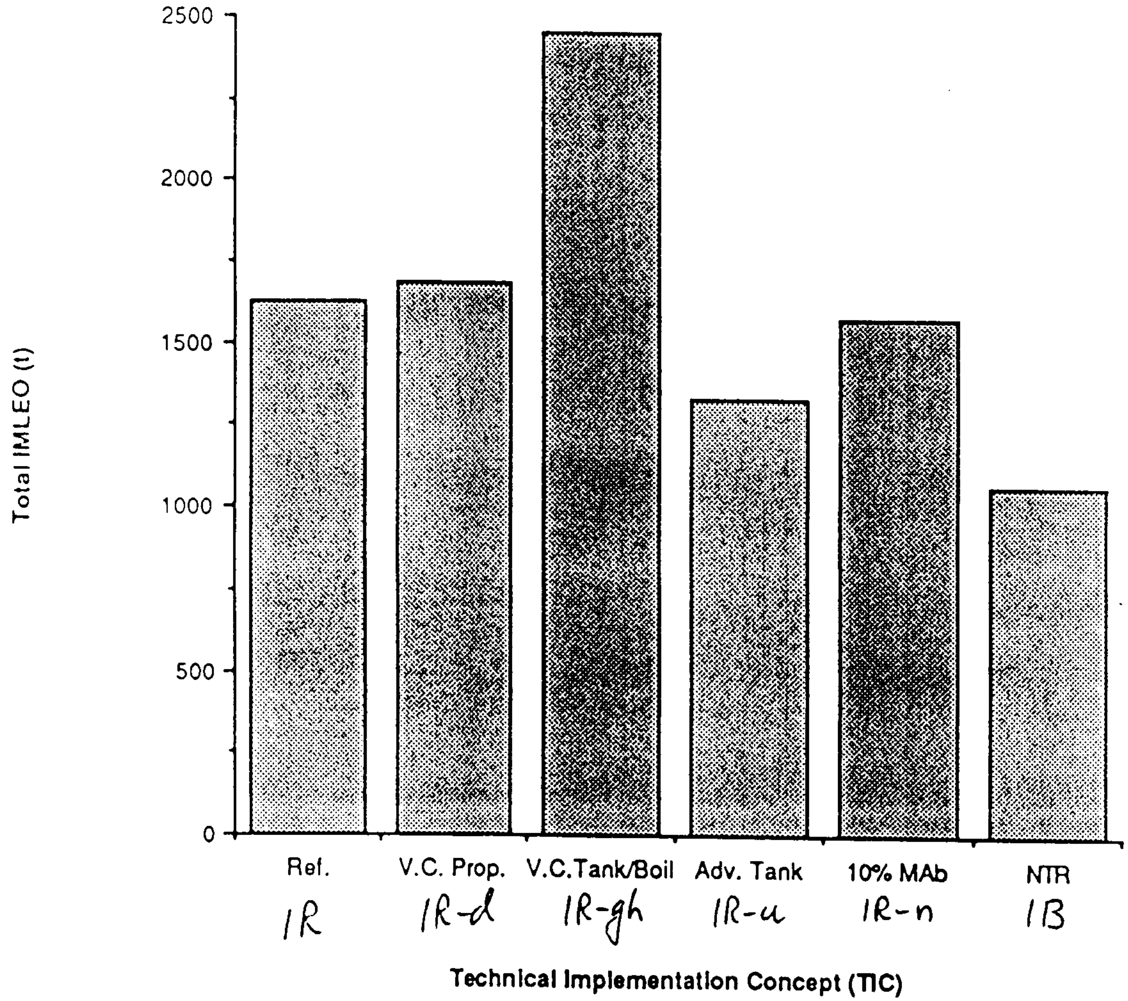


Table 2.1.4-3 Trade: Phobos Human vs Teleoperated Exploration

Trade Issue: Should the human exploration of Phobos and Deimos be accomplished by human flights to and EVAs onto these satellite surfaces,

OR, should the crew deploy specially designed telerobotic scientific exploration spacecraft, to be operated in real time (no Earth communication time-delays) from the Mars Orbiting Vehicle (MOV).

Analysis:

IMLEO:

Teleoperated vehicles are significantly lower in mass; teleoperator has 700 to 1000 kg payload; manned vehicle is an *additional* 1500 to 5000 kg (depending on SPE storm shelter).

Development Costs: The Phobos Excursion Vehicle (PhEV) must be man-rated. Phobos teleoperator will build on FTS development hardware and software.

Numbers: Two teleoperators allows visual observation from remote points. Manned vehicle may desire additional teleoperator spacecraft.

Mission Operations: Exploration via PhEV requires that 2 crew members leave the MOV. Teleoperated exploration allows these crew members to remain available for other critical tasks of MOV health maintenance and support of Mars landed crew.

Mission rule requiring sequential exploration greatly reduces exploration time (from 20 days on Mars surface to one week; minimum sortie time, with 3 days for exploration and 1 day preparation/checkout, to Phobos is one week)

Table 2.1.4-3 (cont.)

Mission Success: Teleoperator spacecraft is simpler than PhEV because no ECLSS.

Scientific Value: On-site human presence enables higher quality perception of Phobos surface characteristics. However, during EVA the human mind is placed under additional stress not present during telerobotic operations. Normal human locomotion on Phobos is impossible.

To achieve near-equality of exploration, teleoperator must contain high quality stereo/color vision. Exaggerated stereo possible with teleoperator.

Human Safety: PhEV requires use of a vehicle that has been dormant for many months, with little time available for crew refresh in actual hardware. Several propulsive operations required. Delayed return of PhEV to MOV could jeopardize MOV access to required TEI launch window (return to Earth).

Multiple EVAs add risk, and will require larger, heavier PhEV to accommodate spacesuit don/doff.

EVA at Phobos surface is technically difficult due to milli-gravity and very near horizon.

EVA at Phobos could be hazardous due to poorly consolidated regolith, including pseudo-fluidization of material, dust hazards (seals, thermal balance, visibility).

Conclusion: Exploration of Phobos (and Deimos) by astronaut-operated telerobotic spacecraft enables lower IMLEO, less mission success risk, more operational flexibility, and eliminates considerable human risk. The science decrement is small or non-existent.

Table 2.2.3-1 Alternative Missions -- Case Study 2

R	Reference mission Split/sprint. Mars aerobrake (nominal; 10%) Advanced propulsion (480/460/320). Dry-launched tanks User accommodation: 1.2 t to Phobos, 7 t MRSR. Separate vehicles for PhEV, ECCV.
A	All-propulsive 481/471/320
B	Nuclear Thermal Rocket (NTR) 850 I_{sp}; TMIs only
C	Conjunction class trajectory
D	All-up vehicle (not split mission)
E	Artificial Gravity

Table 2.2.3-2 Alternatives Mass Summary—Case Study 2

TIC i.d. Concept	Alternatives	IMLEO Hum	IMLEO Car	IMLEO Total	File Miss	File Ref
TIC-1R	Reference (MAb, 2-TMIS, (485/470/320), nom. tank, low boiloff, PhTele, ECCV)	1124.5	503.3	1627.8	c-HG	c-HG
TIC-1A	All propulsive	2632.4	579.2	3211.6	m-HB	c-HG
TIC-1B	NTR (850) for TMI (1-stg)	715.4	352.0	1067.4	m-GY	c-HG
TIC-1B-p	NTR for TMI and TEI	706.6	328.2	1034.8	m-HA	c-HG
TIC-1DE	All-up, artificial gravity, 2004 OpVs			1061.9	c-GU	c-GU
TIC-2CDE	All-up, art-g, 2007 Conj			976.5	c-GY	c-GY
TIC-3CDE	All-up, art-g, 2009 Conj			1017.9	c-GZ	c-GY
TIC-4CDE	All-up, art-g, 2003 Conj			1068.5	c-HA	c-GF
TIC-5CDE	All-up, art-g, 2005 Conj			1126.4	c-HB	c-GC
TIC-4ACDE	All-up, art-g, 2003 Conj, all propulsive			1378.6	c-HC	c-HA
TIC-4CDE	All-up, art-g, 2003 Conj, 2 MDVs			1292.1	c-HD	c-HA
TIC-4CDE	All-up, art-g, 2003 Conj, Science rich			1090.9	c-HI	c-HA
TIC-4CDE	ditto, 2 MDVs			1323.7	c-HJ	c-HJ

Figure 2.2.3-1 Case Study 2, Non-Sprint Compared to Sprint

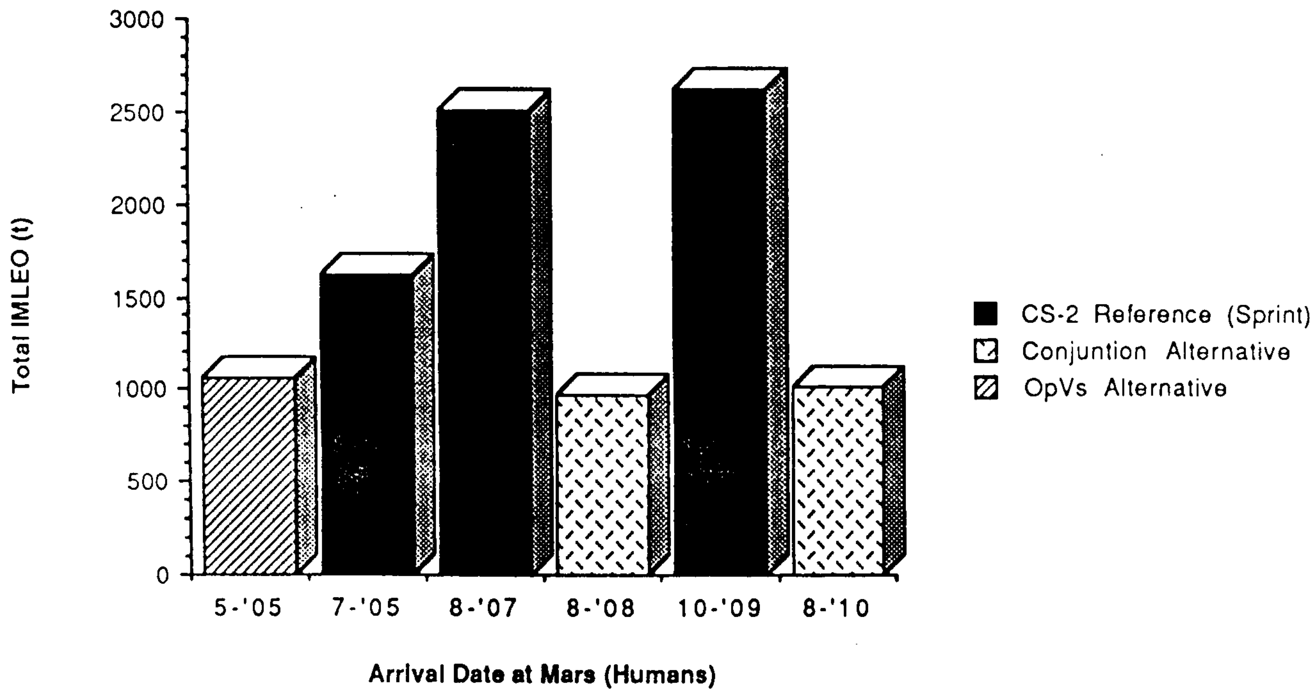


Figure 2.2.3-2 CS-2, Conjunction Alternatives

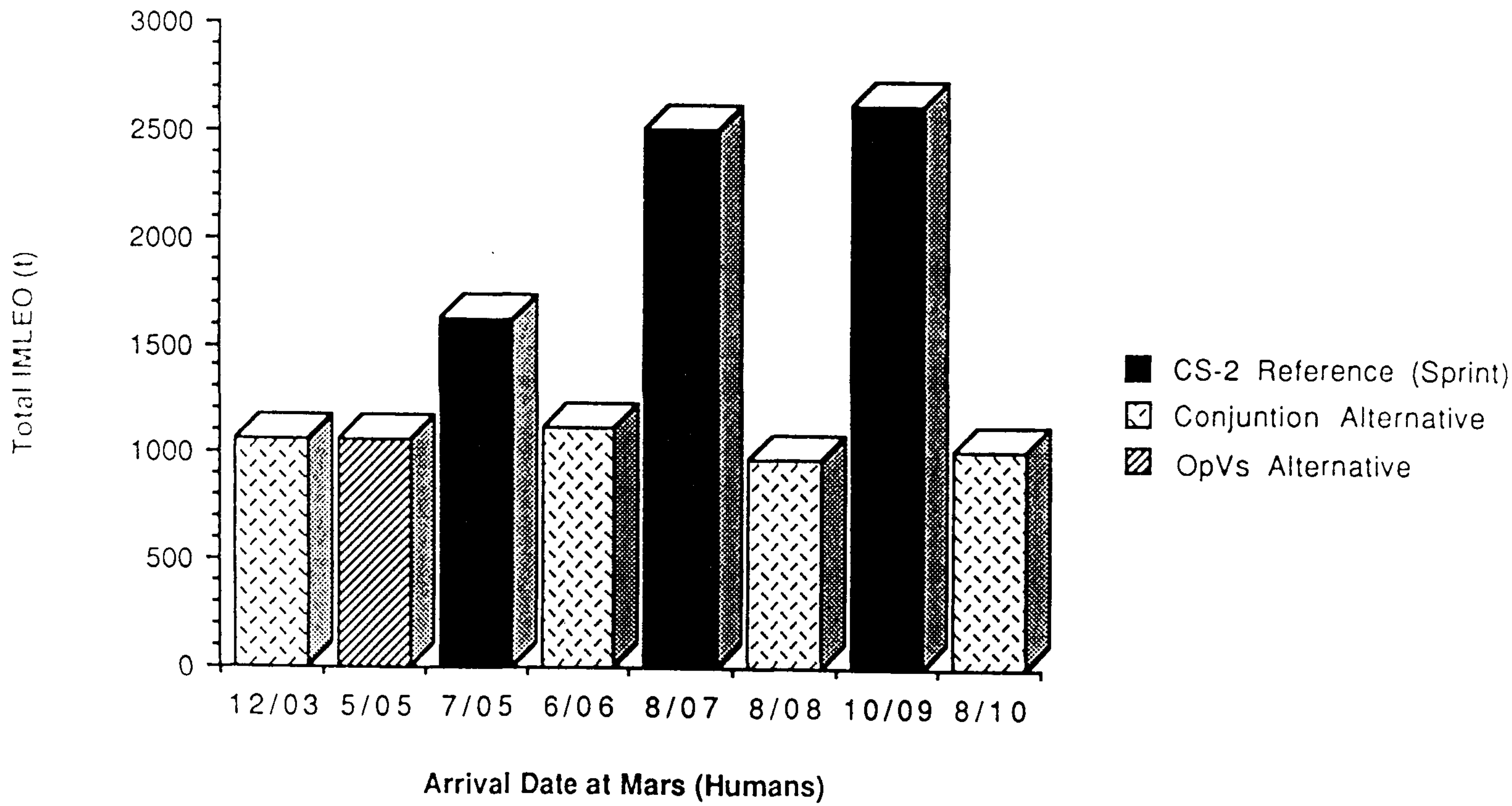


Figure 2.2.3-3a Artificial Gravity Concept

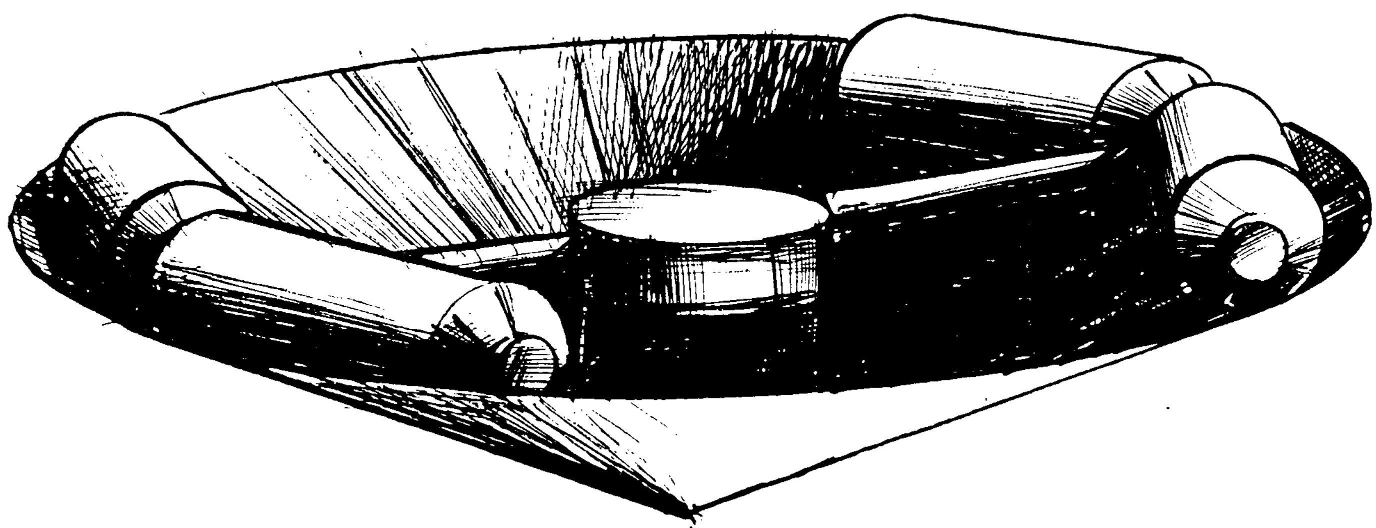


Figure 2.2.3-3b Artificial Gravity Concept

CS-2 (Conjunction Alternative)

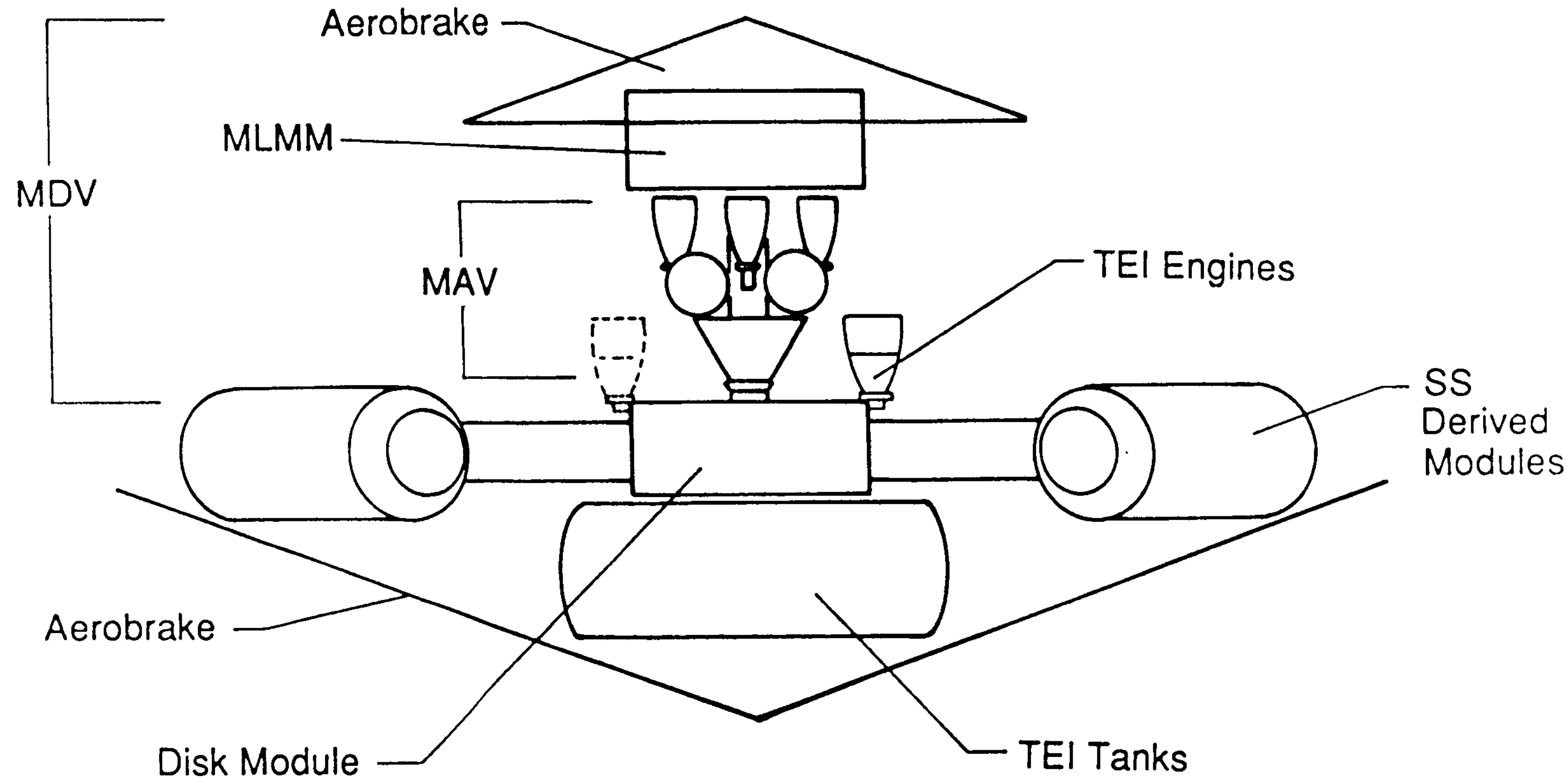


Figure 2.2.3-3c Artificial Gravity Concept

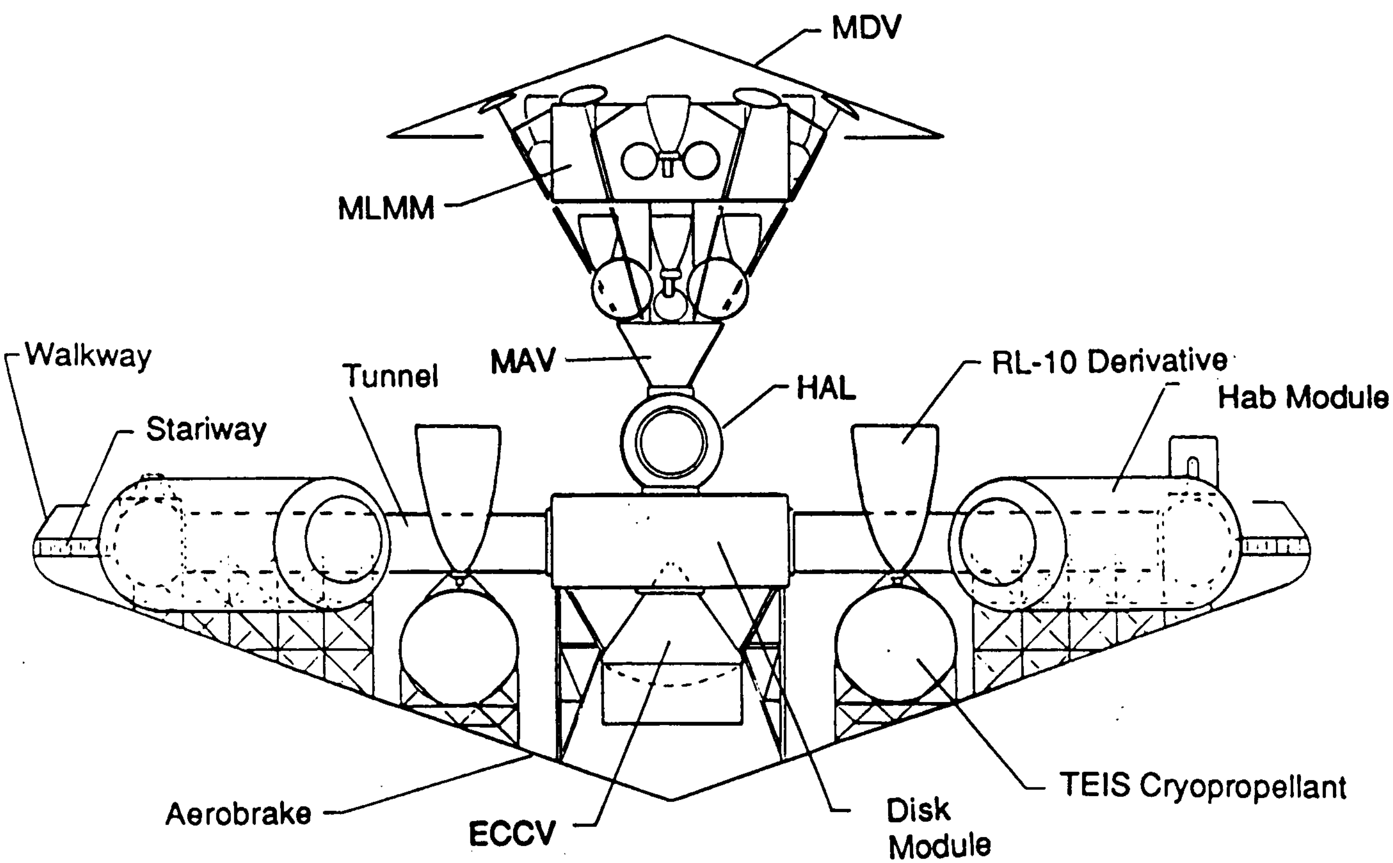


Figure 2.2.3-3d Artificial Gravity Concept, End View

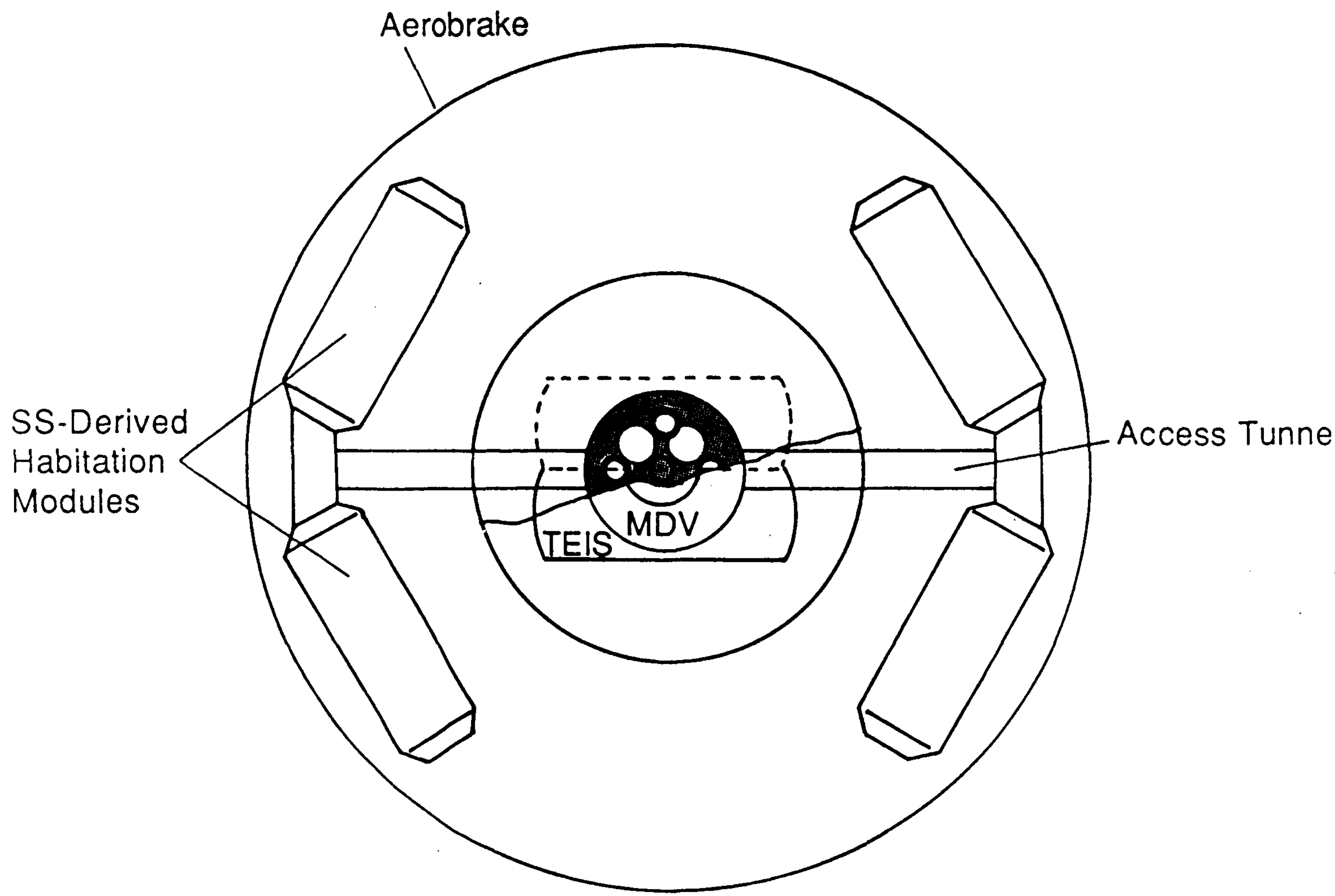


Figure 2.2.3-3e Artificial Gravity Concept

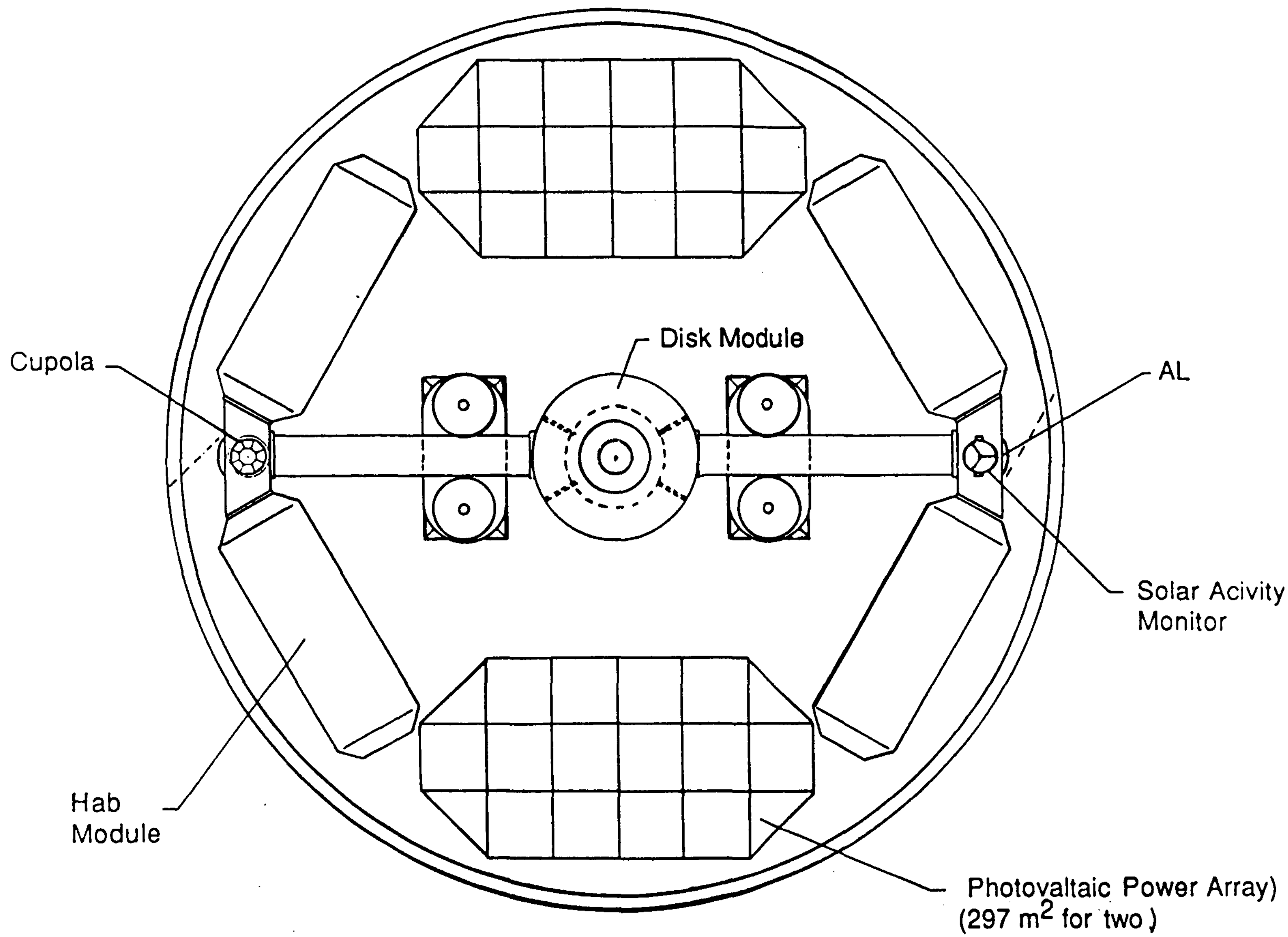


Figure 2.2.3-3f Artificial Gravity Concept, with 1 MDV

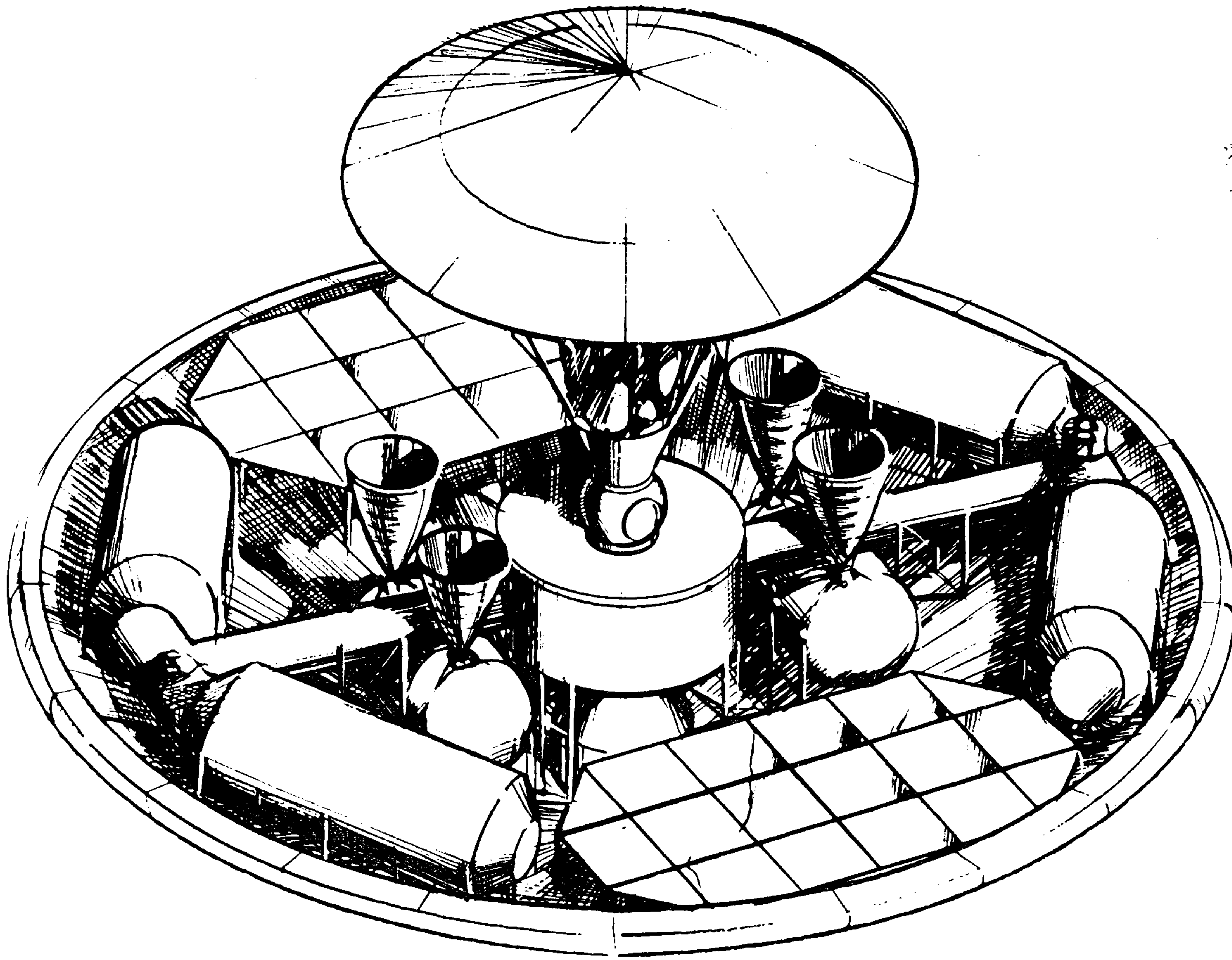


Table 2.2.3-3 Artificial Gravity

Radius	Rot. Rate	Accel.	Tang. Vel.	Vel. g-effect (1 m/s)
55 ft	6 rpm	0.675 g	23.6 mph	0.55-0.81 g
	5	0.47	19.6	0.37-0.58
	4.5	0.38	17.7	0.29-0.48
55	6	0.675		
25	6	0.31		
12.5	6	0.15		
5	6	0.06		
2	6	0.025		
55	4.5	0.38		
25	4.5	0.17		
12.5	4.5	0.086		
5	4.5	0.035		
2	4.5	0.014		

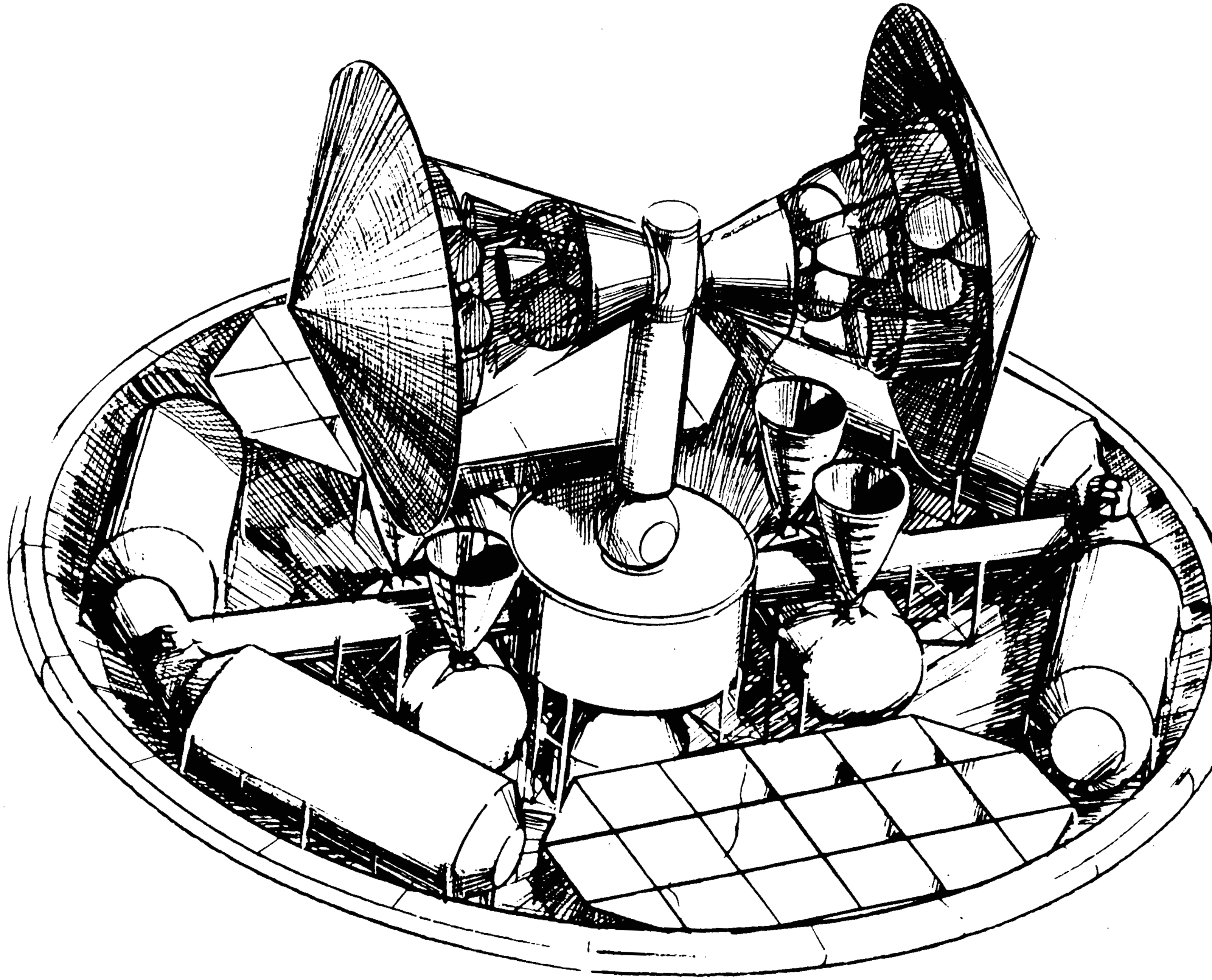
Propulsion Requirements:

6 rpm, ΔV = 5.3 to 10.5 m/s. Mass expenditure = 0.17 to 0.34% (I_{sp}=320)
4.5 rpm, ΔV=4 to 7.9 m/s. Mass expenditure = 0.13 to 0.25%

Table 2.2.3-4. Comparison of Artificial-g/Conjunction-2003 vs Split/Sprint-2004 Missions

Item	Art-g/Conj [c-HI]	Split/Sprint [c-HG]	
IMLEO	1090.9 t	1627.8 t	(split: 503.3 cargo, 1124.5 human)
Total trip time	957 d	440 d	
Interplanetary time	204/191 d	265/144 d	
Gravity environment	up to 0.64 gee	microgravity	
Arrival date	29 Dec 03	2 Jul 05	
Time at Mars	562 d	30 d	
Time on Martian surface	180 sols	20 sols	
TEI propulsion system	integrated	rendez in Mars orbit	
Spaceborne			
No. of SS-derived modules	4	3	
Disk module	31 ft dia.	25 ft dia.	
Other hab space	ECCV MDV MAV tunnels	ECCV	
Total pressurized volume	1271 m^3	737 m^3	conj return leg is 1033 m^3
Power			
Type	fixed	retract/extend	
PVPA area	297 m^2	300 m^2	
Ph/D Teleoperators	4000 kg	2000 kg	conj has 2 ea. PhDTele
MarsSciSat	2000 kg	1000 kg	conj has 2 ea. sats
MRSR (Rovers/sample return)	two 7000 kg	one 3500 kg	
Lander			
Habitat size	31 ft dia.	25 ft dia.	
MLOE			
Science Equipment	3800 kg	3300 kg	
Teleoperated Equip	2000 kg	2000 kg	
Rovers	two, press.	two, unpress.	
mass	3200 kg	810 kg	
range	100 km	10 km	
Construction Equip.	1500 kg	150 kg	
Manufacturing Demo	1000 kg	100 kg	
Landed power	4.5 kW_e	3.5 kW_e	
batteries	50 kWh	25 kWh	
Environment			
Dust storm season?	no (L_s=50°)	possible (L_s=325°)	
Recovery of spaceship (at Earth)	yes	no	
Mass penalty for 2nd MDV	232.8 t	not practical	(for 64.9 t MDV)

Figure 2.2.3-4 Artificial Gravity Configuration with 2 MDVs



Case Study 3
Lunar Observatory

3.0 Transportation Systems Definition

In order to set up an observational station on the lunar farside, several transportational elements will be required. These include a Lunar Transfer Vehicle (LTV) to transfer humans and cargo to low lunar orbit (LLO), a Lunar Descent Vehicle-Piloted (LDV-P) to transfer the crew from LLO to the lunar surface, and a Lunar Descent Vehicle-Cargo (LDV-C) to transfer cargo from LLO to the lunar surface.

3.1 Elements and Systems Description

3.1.1 Transportation Requirments/Assumptions

The lunar observatory is to be set up over a two-year period starting in the year 2000 by using two cargo and two piloted missions. After the observatory is deployed, piloted missions will occur annually. The lunar observatory site will have to be revisited every three years for servicing of the observatory. The other piloted missions will be visits to other areas of interest on the lunar surface. The mission phases are outlined in Table 3.1.1-1 and are examined in more detail in Table 3.1.1-2

The basic assumptions of the lunar observatory transportation system are summarized in Table 3.1.1-3. Each mission assumes a crew of four staying for the length of a single lunar day (14 Earth days), with a safe-haven capability of 55 days. This 55 day period is the assumed minimum rescue time for a stranded lunar vehicle. Additional safety factors include a flyby abort capability for the Lunar Transfer Vehicle, and solar flare protection on the lunar surface.

One of the main features of the transportation system, as specified for this case study is its complete discardability. All of the LTVs will be left either in LEO or in LLO, and new LTVs will be launched from the Earth's surface for each new mission. In addition to this, the LDV-Cs will be left on the lunar surface as will most of the LDV-Ps (only the Lunar Ascent Vehicle (LAV) will return to LEO with the astronauts). This eliminates the problems of refurbishment and refueling of the spacecraft, but also increases the mass that must be lifted into LEO.

3.1.2 Reference System Description

In the baseline scenario, which is outlined in Table 3.1.2-1, the LTV will depart from LEO with the LDV-C attached to the front of the aerobrake (Fig. 3.1.2-1). When this vehicle reaches LLO, the LDV-C will detach from the LTV and make a propulsive descent to the lunar surface, while the LTV will remain in LLO.

The crew will then travel to the lunar surface in a similiar fashion. The LTV will depart from LEO with the LDV-P attached to the front of the aerobrake (Fig. 3.1.2-2a and -2b). As before, when this vehicle reaches LLO, the LDV-P will detach and descend to the surface. After the crew has stayed for 14 days, they will ascend back into LLO in the LAV, which is a part of the LDV-P. The LAV will rendezvous and dock with the LTV, (Fig. 3.1.2-3), which will then transfer the crew back to LEO, using the Earth's atmosphere to aerocapture into LEO. The crew will rendezvous with Space Station in LEO and return to the Earth's surface on the Space Shuttle. The missions using the LDV-C will only occur when setting up the observatory. All flights after that will use only the LDV-P. See Figure 3.1.2-4 for a pictorial of this scenario, and Table 3.1.2-2 for the velocity requirements.

3.1.2.1 Configuration and Mass Allocations

The Lunar Transfer Vehicle is shown in Fig. 3.1.2.1-1a, -1b, and -1c (expanded view). As in all of the vehicles in this case study, it uses LOX/LH_2 propellants and RL-10 engines. This vehicle consists essentially of 4 propellant tanks, 4 engines, an aerobrake, and avionics. A propellant tankage factor of 10% was assumed in sizing the tanks of all of the vehicles in this case study. Figure 3.1.2.1-2 illustrates how ceramic attachment points on the front of the aerobrake will be used to attach both the LDV-P and the LDV-C.

The Lunar Descent Vehicle-Cargo (LDV-C) is shown in Figure 3.1.2.1-3a and -3b. The vehicle is essentially a large cargo bay with a volume capacity of 370 cubic meters. The LDV-C carries 17.5t of payload to the lunar surface by using 4 RL-10's and LOX/LH_2 propellant. The diameter of the cylindrical cargo bay is 7.6 meters with a height of 10 meters. The descent can still be accomplished if one engine fails, by turning off the engine on the opposite side of the failed engine.

The LDV-C employs an elevator to lower the cargo to the lunar surface from the cargo bay. This negates the need for cranes and other unloading equipment, thus minimizing the surface infrastructure required. This translates directly to a savings in mass in LEO.

The Lunar Descent Vehicle-Piloted (LDV-P) is shown in Figure 3.1.2.1-4. It is based on the Mars Descent Vehicle (MDV) in Case Study 2. The bottom level of the LDV-P is a 7.6 meter diameter, 3 meter high habitation module where the crew will live for the 14 day staytime. Braced on top of the module is the Lunar Ascent Vehicle (LAV) which carries the crew back to LLO. The LDV-P makes a descent to the lunar surface using 4 RL-10 engines and LOX/LH_2 propellant. Again, the descent can be made if one engine fails, by turning off the engine on the opposite side of the failed engine.

The Lunar Ascent Vehicle (LAV) is shown in Figure 3.1.2.1-5 and is based on the Mars Ascent Vehicle in Case Study 2. The LAV is a Gemini-based, light-weight capsule which carries the four crew members from the lunar surface to LLO and rendezvous with the LTV waiting in orbit. It uses RL-10 derivative engines which don't have as high a thrust level as the ones used for the LTV, LDV-P, and LDV-C. This vehicle allows a sufficient thrust level for an engine-out capability.

3.1.2.2 ETO, On-orbit Assembly, and Servicing Needs

Assuming an HLLV capability of 91 t, a minimum of 6 launches will be required to initially set up the lunar observatory during the years 2000 and 2001. The ETO manifest is given in Table 3.1.2.2-1. After the initial set-up, only two HLLV launches a year will be required to maintain the observatory and investigate new sites on the lunar surface.

Details of each launch and any required on-orbit assembly are given in Table 3.1.2.2-2. The HLLV lift capability will allow most of the transportation elements to be sent up in a single launch, thus requiring no on-orbit assembly. The LDV-Cs and LDV-Ps will be launched into LEO fully assembled and fueled. The LTVs will be sent up fully assembled also, but they will not be fueled. Cryogenic propellant will be sent on a separate launch and in-space propellant transfer will be necessary to fuel the LTVs. The LDV-Ps and the LDV-Cs will also need to be attached to the LTVs in orbit. This can be accomplished telerobotically, or by the crew of the Space Station. STS flights will be necessary to bring up the crew from the Earth's surface at the start of the mission and to bring the crew back down to the surface at the end of the mission.

3.1.3 Transportation Program Development Schedule

The schedule for development and flight-testing of the transportation system as well as the schedule for required precursors is shown in Figure 3.1.3-1.

3.1.4 Trades/Options

One important tradeoff that was evaluated was reusability of the LTVs. By refurbishing and refueling the LTVs rather than launching a new vehicle for every mission, there is a savings of almost 14 t per mission that does not need to be lifted off the Earth's surface. In addition, the LTVs must be fueled in orbit anyway, so there is no real benefit to discarding them.

Also, if the LTVs from the cargo missions were to be reused, an additional 11 t of propellant would be required to return the LTV to LEO. This mass penalty is still more than balanced by the savings in mass from not needing to launch more LTVs. There is also a manufacturing cost savings, since only one LTV needs to be produced. Finally, discarding LTVs will result in a build-up of significant amounts of risidual hardware in LEO and LLO, resulting in hazardous orbiting conditions.

The major alternative of using direct descent to the lunar surface (eliminating LLO as a stopover) is discussed in section 3.3.

3.2 Enabling Technology Needs

3.2.1 Propulsion System

Increased performance of the RL-10 engines must be verified. The feasibility of small, very low boiloff cryogenic propellant tanks must be evaluated.

3.2.2 Aerobrake Technology

The feasibility of using an aerobrake to capture the LTV into LEO must be evaluated. An aerobrake must be designed and flight-tested. If an effective aerobrake cannot be designed, propulsive breaking options will have to be considered. Also, the ability of the LDV-Ps and the LDV-Cs to be attached to the front of the aerobrake using ceramic hardpoints must be verified.

3.2.3 Precursor Missions

Communication satellites will need to be set up in LLO. Precursor missions will need to determine the location of the site for the lunar observatory, as well as other sites of interest for future missions.

3.3 System Alternatives and Opportunities

One major alternative to the baseline lunar observatory mission scenario, would be to eliminate the stopover of the space vehicles in LLO and in LEO. A direct descent to the lunar surface and a direct Earth return would allow a mass savings of over 17%. In addition to that, this alternative would allow the astronauts to return directly to the Earth's surface. No STS flight would be necessary to return the crew to Earth. A comparison of the total mass in LEO for the reference mission versus the direct alternative is given in Table 3.3-1 and the velocity changes for the direct alternative are listed in Table 3.3-2. A summary of the alternatives along with a mass comparison is shown in Table 3.3-3.

Direct re-entry at Earth could also be used in either the baseline scenario or in the recoverable LTV option. In both of these cases, the crew would separate from the LTV and descend directly to the surface. This would eliminate the necessity of rendezvous with the Space Station as well as save a shuttle mission that would be needed to recover the astronauts. The LTV could still be aerobraked into LEO and be recovered for refurbishment and refueling.

Table 3.1.1-1 Mission Phases—CS-3

Lunar Observatory Missions

A.	Pre-launch	
B.	Earth orbital	
C.	Earth escape	(TLI)
D.	Transfer to Moon	
E.	Lunar orbital capture	(LOC)
F.	Lunar orbital/landed	
	Subphase 1 - Pre-landing	
	Subphase 2 - Lunar Landing	(LL)
	Subphase 3 - Post-landing	
	Subphase 4 - Ascent, Rendezvous, and Docking	(ARD)
	Subphase 5 - Post-ascent	
G.	Lunar escape	(TEI)
H.	Transfer to Earth	
I.	Earth capture/recovery	(EOC)
J.	Post-Landing on Earth	

Notes:
- A. Exec. authoriz.; Program office levels I, II, III; Contractor phases A/B, C, D; Crew selection/training
- B. On-orbit assembly, checkout, fuel-up
- H. Orbital capture; direct entry or rendezvous and transfer
- J. Hardware inspections and test; Astronaut de-briefings, re-assignment to future programs or other activities

Table 3.1.1-3 Transportation Requirements and Assumptions

Requirements from the SRD

- Chemical propulsion
- 2 Setup flights in 2000 (1 cargo, 1 crew); 2 setup crew flights in 2001 (1 cargo, 1 crew);
  1 operational crew flight every year thereafter
- Observatory instruments require servicing every 3 years; crew visits other sites between servicing missions
- Roundtrip crew mission time of 20 days; less than 14 days at the lunar surface
- No node beyond LEO
- Flyby abort capability
- Landing footprint of 30 meters
- Solar flare protection and "solar monitoring"
- Safe haven for 55 days
- User allocation: 17.5 metric tons per cargo flight, 6.5 metric tons per crew flight
- Crew of 4

Assumptions for Baseline

- All propellants are LOX/LH_2
- All engines are RL-10, I_{sp} = 460 seconds
- No boiloff of cryopropellant
- Aerocapture of Lunar Transfer Vehicle (LTV) into Earth orbit
- Rendezvous with Space Station
- Fully discardable transportation system

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- Rendezvous with Space Station
- Fully discardable transportation system

Table 3.1.2-1 Options Selected for Case Study 3

Scenario TIC-1R

OPTIONS

Options Selected: (Date & Your Name)

proven, or under development -->> unproven, or must be developed / analyzed

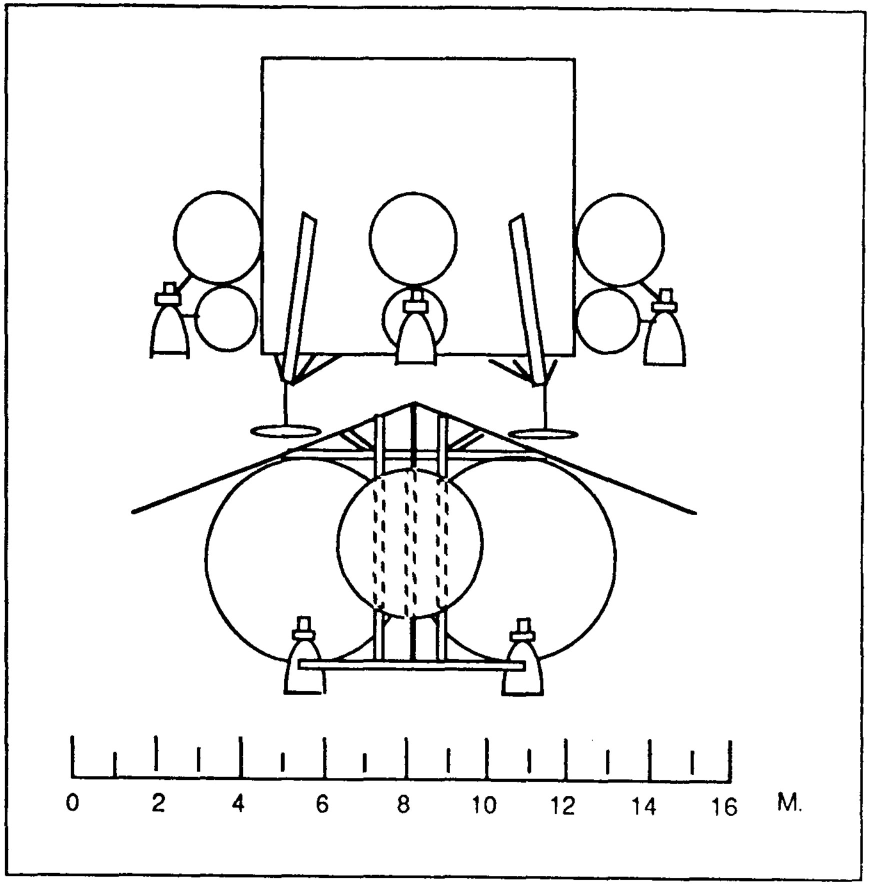
Earth departure location	LEO	HEO	GEO	L1,L2				
On-orbit assembly	**SS, attached**	SS, free-flyer	Min. SS	no SS req'd.				
Hardware staging	integrated	**split, Hab. mod**	**split, LLSE**	split, TEIS				
Launch dates	1990's	**2000's**	2010's	2020's	2030's			
Crew size, total	3	**4**	5	6	7	8	9-11	12-24
Cabin pressure	4.3 psi	**14.7 psi**	10.2 psi					
Gravity environment	**microgravity**	artificial gravity						
Rotation rate	**0 rpm**	1 rpm	2 rpm	4 rpm	6 rpm			
Radiation protection	none	**one storm shelter**	two storm shelters	GCR shield				
Science equipment	**spaceborne**	**Lunar orbit**	**Lunar surface**					
ECLSS, spaceborne	consumables	SS ECLSS	**water recycling**	low mass/power	CELSS			
TLI launch propellant	hybrid chemical	**LH2/LOX**	Hc/LOX	waste/LOX	SEP	NTR	NEP	solar sail
engines, cryo.	**RL-10 growth**	F1 derived	SSME derived	advanced cryo				
growth	**max.-sized tank**	stretch	cluster					
reusability	**non-recoverable**	engines, avionics	all recoverable					
recovery method	**none**	turn-around	re-encounter					
Cryoprop storage	**passive**	active, refrigeration	active, reliquifaction	from H2O				
Power, spaceborne	PVPA	**fuel cells**	RTGs	nucl. reactor	DIPS	solar th.-dy.		
TL abort capabilities	**Lunar swingby**	propulsive abort						
Lunar orbit	none; direct descent	**LLO**	L2					
Satellites Relay Com.	none	**LLO**	L2					
Lunar Science Orbiters	**none**	polar	circ.	elliptical				
Unmanned Landers	**none**	penetrator(s)	rover(s)	sample return				

	OPTIONS (cont.) proven, or under development — new developments — unproven, or must be developed / analyzed						
Number of LDV's	none			one			two
Time on surface	0	1 wk.	2 wk.	6wk.	6 mo.	1 yr.	PMP
Crew size, landed	0	2	3	4	5		6-20
Propellants, LDV	LH2/LOX	Biprop		H2O2			CH4/LOX
LELS	de-orbit prop.			terminal prop.			hover/translate
Landing hazard	large, safe areas			pinpoint landing			terminal H.A.
Power generation	RTE	fuel cells	PVPA	DIPS	nucl. reactor		solar th.-dy.
Power storage	batteries, Ni-H			Regen. FC			HEDRB
ECLSS, Lunar landed	consumables	SS ECLSS		low mass/power	ISCP		CELSS
LLSE	RVR	analyt. eq.	geopys. pkg.	meteorol. pkg.	biol. eq.		drill rig
LLOE mass	none	0.5 t	1 t	2 t	4 t	8 t	16 t
RVR, manned	none		unpressurized		press., 5 day		press., 20 day
ISPP	none						LOX
ISRP demos	none			GOX			LLOX
Propellants, LAV	UDMH/N2O4		LOX/LH2		solids		ISPP
Earth orbit capture ETV	not recovered	direct entry	propulsive braking		aerobraking		prop/aero hybrid
ECCV	none	Ab		prop. braking	direct entry		ETV orb. capture
Orbital recovery	STS to SS		STV to SS				STS to Earth

Form Revised by: S. Geels 6-16-88

Figure 3.1.2-1 Lunar Transfer Vehicle (LTV), With LDV-C Payload

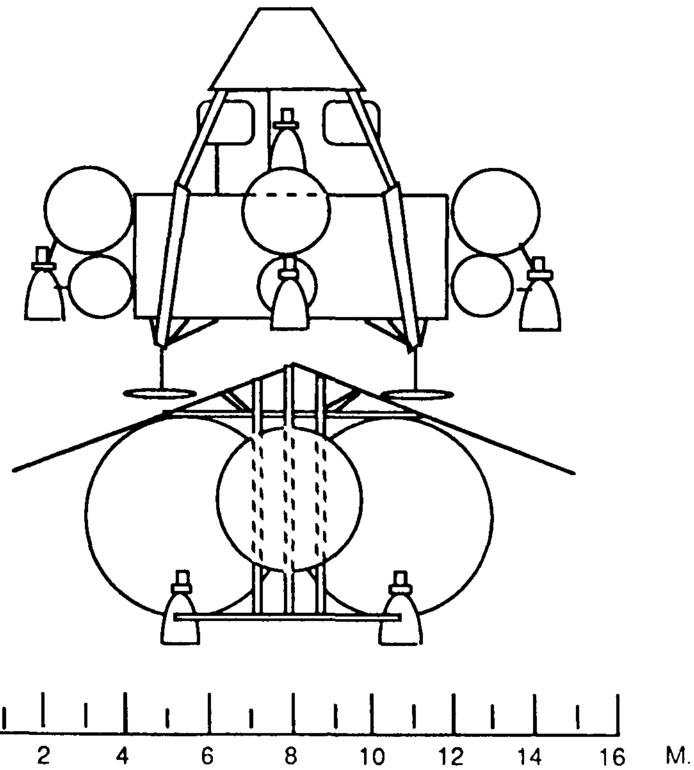
Dry Mass	
(includes wet payload)	47,531 kg
Payload Mass	
(LDV-C)	34,287 kg
Payload Volume	
(cyl. - 8m dia., 10m ht.)	500 m^3
Propulsion System	
Propellant Type	LOX/LH_2
Engines	
Number	4
Type	RL10B-2
Mass (ea.)	191 kg
Thrust (total)	392 kN (88.1 klb_f)
I_{sp} (460 sec)	4.51 kN-s/kg
Propellant Mass*	77,916 kg
Tank Mass	8,086 kg
Initial T/W_E	0.32 @ 100% throttle
Mass Fraction	
Transfer to Moon from LEO	0.406
Transfer to Earth from LLO	0.798
Total Mass	125,447 kg



*Includes 74,553 kg for transfer to Moon; 3,363 kg for LTV return

Figure 3.1.2-2a Lunar Transfer Vehicle (LTV), With LDV-P Payload

Dry Mass	
(includes wet payload)	48,084 kg
Payload Mass	
(LDV-P)	34,840 kg
Payload Volume	
(cone - 8m dia., 10m ht.)	170 m^3
Propulsion System	
Propellant Type	LOX/LH_2
Engines	
Number	4
Type	RL10B-2
Mass (ea.)	191 kg
Thrust (total)	392 kN (88.1 klb_f)
I_{sp} (460 sec)	4.51 kN-s/kg
LTV Propellant Mass*	80,859 kg
Tank Mass	8,086 kg
Initial T/W_E	0.31 @ 100% throttle
Mass Fraction	
Transfer to Moon from LEO	0.406
Transfer to Earth from LLO	0.798
Total Mass	128,943 kg



*Includes 76,631 kg for Transfer to Moon; 4,228 kg for Return with LAV

Figure 3.1.2-2b LTV Transportation of LDV-P

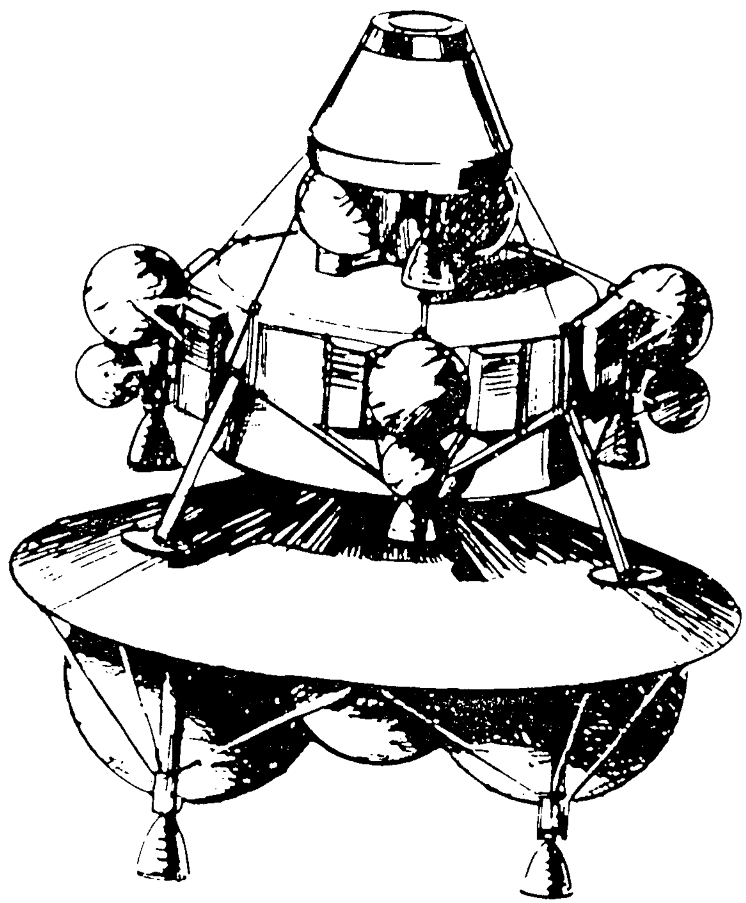


Figure 3.1.2-3 Lunar Transfer Vehicle (LTV) With LAV Payload

(Transfer from LLO to LEO)

Dry Mass (includes Payload)	16,652 kg
Payload Mass (LAV)	3,408 kg
Payload Volume (cone - 3.6m dia., 2.3m ht.)	$8 m^3$
Propulsion System	
Propellant Type	LOX/LH_2
Engines	
Number	4
Type	RL10B-2
Mass (ea.)	191 kg
Thrust (total)	392 kN (88.1 klb_f)
I_{sp} (460 sec)	4.51 kN-s/kg
Propellant Mass	4,228 kg
Tank Mass	8,086 kg
Initial T/W_L	12.3 @ 100% throttle
Mass Fraction	
Transfer to Earth	0.798
Total Mass	20,880 kg

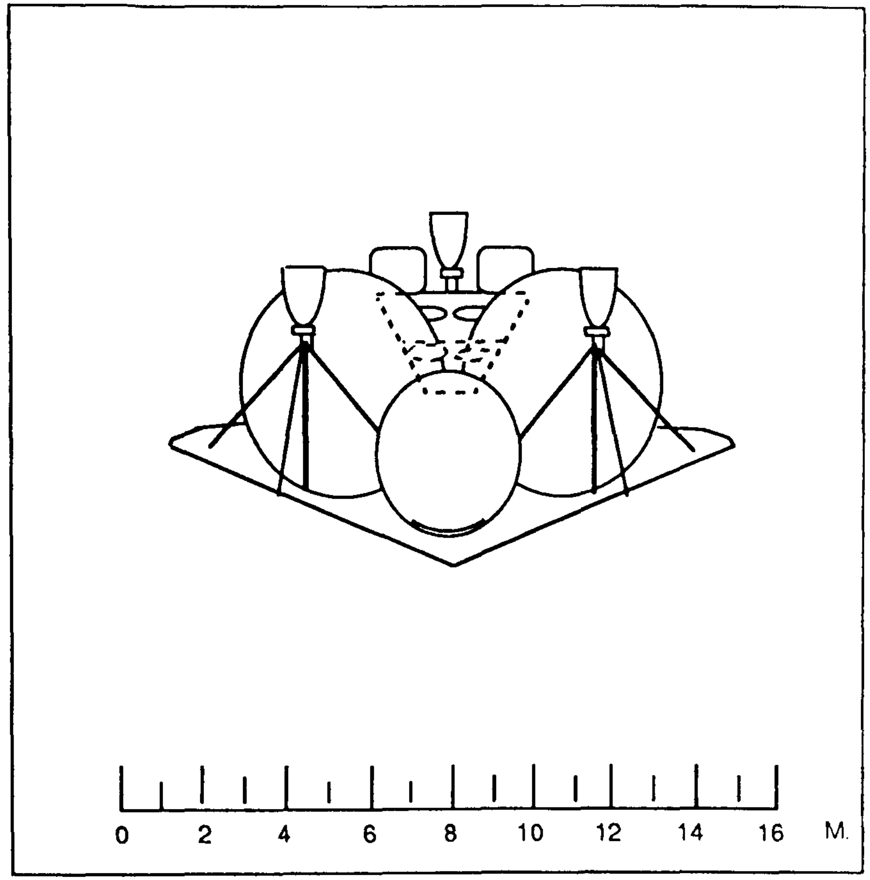


Figure 3.1.2-4 Case Study - 3 Lunar Observatory

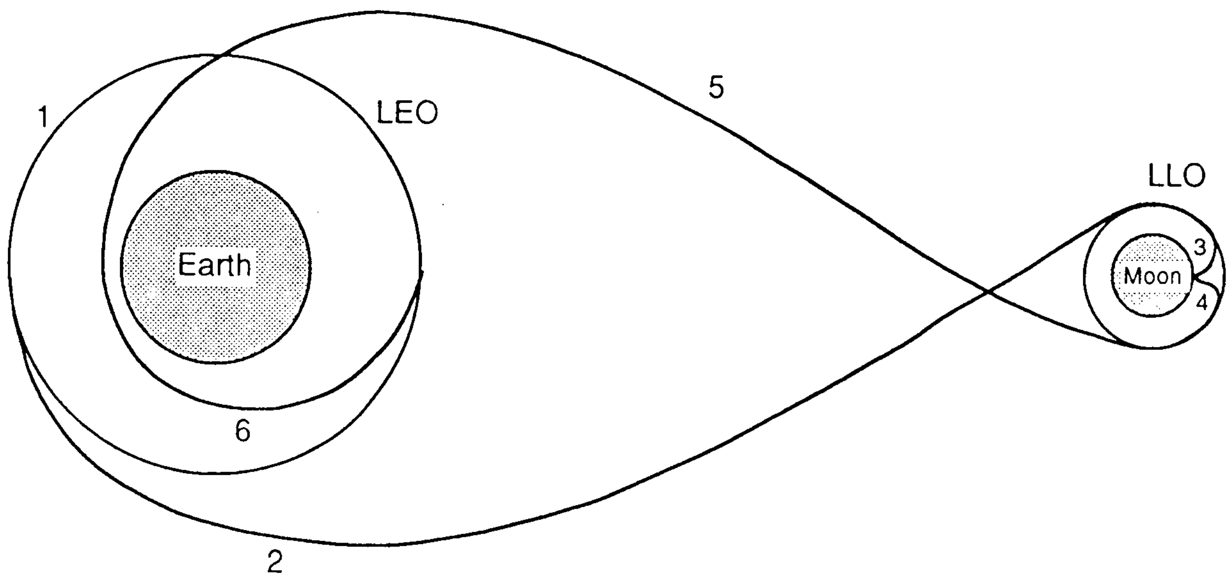


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Table 3.1.2-2 ΔV Requirements for Lunar Scenario

	ΔV (m/s)
TLI	3150
Mid-course correction	50
LOI	867
Total, outbound	4067
Lunar landing	2100
Ascent to LLO	2100
TEI	820
Mid-course correction	50
Aerobrake at Earth	---
Post-aeroassist circularize	150
Total, inbound	1020
Round trip total	9287

Figure 3.1.2.1-1a Lunar Transfer Vehicle (LTV)

(Aerobrake Provided for Earth Return)

Dry Mass	13,244 kg
Propulsion System	
Propellant Type	LOX/LH_2
Engines	
Number	4
Type	RL10B-2
Mass (ea.)	191 kg
Thrust (total)(88.1 klb_f)	392 kN
I_{sp} (460 sec)	4.51 kN-s/kg
Max. Propellant Capacity	80,859 kg
Tank Mass	8,086 kg
Total Mass	94,103 kg

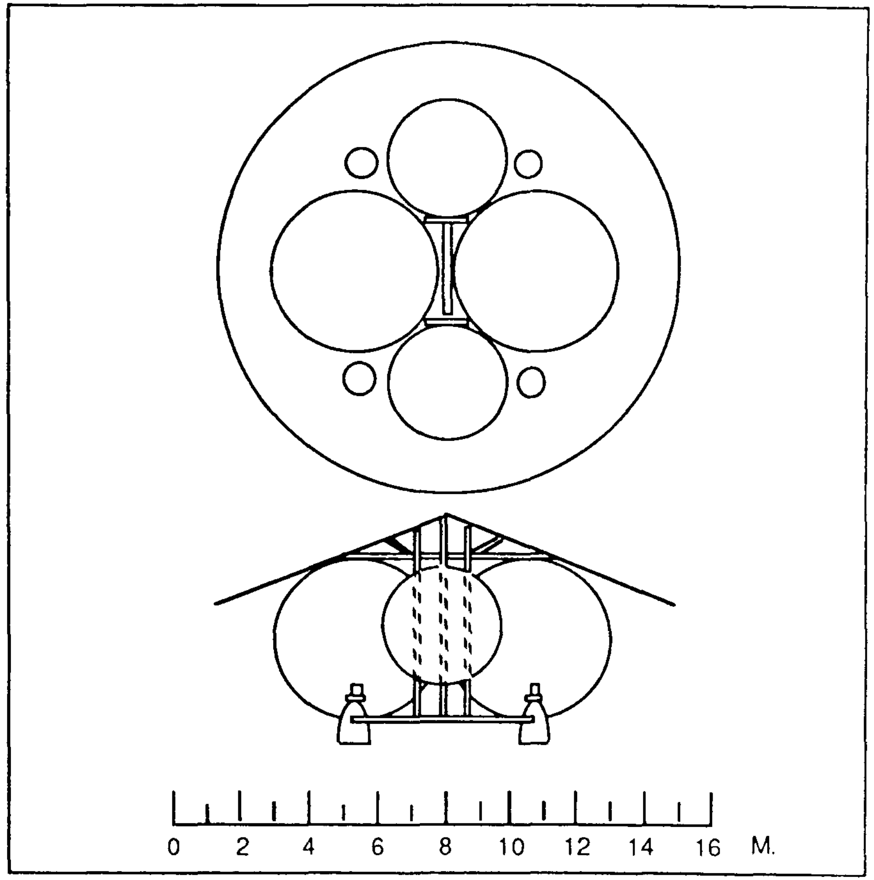
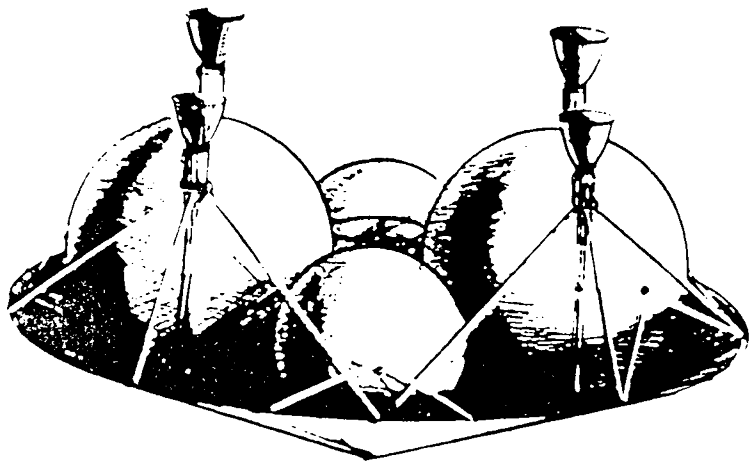
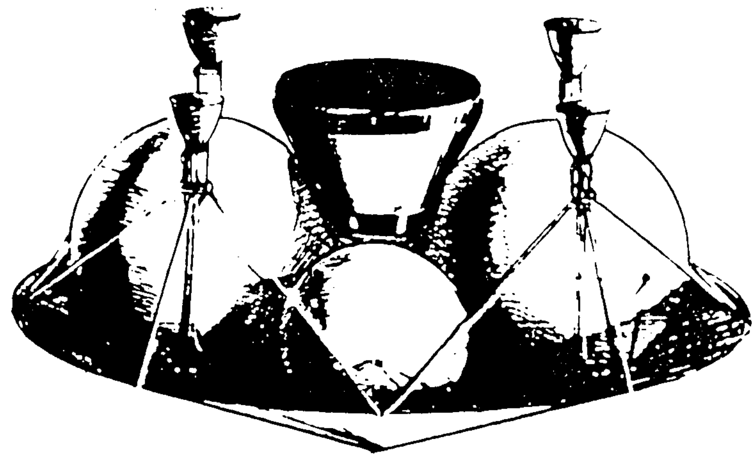


Figure 3.1.2.1-1b Lunar Transfer Vehicle (LTV)



LTV Return to Earth



Crew Return to Earth

Figure 3.1.2.1-1c Lunar Transfer Vehicle

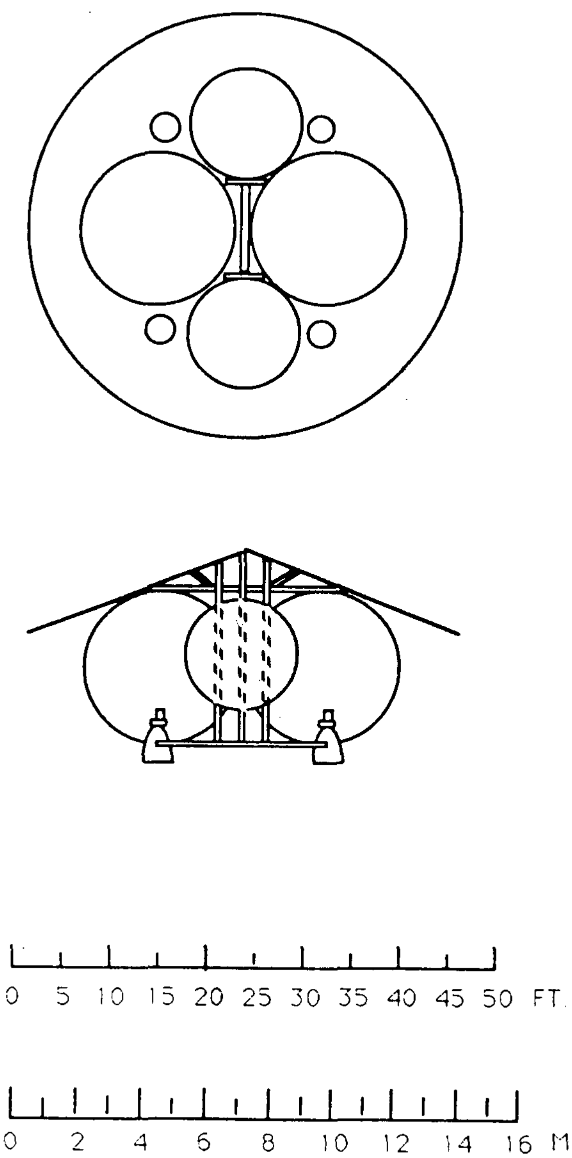


Figure 3.1.2.1-2 LDV Attachment to LTV Aerobrake

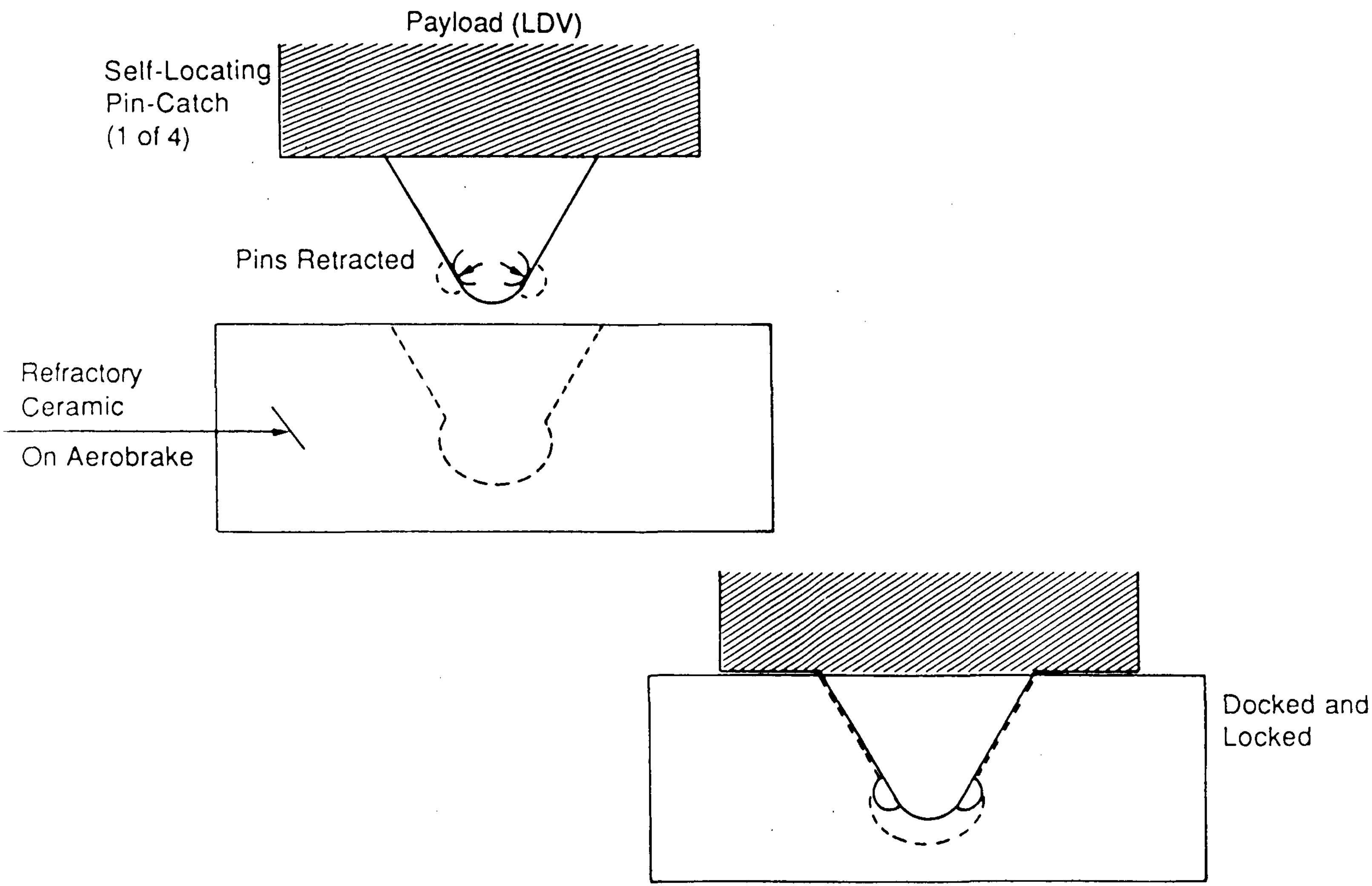


Figure 3.1.2.1-3a Lunar Descent Vehicle-Cargo (LDV-C)

Dry Mass	
(includes payload)	21,519 kg
Payload Mass	
(user payload)	17,500 kg
Payload Volume	
Cargo Bay	
(cyl.- 7.6m dia., 8.1m ht.)	370 m^3
Propulsion System	
Propellant Type	LOX/LH_2
Engines	
Number	4
Type	RL10B-2
Mass (ea.)	191 kg
Max. Thrust (total)	392 kN (88.1 klb_f)
I_{sp} (460 sec)	4.51 kN-s/kg
LTV Propellant Mass	12,768 kg
Tank Mass	1,277 kg
Initial T/W_L	7.0 @ 100% throttle
Final T/W_L	11.2 @ 100% throttle
Mass Fraction	
Descent from LLO	0.628
Total Mass	34,287 kg

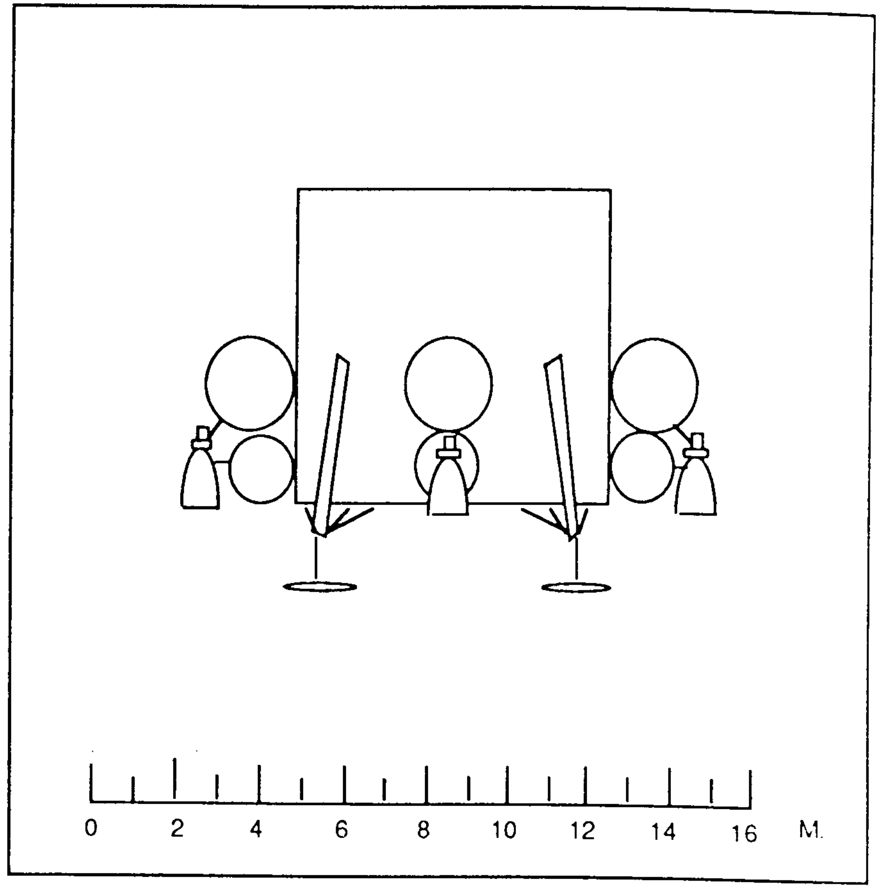


Figure 3.1.2.1-3b Lunar Descent Vehicle—Cargo (LDV-C)

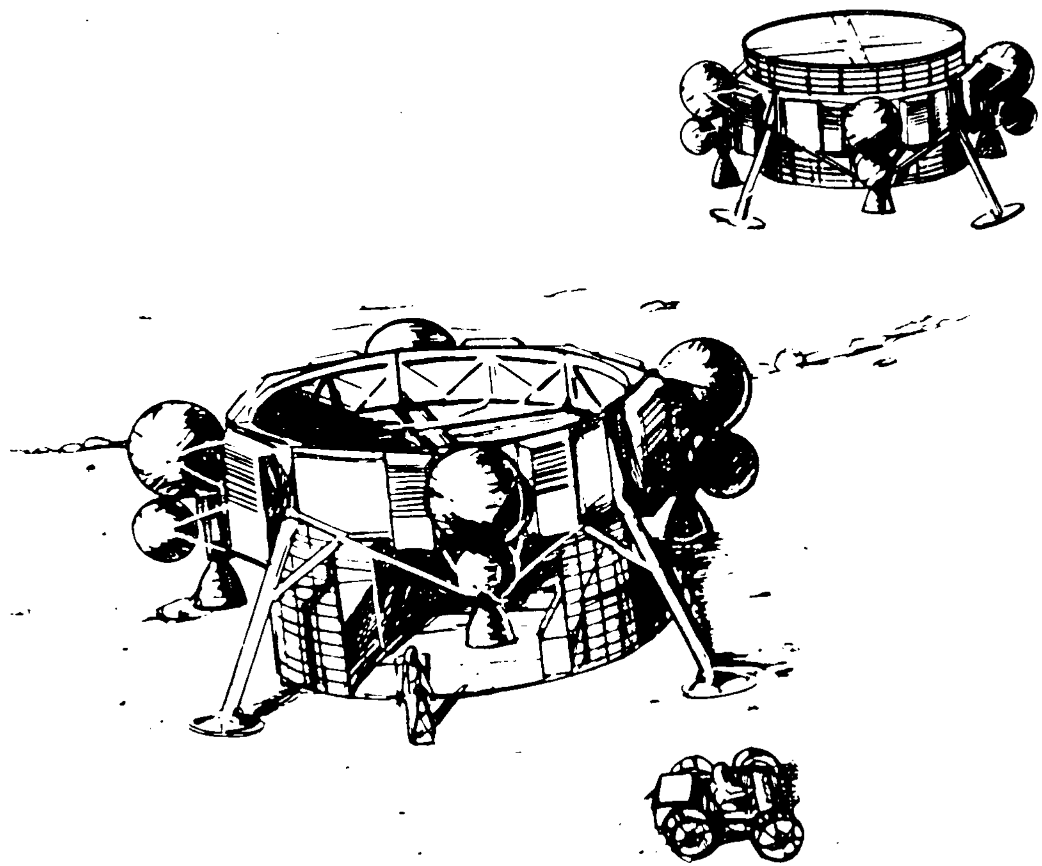


Fig. 3.1.2.1-4 Lunar Descent Vehicle-Piloted (LDV-P)

(14 day staytime, 55 day contingency, 4 crew)

Dry Mass (includes payload)	21,866 kg
Payload Mass	
(crew equipment & supplies)	8,044 kg
Payload Volume	
LAV	
(cone - 3.6m dia., 2.3m ht.)	8 m^3
HAB Mod	
(cyl.- 7.6m dia., 3m ht.)	136 m^3
Propulsion System	
Propellant Type	LOX/LH_2
Engines	
Number	4
Type	RL10B-2
Mass (ea.)	191 kg
Thrust (total)	392 kN (88.1 klb_f)
I_{sp} (460 sec)	4.51 kN-s/kg
Propellant Mass	12,974 kg
Tank Mass	1,297 kg
Initial T/W_L	6.9 @ 100% throttle
Final T/W_L	11.0 @ 100% throttle
Mass Fraction	
Descent from LLO	0.628
Total Mass	34,840 kg

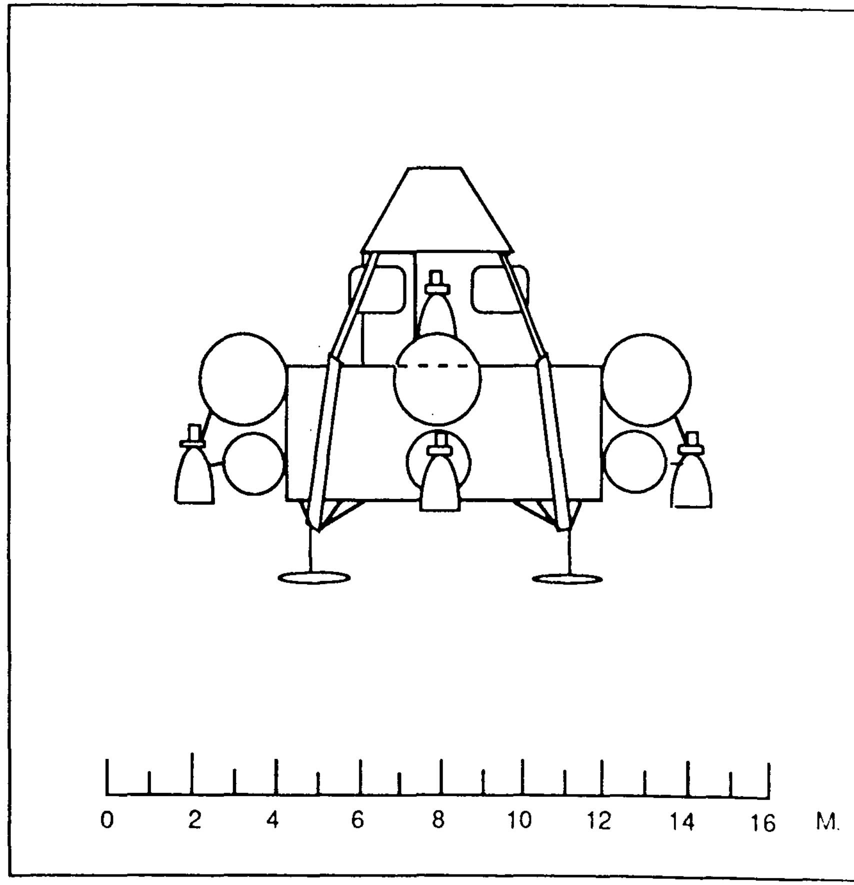


Fig. 3.1.2.1-5 Lunar Ascent Vehicle (LAV)

Dry Mass (includes payload)	3,408 kg
Payload Mass (crew & supplies)	430 kg
Payload Volume (cone - 3.6m dia., 2.3m ht.)	8 m^3
Propulsion System	
Propellant Type	LOX/LH_2
Engines	
Number	3
Type	RL10B-2'
Mass(ea.)	145 kg
Thrust (total)	294 kN (66.1 klb_f)
I_{sp} (460 sec)	4.51 kN-s/kg
Propellant Mass	2,022 kg
Tank Mass	202 kg
Initial T/W_L (1.10 gee at 20% throttle)	33.1 @ 100% throttle
Mass Fraction	
Ascent to LLO	0.628

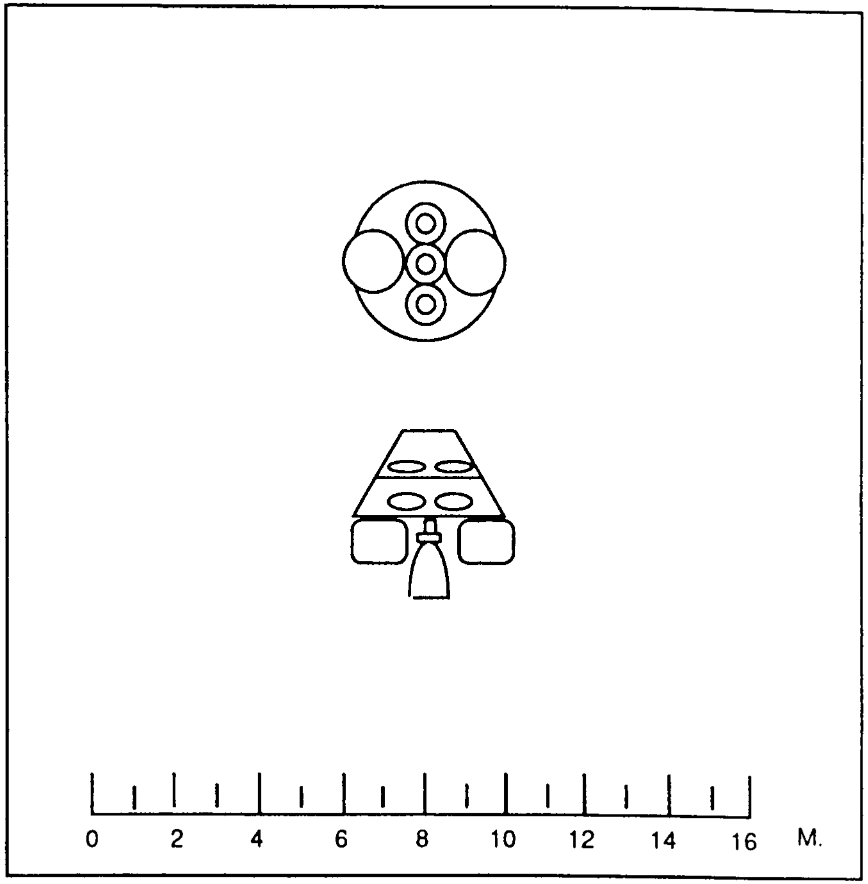


Table 3.1.2.2-1 ETO Manifest -- Case Study 3

Lunar Observatory and Exploration Component Launches to LEO

Assumptions:

1. All HLLV launches require the larger 12.8 m diameter by 38.1 m length, with a capability of 91 t to SS orbit.
2. The stack height includes the height of the payload vehicles plus 10% to allow for securing them inside the HLLV. This value appears in the launch column beneath the launch vehicle identification as (height of payload vehicle in m).
3. The mass shown for each payload vehicle includes a 5% allowance for the mass mass required to secure them inside the HLLV.
4. Payloads vehicles are dry (D), wet (W) or partially (*) filled with propellant. The mass of the propellant carried within the vehicle shown as (* mass in t). This value is included in the column "Mass"

Launch	Date	Item(s)	Mass (t)
1. HLLV-1 (33.6 m)	Mar '00	LTV-1 (* 19.0 t)	32.7
		LDV-C-1 (W)	36.0
		LDV-P-1 (D)	22.2
			90.9

Table 3.1.2.2-1 (cont'd) ETO Manifest -- Case Study 3

Launch	Date	Item	Mass (t)
2. HLLV-2	Apr '00	Propellant	62.5
		LTV-2 (D)	13.9
			76.4

Fill LTV-1 and LDV-P-1 with propellant, perform Mission (Car-1)

Launch	Date	Item	Mass (t)
3. HLLV-3	Jun '00	Propellant	80.9

Fill LTV-2 with propellant, perform Mission (Hum-1)

Launch	Date	Item	Mass (t)
4. HLLV-4	Feb '01	LTV-3 (* 19.0 t)	32.7
(33.6)		LDV-C-2 (W)	36.0
		LDV-P-2 (D)	22.2
			90.9
5. HLLV-5	Mar '01	Propellant	62.5
		LTV-4 (D)	13.9
			76.4

Fill LTV-3 and LDV-P-2 with propellant, perform Mission (Car-2)

Launch	Date	Item	Mass (t)
6. HLLV-6	Jun '01	Propellant	80.9

Fill LTV-4 with propellant, perform Mission (Hum-2)

The Lunar Observatory is now complete.

Table 3.1.2.2-1 (cont'd) ETO Manifest -- Case Study 3 (cont'd)

Launch	Date	Item	Mass (t)
7. HLLV-7 (23.7 m)	Mar '02	LDV-P-3 (W) LTV-5 (D)	36.6 13.9 50.5
8. HLLV-8	Apr '02	Propellant	80.9
Fill LTV-5 with propellant, perform Mission (Hum-3) to an alternate lunar site.			
9. HLLV-9 (23.7 m)	Mar '03	LDV-P-4 (W) LTV-6 (D)	36.6 13.9 50.5
10. HLLV-10	Apr '03	Propellant	80.9
Fill LTV-6 with propellant, perform Mission (Hum-4) to an alternate lunar site.			
11. HLLV-11 (23.7 m)	Mar '04	LDV-P-5 (W) LTV-7 (D)	36.6 13.9 50.5
12. HLLV-12	Apr '04	Propellant	80.9
Fill LTV-7 with propellant, perform Mission (Hum-5) a return to the Lunar Observatory.			

Table 3.1.2.2-1 (cont'd) ETO Manifest -- Case Study 3 (cont'd)

Launch	Date	Item	Mass (t)
13. HLLV-13 (23.7 m)	Mar '05	LDV-P-6 (W) LTV-8 (D)	36.6 13.9 50.5
14. HLLV-14	Apr '05	Propellant	80.9

Fill LTV-8 with propellant, perform Mission (Hum-6) to an alternate lunar site.

Launch	Date	Item	Mass (t)
15. HLLV-15 (23.7 m)	Mar '06	LDV-P-7 (W) LTV-9 (D)	36.6 13.9 50.5
16. HLLV-16	Apr '06	Propellant	80.9

Fill LTV-9 with propellant, perform Mission (Hum-7) to an alternate lunar site.

Launch	Date	Item	Mass (t)
17. HLLV-17 (23.7 m)	Mar '07	LDV-P-8 (W) LTV-10 (D)	36.6 13.9 50.5
18. HLLV-18	Apr '07	Propellant	80.9

Fill LTV-10 with propellant, perform Mission (Hum-8) a return to the Lunar Observatory.

etc.

Table 3.1.2.2-2 Case Study 3, On-Orbit Activities

Note: (W) = Payload Vehicle Launched Wet
(D) = Payload Vehicle Launched Dry
(*) = Payload Vehicle Launched Partially Filled with Propellant

HLLV # 1 LTV-1(*), LDV-C-1(W) and LDV-P-1(D) to LEO

HLLV # 2 Propellant

Fill LTV-1 and LDV-P-1 with propellant, perform Mission (Car-1).

The first 4 missions establish the Lunar Observatory.

Mission (Car-1)

Step #	**Activity Description**
1	LTV-1 with LDV-C-1 from LEO to LLO
2	LDV-C-1 separation from LTV-1
3	LDV-C-1 descent to moon
4	LTV-1 returns to LEO

HLLV # 3 Propellant

5 Fill LTV-1 with propellant, perform Mission (Hum-1).

Mission (Hum-1)

6	LTV-1 docks with LDV-P-1
7	LTV-1 with LDV-P-1 from LEO to LLO
8	LDV-P-1 separation from LTV-1 (LTV-1 remains in LLO)
9	LDV-P-1 descent to moon
10	LAV-1 ascent from moon to LLO
11	LAV-1 docks with LTV-1
12	LTV-1 with LAV-1 returns to LEO

HLLV # 4 LDV-C-2 (W) and LDV-P-2 (W) to LEO

HLLV # 5 Propellant

13 Fill LTV-1 with propellant, perform Mission (Car-2).

Mission (Car-2)

14	LTV-1 docks with LDV-C-2
15	LTV-1 with LDV-C-2 from LEO to LLO

Table 3.1.2.2-2 (cont.) Case Study 3, On-Orbit Activities

16 LDV-C-2 separation from LTV-1
17 LDV-C-2 descent to moon
18 LTV-1 returns to LEO
HLLV # 6 Propellant

19 LTV-1 fill with propellant, perform Mission (Hum-2).

Mission (Hum-2)

20 LTV-1 docks with LDV-P-2
21 LTV-1 with LDV-P-2 from LEO to LLO
22 LDV-P-2 separation from LTV-1 (LTV-1 remains in LLO)
23 LDV-P-2 descent to moon
24 LAV-2 ascent from moon to LLO
25 LAV-2 docks with LTV-1
26 LTV-1 with LAV-2 returns to LEO

The Lunar Observatory has been completed.

HLLV # 7 LDV-P-3(W) to LEO

HLLV # 8 Propellant

27 LTV-1 fill with propellant, perform Mission (Hum-3).

The following mission is used to explore an alternate lunar site:

Mission (Hum-3)

28 LTV-1 docks with LDV-P-3
29 LTV-1 with LDV-P-3 from LEO to LLO
30 LDV-P-3 separation from LTV-1 (LTV-1 remains in LLO)
31 LDV-P-3 descent to moon
32 LAV-3 ascent from moon to LLO
33 LAV-3 docks with LTV-1
34 LTV-1 with LAV-3 returns to LEO

HLLV # 9 LDV-P-4(W) to LEO

HLLV # 10 Propellant

35 LTV-1 fill with propellant, perform Mission (Hum-4)

The following mission is used to explore an alternate lunar site:

Mission (Hum-4)

Table 3.1.2.2-2 (cont.) Case Study 3, On-Orbit Activities

36	LTV-1 docks with LDV-P-4
37	LTV-1 with LDV-P-4 from LEO to LLO
38	LDV-P-4 separation from LTV-1 (LTV-1 remains in LLO)
39	LDV-P-4 descent to moon
40	LAV-4 ascent from moon to LLO
41	LAV-4 docks with LTV-1
42	LTV-1 with LAV-4 returns to LEO
HLLV # 11	LDV-P-5(W) to LEO
HLLV # 12	Propellant
43	LTV-1 fill with propellant, perform Mission (Hum-5).

The following mission is a return to the observatory site:

Mission (Hum-5)

44	LTV-1 docks with LDV-P-5
45	LTV-1 with LDV-P-5 from LEO to LLO
46	LDV-P-5 separation from LTV-1 (LTV-1 remains in LLO)
47	LDV-P-5 descent to moon
48	LAV-5 ascent from moon to LLO
49	LAV-5 docks with LTV-1
50	LTV-1 with LAV-5 returns to LEO
HLLV # 13	LDV-P-6(W) to LEO
HLLV # 14	Propellant
51	LTV-1 fill with propellant, perform Mission (Hum-6).

The following mission is used to explore an alternate lunar site:

Mission (Hum-6)

52	LTV-1 docks with LDV-P-6
53	LTV-1 with LDV-P-6 from LEO to LLO
54	LDV-P-6 separation from LTV-1 (LTV-1 remains in LLO)
55	LDV-P-6 descent to moon
56	LAV-6 ascent from moon to LLO
57	LAV-6 docks with LTV-1
58	LTV-1 with LAV-6 returns to LEO
HLLV # 15	LDV-P-7(W) to LEO
HLLV # 16	Propellant

Table 3.1.2.2-2 (cont.) Case Study 3, On-Orbit Activities

59 LTV-1 fill with propellant, perform Mission (Hum-7)

The following mission is used to explore an alternate lunar site:

Mission (Hum-7)

60 LTV-1 docks with LDV-P-7
61 LTV-1 with LDV-P-7 from LEO to LLO
62 LDV-P-7 separation from LTV-1 (LTV-1 remains in LLO)
63 LDV-P-7 descent to moon
64 LAV-7 ascent from moon to LLO
65 LAV-7 docks with LTV-1
66 LTV-1 with LAV-7 returns to LEO

HLLV # 17 LDV-P-8(W) to LEO

HLLV # 18 Propellant

67 LTV-1 fill with propellant, perform Mission (Hum-8).

The following mission is a return to the observatory site:

Mission (Hum-8)

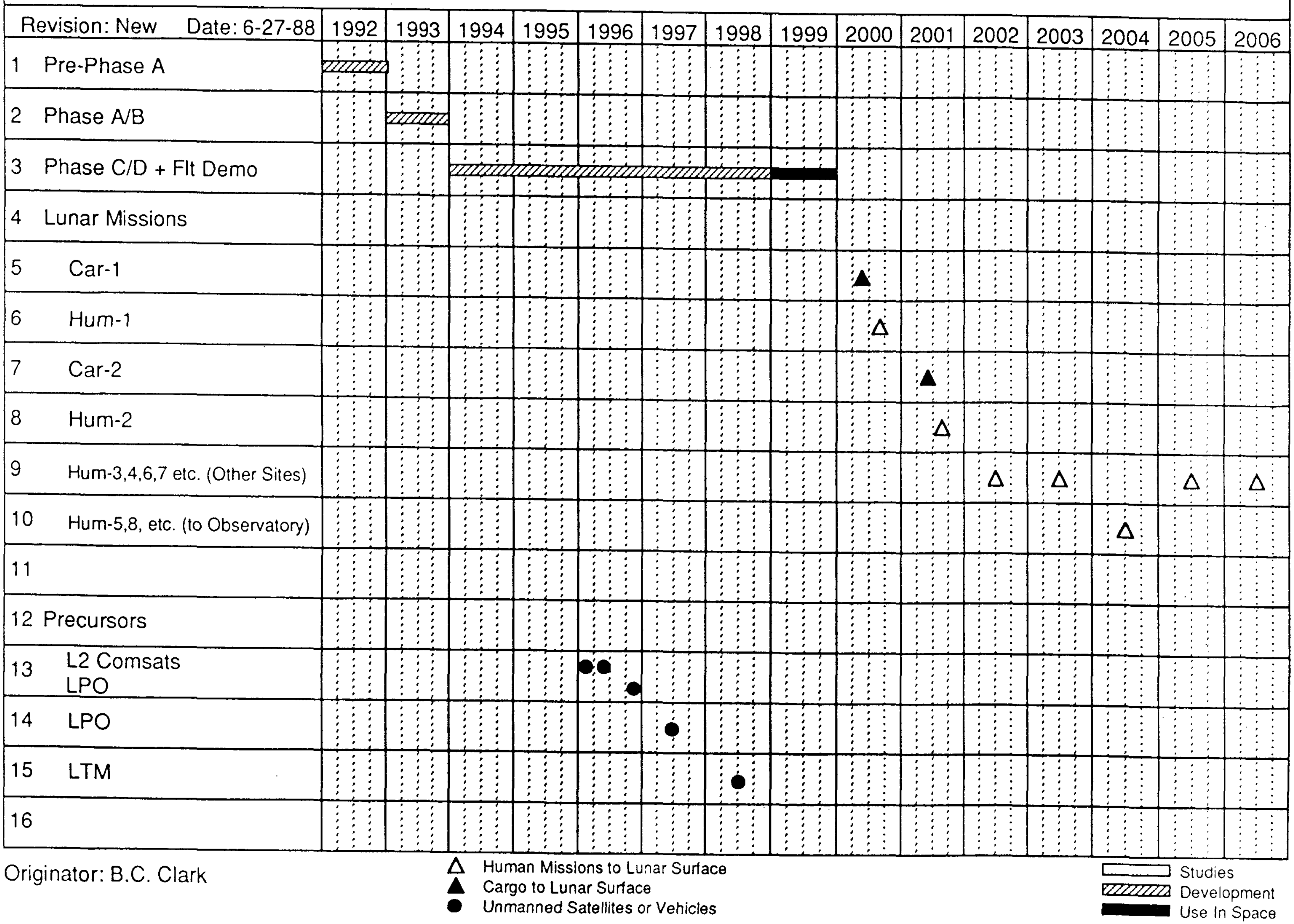
68 LTV-1 docks with LDV-P-8
69 LTV-1 with LDV-P-8 from LEO to LLO
70 LDV-P-8 separation from LTV-1 (LTV-1 remains in LLO)
71 LDV-P-8 descent to moon
72 LAV-8 ascent from moon to LLO
73 LAV-8 docks with LTV-1
74 LTV-1 with LAV-8 returns to LEO

Note # 1: Each LDV-P-# includes a LAV-#.

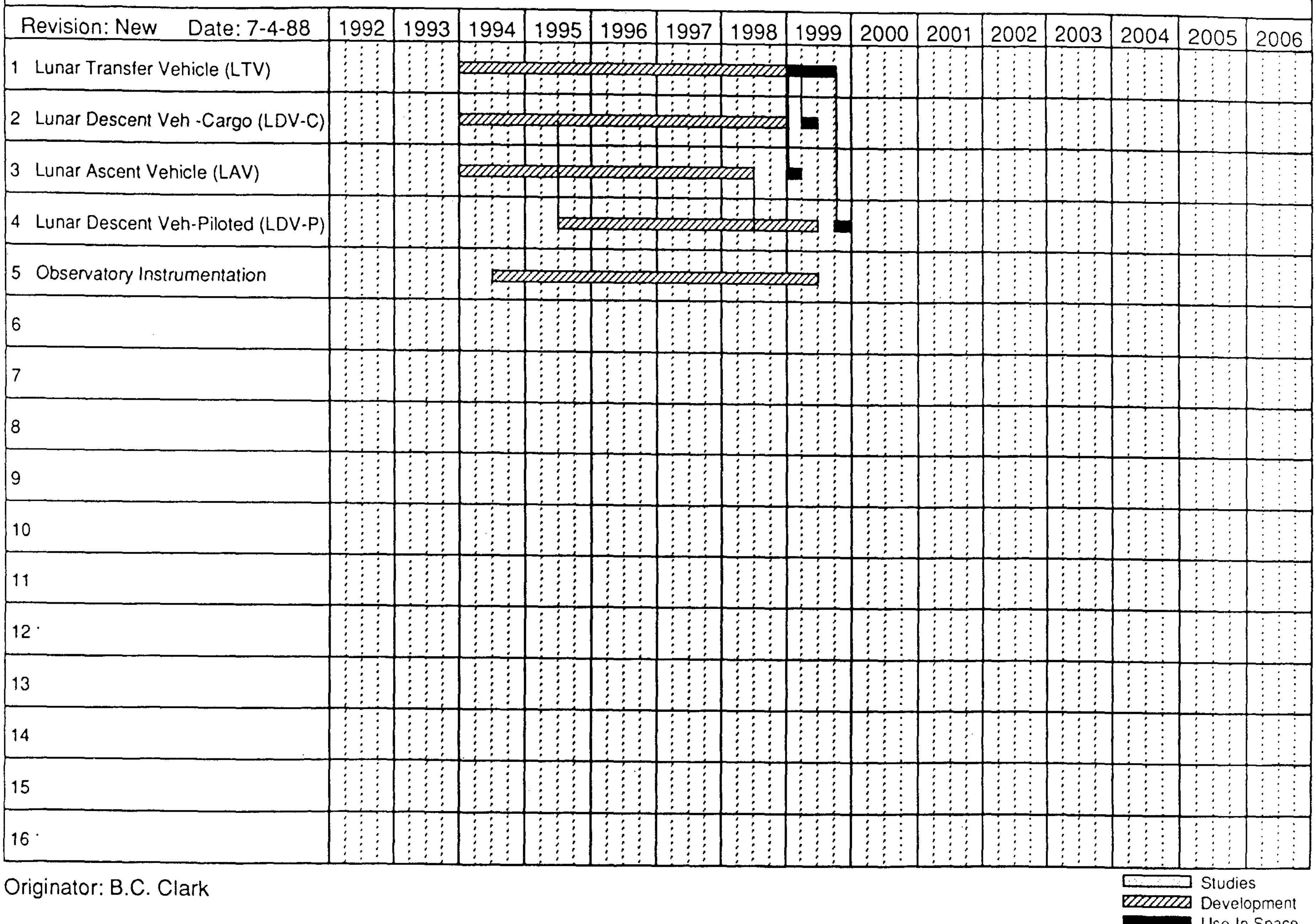
Note # 2: All payload vehicles are launched on a HLLV (large diameter 12.8 M x 38.1 M length).

Note # 3: Propellant launches assume a capability of 82 t to SS orbit.

Title: **Figure 3.1.3-1. Case Study 3 -- Transportation Program Development Schedule**



Title: **Figure 3.1.3-1 (cont.). Case Study 3 -- Transportation Program Development Schedule**



Originator: B.C. Clark

Table 3.3-1 Reference Mission vs Direct Alternative

	TIC-1R (Reference)	TIC-1A (Direct Alternative)
LTV for LDV-P	94.1 t	---
LTV for LDV-C	82.9 t	---
LDV-P	34.8 t	115.1 t
LDV-C	34.3 t	96.4 t
Total Mass in LEO	246.1 t	211.5 t

C-3

Table 3.3-2
ΔV Requirements for Lunar Scenario - Direct Alternative

	ΔV (m/s)
TLI	3150
Mid-course correction	50
Descent to Lunar surface	2509
Total, Lunar landing	5709
TEI w/Lunar ascent	2509
Mid-course correction	50
Direct Earth entry	---
Total, inbound	2559
Round trip total	8268

Table 3.3.-3 Launch Summary for Various Options

Year	'00	'01	'02	'03	'04			
# of Launches	3	3	2	2	2			
IMLEO (t)								
Baseline	246	246	129	129	129			
Direct entry	212	212	115	115	115	Reuseable LTV*	242	228
116	116	116						

*Returns LTV used for cargo, as well as LTV/LAV

Case Study 4

Lunar Outpost-to-Mars Outpost

4.0 Introduction

In this scenario a lunar base is first established, and from that base Mars missions are launched. As was mentioned in the introduction, only certain portions of this case study were examined in any detail because of changing requirements for this Moon/Mars Evolution pathway. In particular, several concepts for vehicles were proposed, as well as a Nulclear Electric Propulsion system which would serve as an alternative to a cargo vehicle's TMIS.

4.1 Transportation Elements and Systems Description

The transportation concepts for Case Study 4 consist of both the vehicles necessary to place cargo and crew on the lunar surface, as well as the vehicles to transport the astronauts and cargo to Mars. The Lunar Cargo Lander (LCL) is shown in Figures 4.1-1a and -1b (expanded view) along with it's mass breakdown. The lunar crew vehicle, or Lunar Transfer Vehicle-Piloted (LTV-P, Figure 4.1-2), is similar to the LTV in Case Study 3. A LAV-like module nestles amid the circular propulsion tanks and serves for crew transport between Earth and the Moon. To reach the lunar surface, astronauts will use the Lunar Piloted Lander (LPL) shown in Figure 4.1-3.

Once a permanent base has been established on the moon, it is assumed that it could somehow serve as a stepping stone to Mars. The Interplanetary Crew System with it's TMIS is depicted in Figure 4.1-4a and -4b (expanded view). Two vehicles were considered for crew transport to the suface of Mars. The first, the Mars Logistics Lander (MLL) (Figure 4.1-5) uses aerocapture to assist in the landing, and once on the surface, it makes no further excursions. The second vehicle, the Mars-Phobos Excursion Vehicle (MPEM) (Figure 4.1-6) has the capability to visit Phobosas well as landing on Mars (aerobrake not shown). The return vehicle, again very similar to its analog in Case Study 2, is depicted with its mass breakdown in Figure 4.1-7.

As was mentioned earlier, Nuclear Electric Propulsion has been considered for cargo transport to Mars. The assumptions and guidelines for this system are given in Table 4.1-1, and it's mass allocation is summarized in Table 4.1-2. Figures 4.1-8a, -8b, and -8c depict the NEP, with the first figure providing the mass data.

Figure 4.1-1a Lunar Cargo Lander (LCL)

Dry Mass	
(includes LLOX payload)	93,967 kg
Payload Mass	
(LLOX, ascent only)	80,000 kg
(user payload, descent only)	40,000 kg
Propulsion System	
Propellant Type	LOX/LH_2
Engines	
Number	4
Type	Advanced
Mass (ea.)	600 kg
Thrust (total)	1,779 kN
I_{sp} (470 sec)	4.62 kN-s/kg
Propellant Mass	106,782 kg
Tank Mass	5,339 kg
Initial T/W	5.43
Mass Fraction	
Ascent to LLO	0.6276
Descent to LS	0.6276
Total Mass	200,749 kg

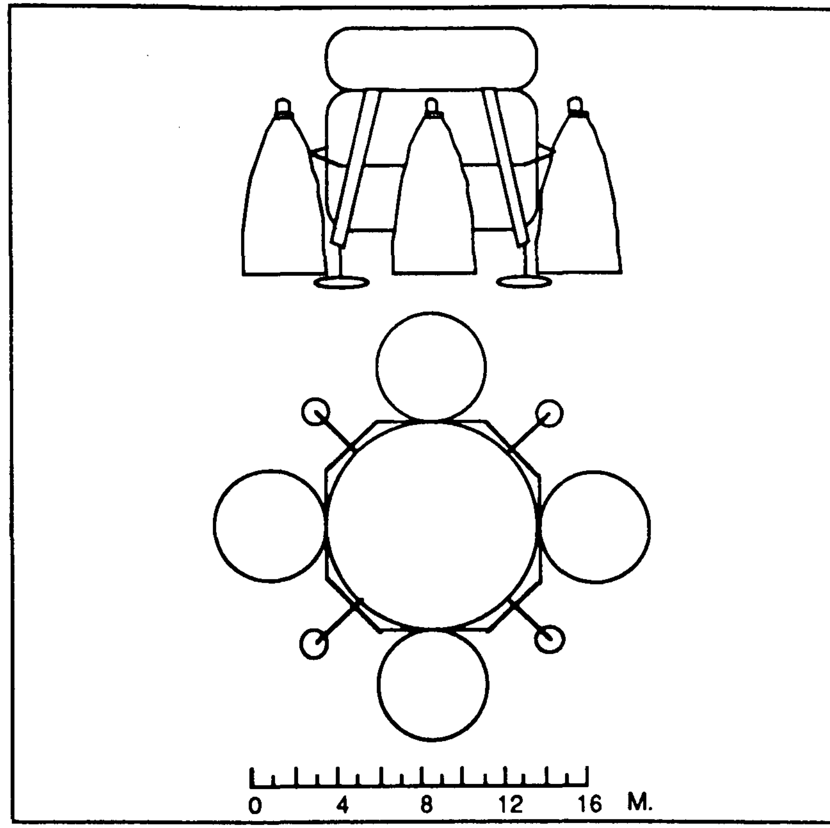


Figure 4.1-1b Lunar Cargo Lander (LCL)

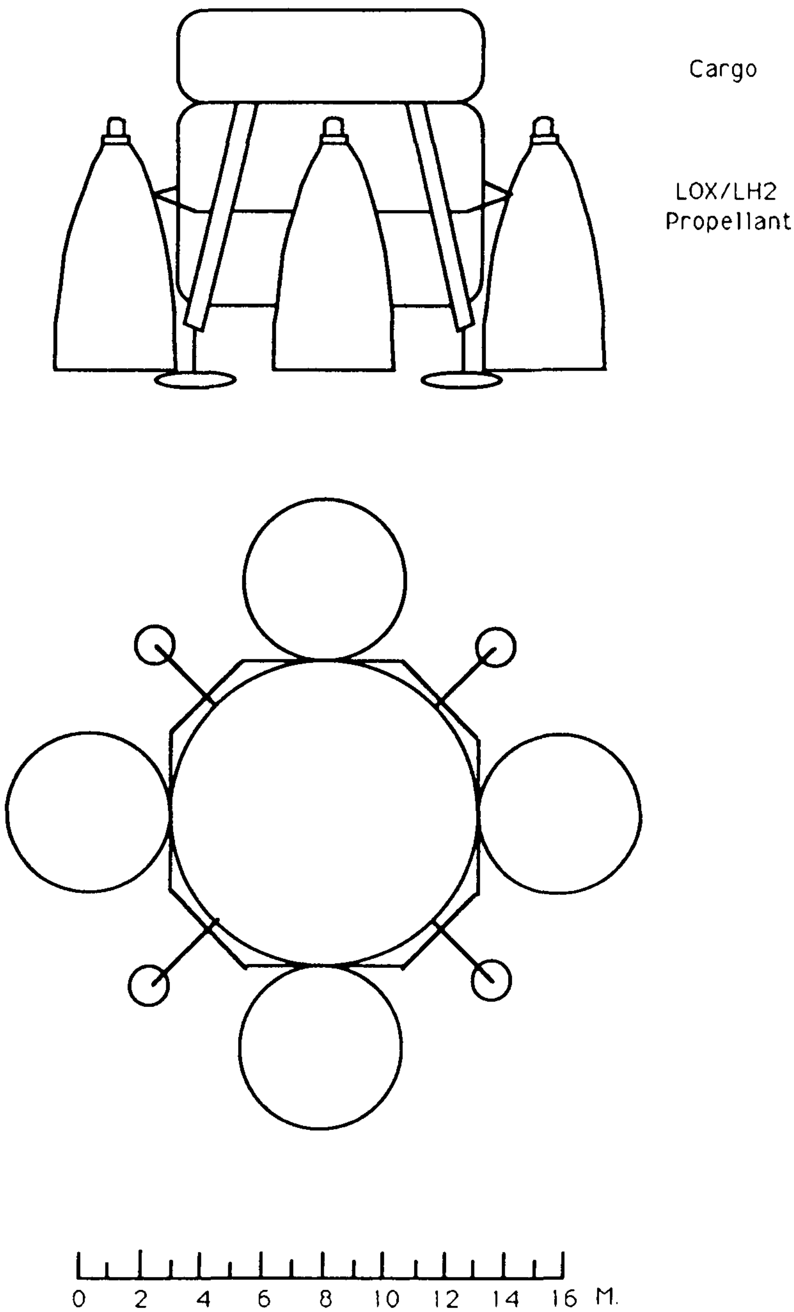


Figure 4.1-2 Lunar Transfer Vehicle-Piloted (LTV-P)

Dry Mass	
(includes payload)	9972 kg
Payload Mass	
(HAB Module)	2729 kg
Payload Volume	
HAB Module	
(cone - 3.6m dia., 2.3m ht.)	8 m^3
Propulsion System	
Propellant Type	LOX/LH_2
Engines	
Number	4
Type	RL10B-2
Mass (ea.)	191 kg
Thrust (total)	392 kN
I_{sp} (460 sec)	4.51 kN-s/kg
Propellant Mass	20,850 kg
Tank Mass	2,085 kg
Initial T/W_E	1.3
Mass Fraction	
Total	0.324
Total Mass	30,822 kg

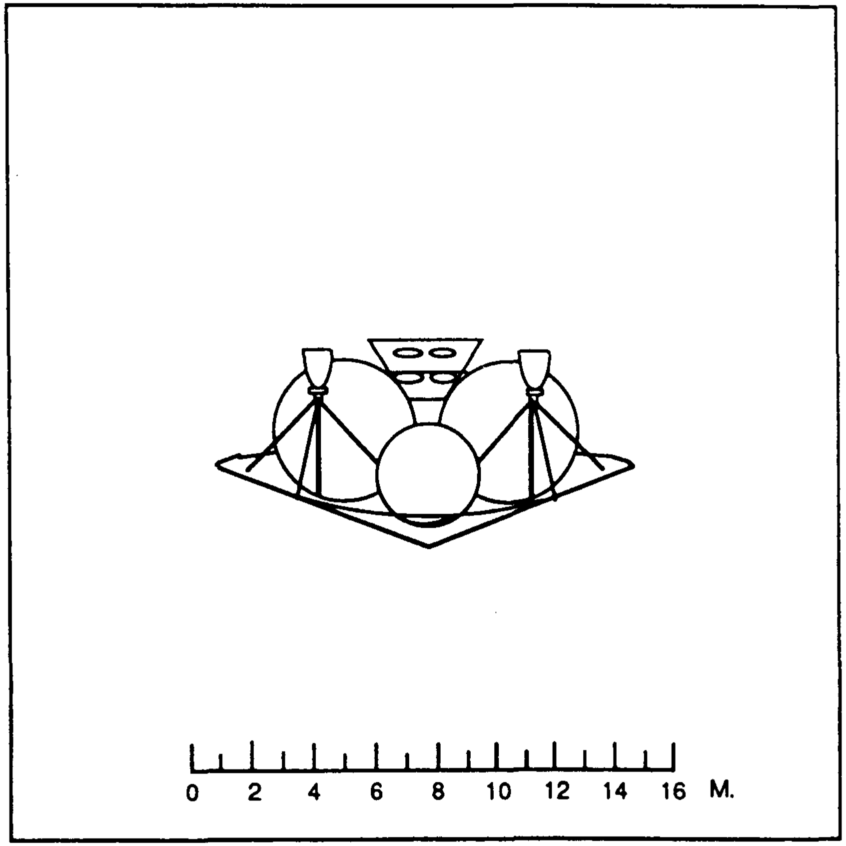


Figure 4.1-3 Lunar Piloted Lander (LPL)

Dry Mass	
(includes payload)	3,489 kg
Payload Mass	
(Crew & Supplies)	324 kg
Propulsion System	
Propellant Type	LOX/LH_2
Engines	
Number	3
Type	RL10-
derivative	
Mass (ea.)	145 kg
Thrust (total)	210 kN
I_{sp} (470 sec)	4.62 kN-s/kg
Propellant Mass	5,369 kg
Tank Mass	537 kg
Initial T/W	14.5 @ 100%
Throttle	
(0.48 gee at 20% throttle)	
Mass Fraction	
Ascent and Descent from LLO	0.394
Total Mass	8,858 kg

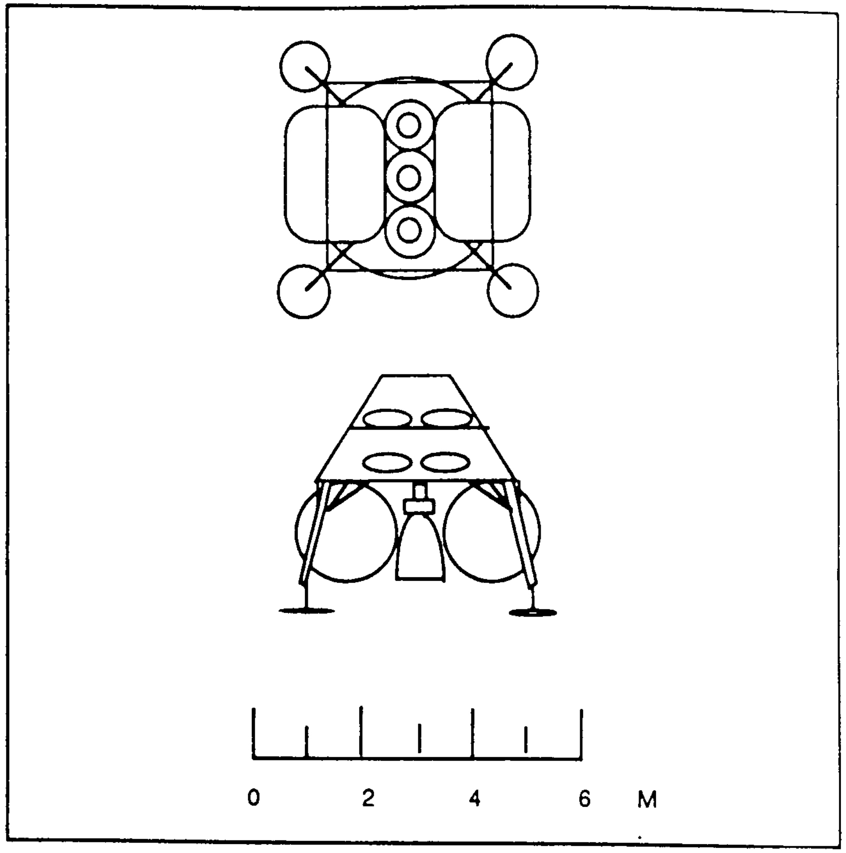


Figure 4.1-4a IP Crew System with TMIS (IPCS w/ TMIS)

Dry Mass	
(includes payload)	216,390 kg
Payload Mass	
(MTV)	186,490 kg
Propulsion System	
Propellant Type	LOX/LH_2
Engines	
Number	1
Type	SSME-
derivative	
Mass (ea.)	3,175 kg
Thrust (total)	2,415 kN
I_{sp} (480 sec)	4.71 kN-s/kg
Propellant Mass	598,000 kg
Initial T/W	0.30
Mass Fraction	
Total	0.27
Total Mass	814,765 kg

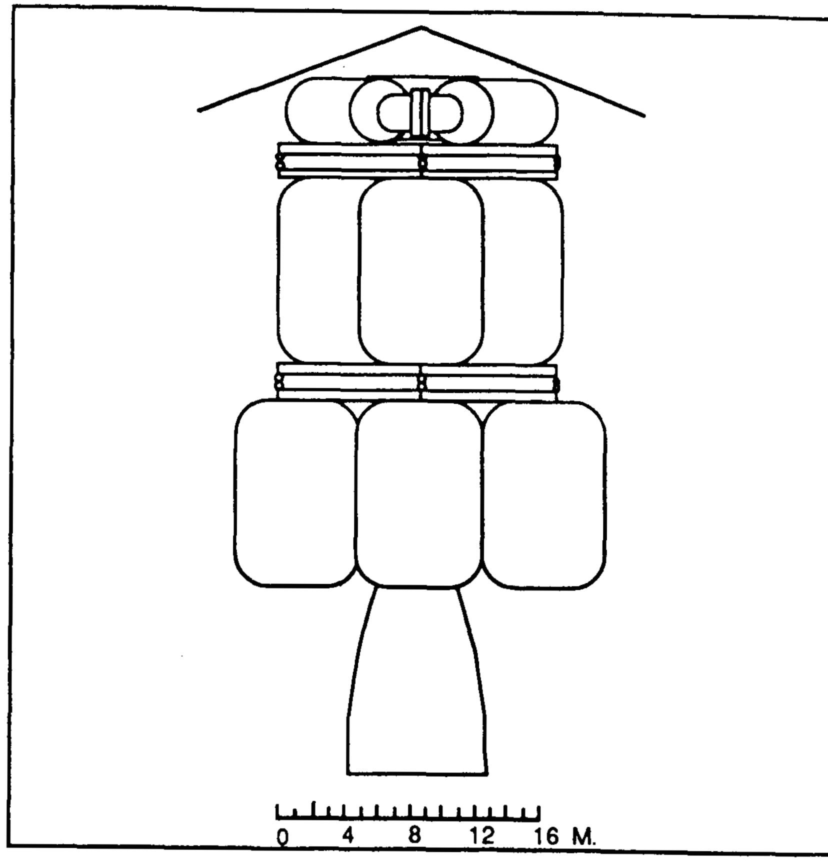


Figure 4.1-4b IP Crew System with TMIS

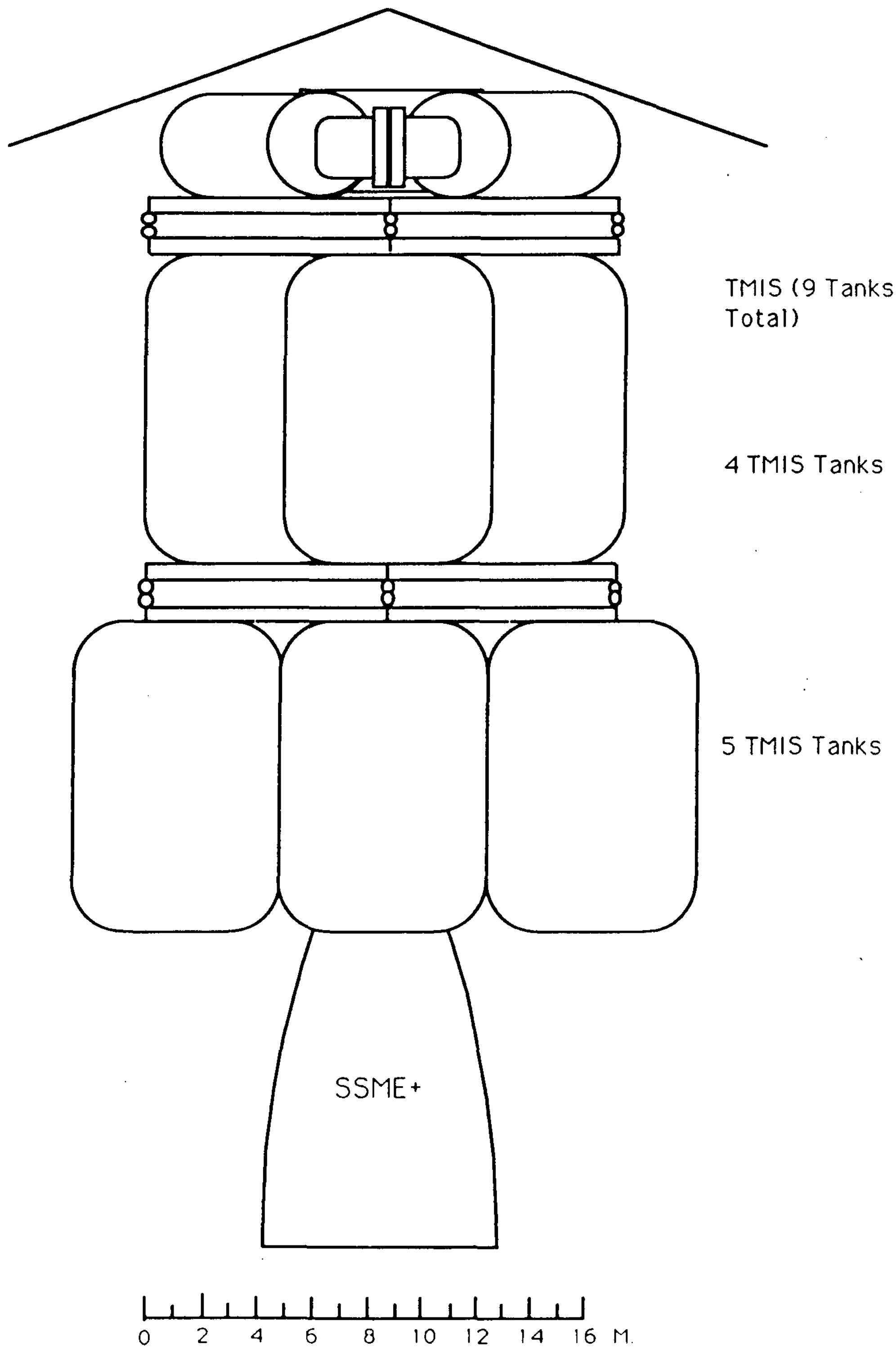


Figure 4.1-5 Mars Logistics Lander (MLL)

Dry Mass	
(includes payload)	60,974 kg
Payload Mass	
(Crew & equipment)	33,794 kg
Payload Volume	
HAB Module	
(cyl. - 9.4m dia., 6.1m ht.)	427 m^3
Propulsion System	
Propellant Type	MMH/N_2O_4
Engines	
Number	6
Type	Delta
Mass (ea.)	100 kg
Thrust (total)	267 kN
I_{sp} (320 sec)	3.14 kN-s/kg
Propellant Mass	4,850 kg
Tank Mass	485 kg
Initial T/W	1.10
Mass Fraction	
Terminal descent	
w/ parachute	0.93
Total Mass	65,824 kg

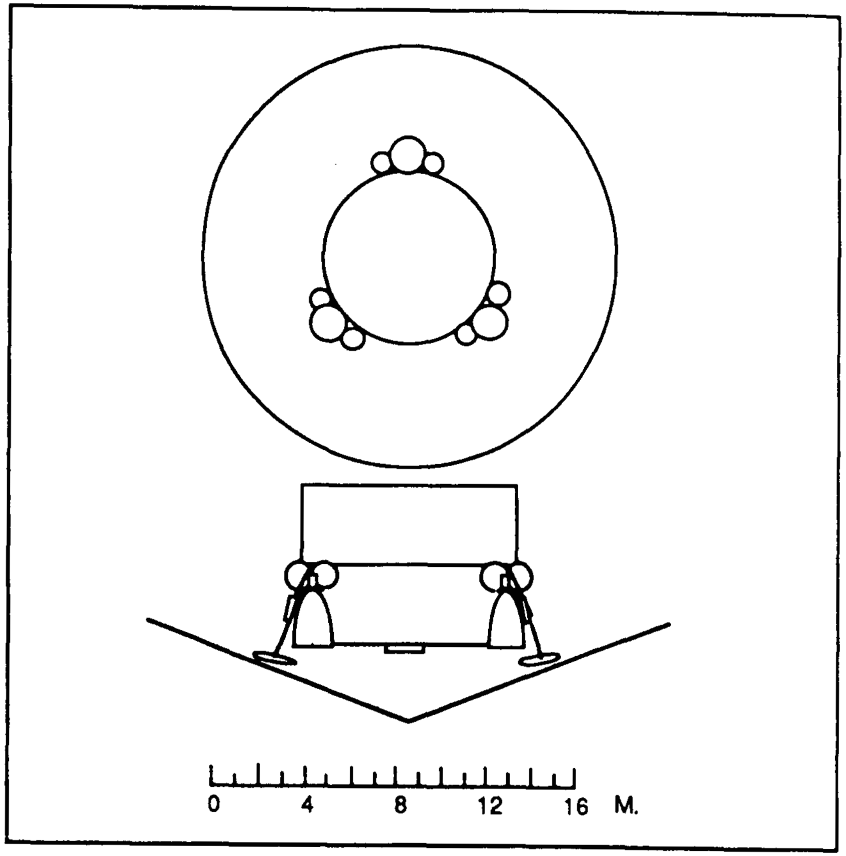


Figure 4.1-6 Mars-Phobos Excursion Module (MPEM)

Dry Mass	
(includes payload)	11,291kg
Payload Mass	
(Crew &Supplies)	316 kg
Propulsion System	
Propellant Type	MMH/N_2O_4
Engines	
Number	6
Type	Advanced
Mass (ea.)	100 kg
Thrust (total)	267 kN
I_{sp} (470 sec)	4.62 kN-s/kg
Propellant Mass	34,087 kg
Tank Mass	3,409 kg
Initial T/W	1.60
Mass Fraction	
De-orbit from Phobos circ.	0.8325
Terminal Descent	
w/ parachute	0.9533
Ascent to Phobos circ.	0.1856
Total Mass	45,378 kg

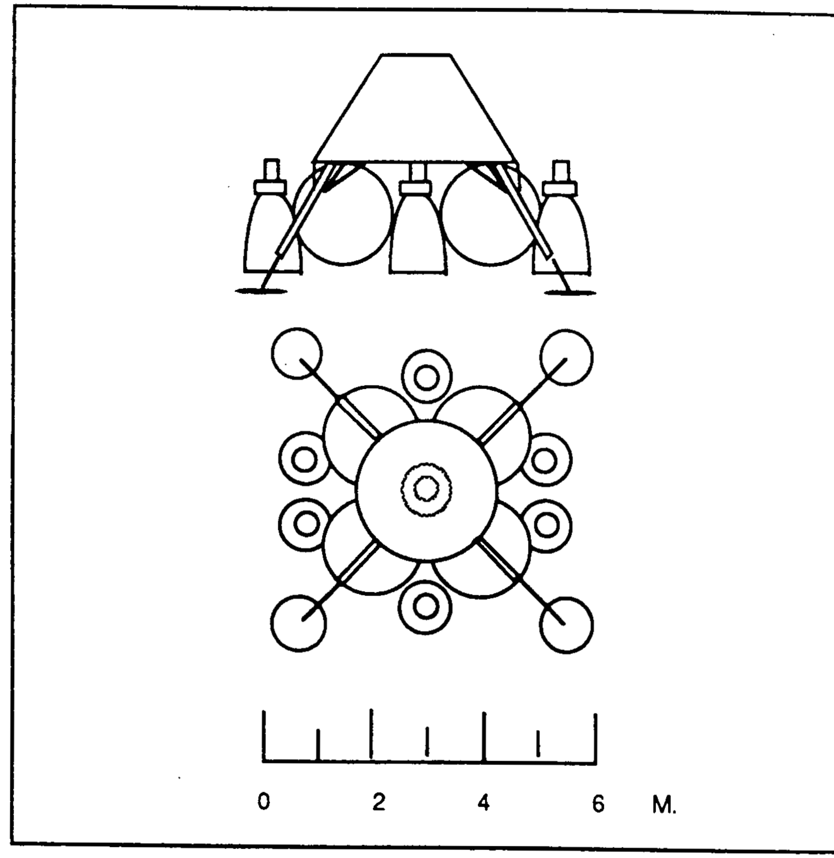


Figure 4.1-7 Trans-Earth Injection System (TEIS)

(sprint class mission)

Dry Mass	
(includes payload)	116,086 kg
Payload Mass	
(ETV)	86,340 kg
Propulsion System	
Propellant Type	LOX/LH_2
Engines	
Number	6
Type	RL-10B-2
Mass (ea.)	191 kg
Thrust (total)	588 kN
I_{sp} (480 sec)	4.71 kN-s/kg
Propellant Mass	210,000 kg
Initial T/W	0.49
Mass Fraction	
Total	0.356
Total Mass	326,086 kg

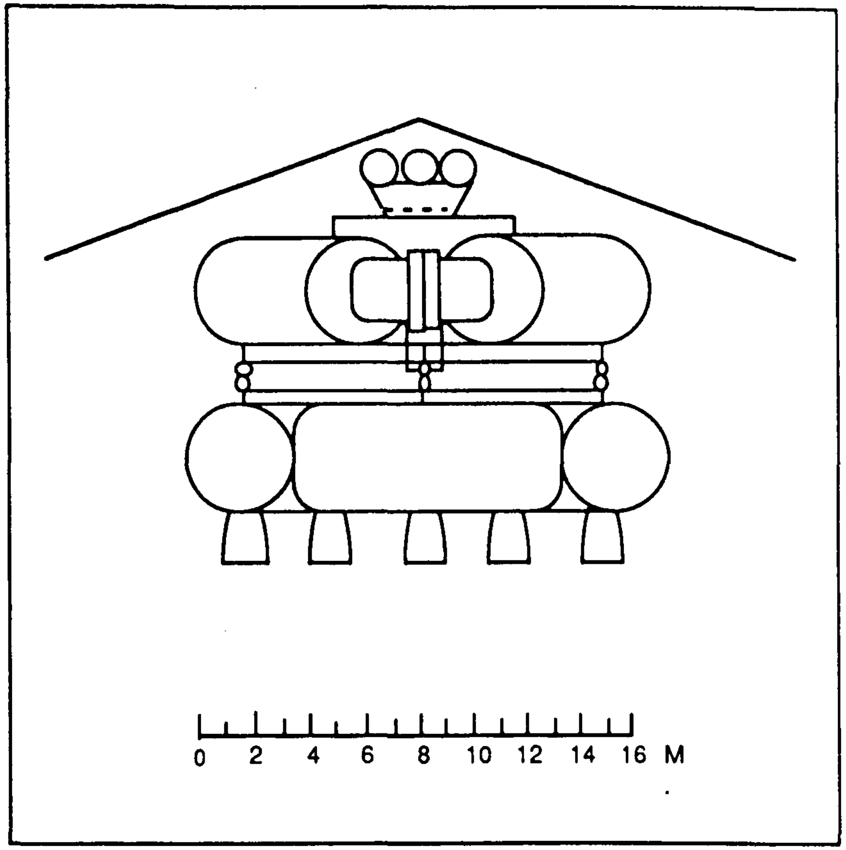


Table 4.1-1 NEP Assumptions and Guidelines

(1) alpha: 15 kg/kW_e (total system)

(2) Rx output: 5 MW_e

(3) System efficiency: 63%
    Power Control and Distribution Assembly (PCDA): 90%
    Ion thruster: 70%*

(4) Isp: 6000 s

(5) Inert mass assumptions:
    a. Tankage and propellant reserves: 10% of nominal propellant mass
    b. Payload adaptor structure: 5% of max payload (930 t max.x0.05=46.5 t)

(6) Total dry mass: 142.5 t

(7) Thruster efficiency
    see item 3 above

* per memo from Van Landingham. Higher efficiencies are normally quoted for smaller ion engines.

Table 4.1-2 Electric Vehicle -- Mass Allocation Summary*

Subsystem	Mass (tonnes)	
Nuclear Subsystem		42.8
Reactor	6.1	
4-Pi Radiation Shield	36.7	
Power Generation		5.0
Turbines	2.8	
Alternators	2.2	
Thermal Subsystem		3.6
Bubble Membrane Radiator	1.7	
Auxiliary Cooling	1.0	
Coolant (potassium)	0.9	
Power Conditioning and Distribution		5.2
Ion Engines		3.9
Structure		2.0
Contingency (20%)		12.5
Total		75.0

*Source: Coomes et al. (1987), AIAA-87-1038, 19th Int'l Electric Propulsion Conference; gross mass allocations made by SAIC (A. Friedlander, private c
engineering estimates by Martin Marietta

Figure 4.1-8a Electric Cargo Vehicle

Dry Mass (includes Max. Payload)	1,072,000 kg
Payload Mass (Max.)	910,000 kg
Payload Volume (as shown)	804m^3
Propulsion System	
Propellant Type	Argon
Engines	
Number	7 (2 spare)
Type	ion
Thrust (total)	21 N
I_{sp} (6000 sec)	58.9 kN-s/kg
Propellant Mass(max.)	212,000 kg
Tank Mass	21,200 kg
Total Mass	1,284,000 kg

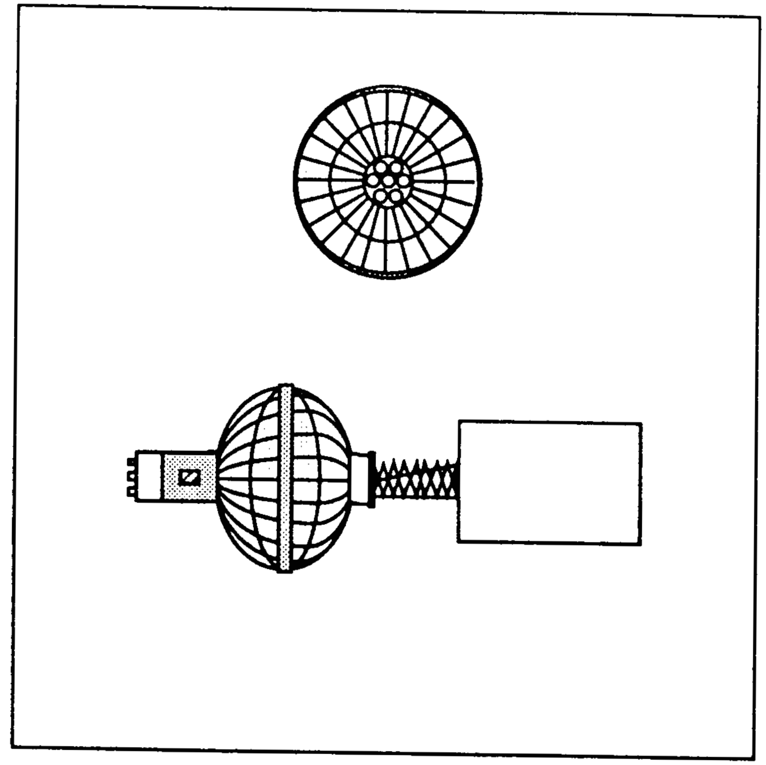
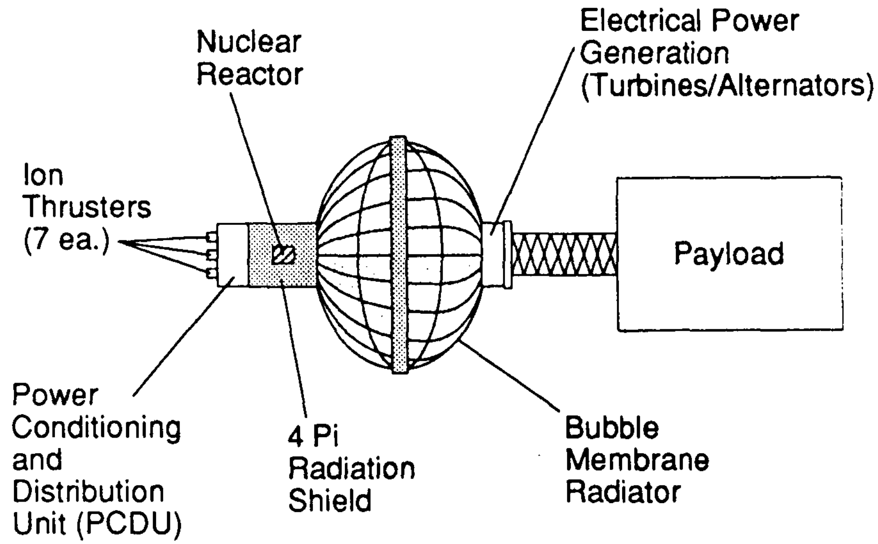
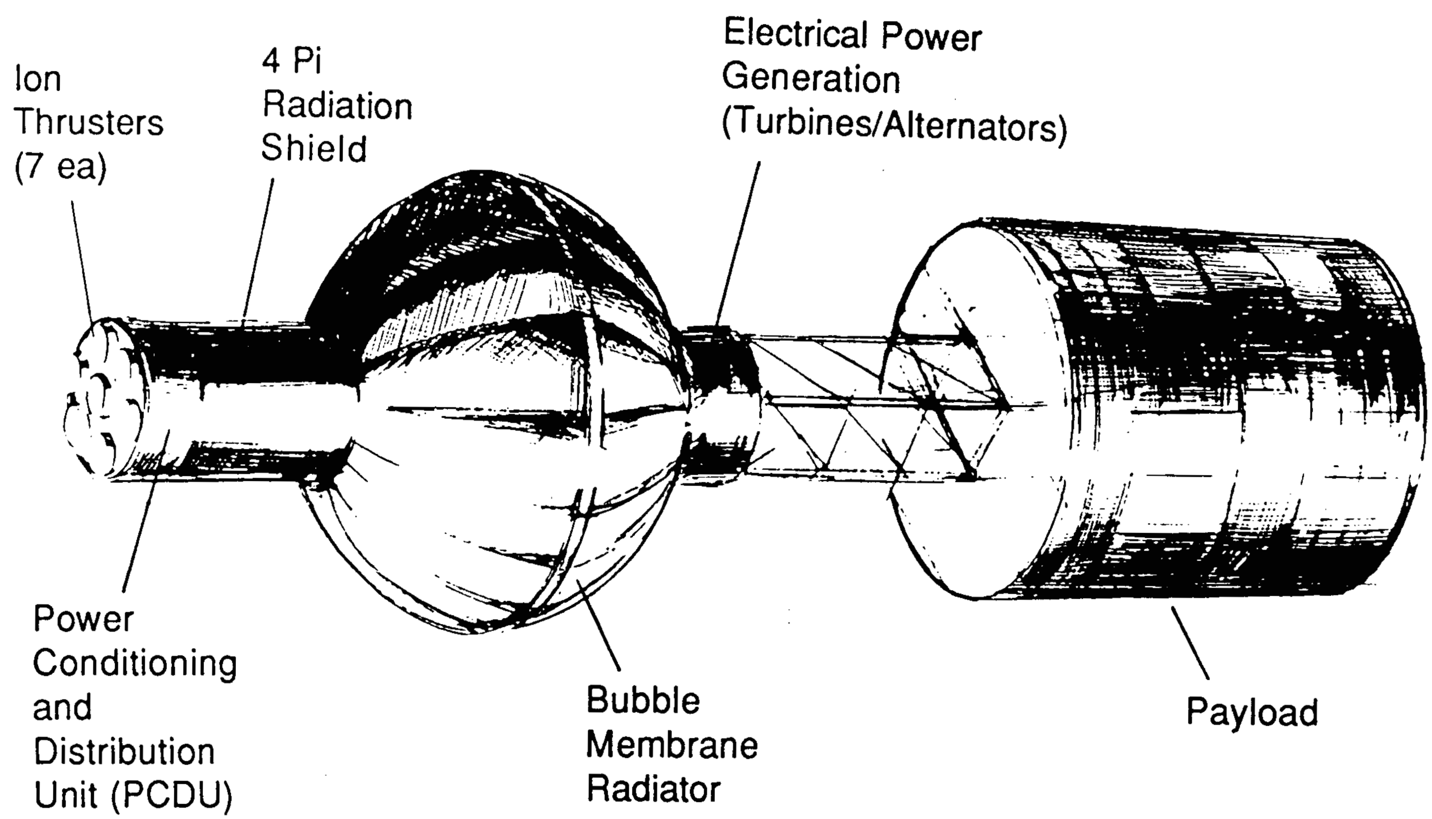


Figure 4.1-8b Nuclear Electric Propulsion



Thornton
6-30-88

Figure 4.1-8c Nuclear Electric Cargo Vehicle



Parametrics and Special Studies

5.1 Spaceship Configurations

Throughout the course of this study, several differrent configurations for spaceships have been considered. All that were studied use the Space Station cylindrical module as a basic component, to which is added tunnels, nodes, and squat-cylinder living habitats, as necessary. Certain configurations also made provisions for additional modules, should they be needed in the future.

For zero-gravity configurations, we considered from two to six space station modules. No less than two habitat modules were considered for reasons of safety; i.e., one module always serves as a safe haven in case of problems with the other module. Those with two space station modules are depicted in Figure 5.1-1, and those with three are shown in Figure 5.1-2. Several of the zero-gravity configurations with four space station modules (Figure 5.1-3) and those with five and six (Figure 5.1-4) are direct extensions of configurations with two of three modules. This compatability in design makes possible the option of expansion if needed.

Spaceships with artificial gravity are depicted in Figure 5.1-5 and 5.1-6. Those in Figure 5.1-5 have modules that are connected by tunnels, and all have a central habitat. The latter are tethered systems which rotate about the center of mass to produce gravity and EVA is necessary for travel between modules. The additional space that accompanies missions with aerobrakes could be used to transport pressurized rovers. Figures 5.1-7 and 5.1-8 show two ways that the rovers could be packed about the central habitat.

Finally, Table 5.1-1 gives the pressurized volume and atmospheric mass associated with central habitats of varying diameter.

Figure 5.1-1 Zero Gravity Configurations: Two Space Station Modules

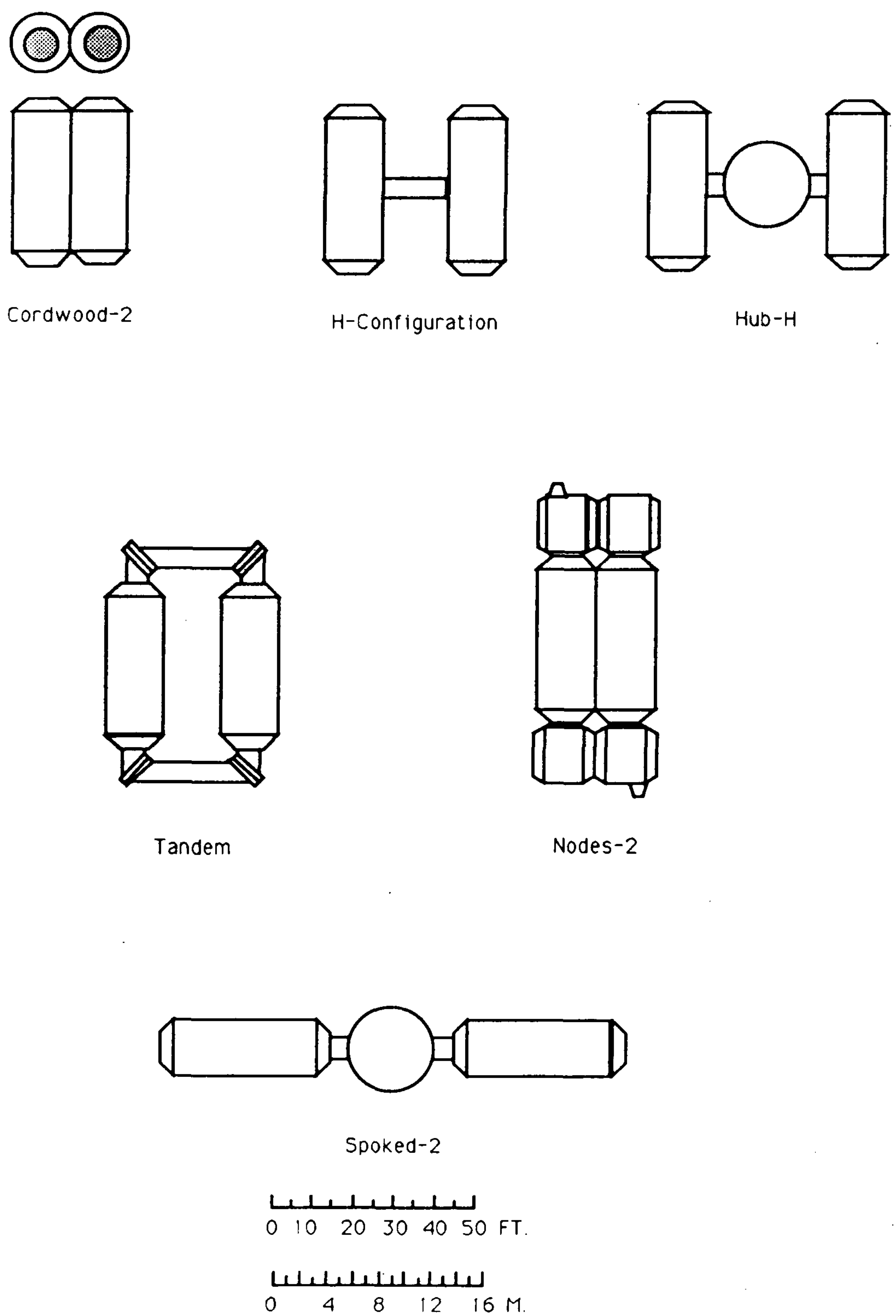


Table 5.1-2 Zero Gravity Configurations: Three Space Station Modules

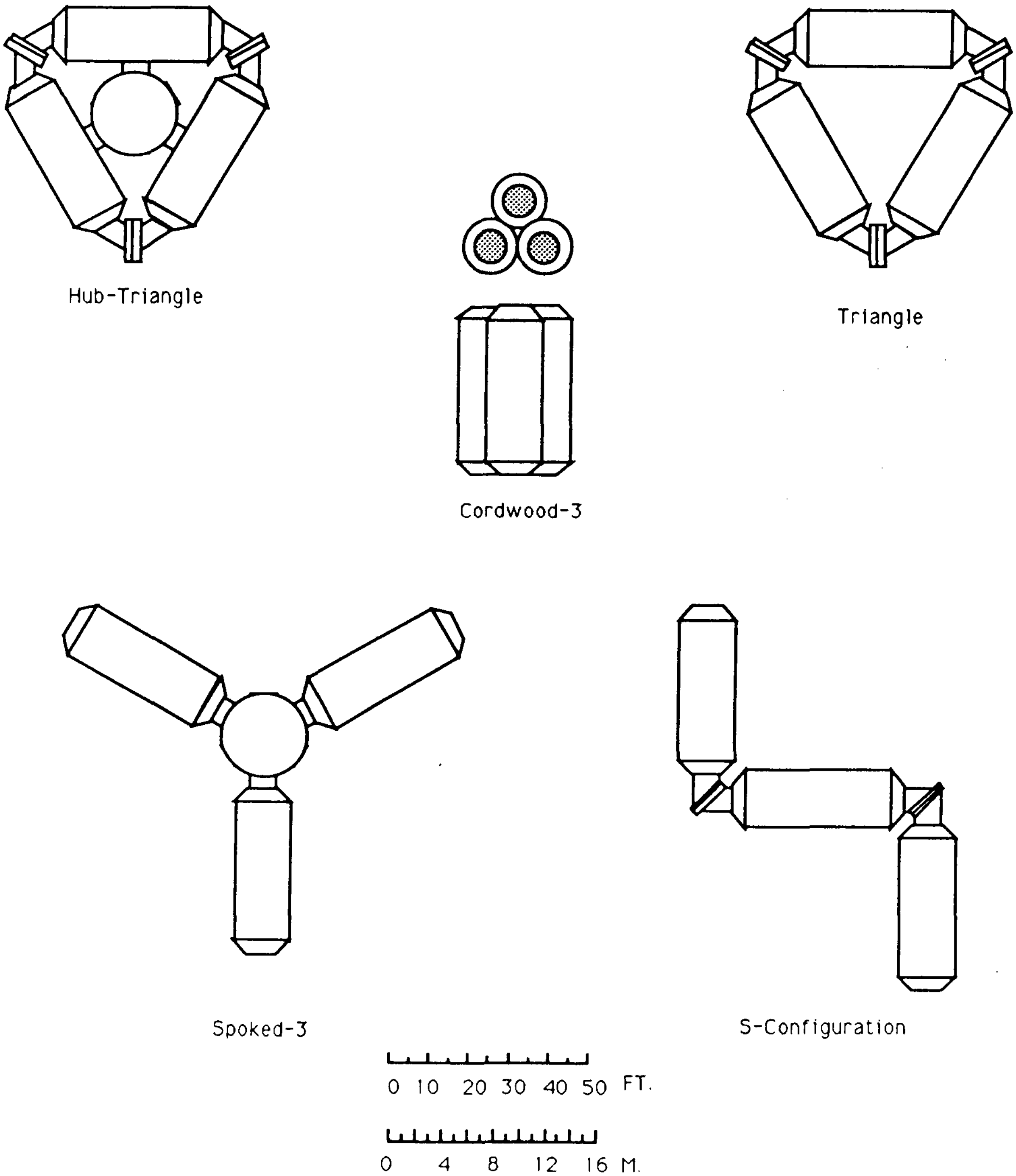


Figure 5.1-3 Zero Gravity Configurations: Four Space Station Modules

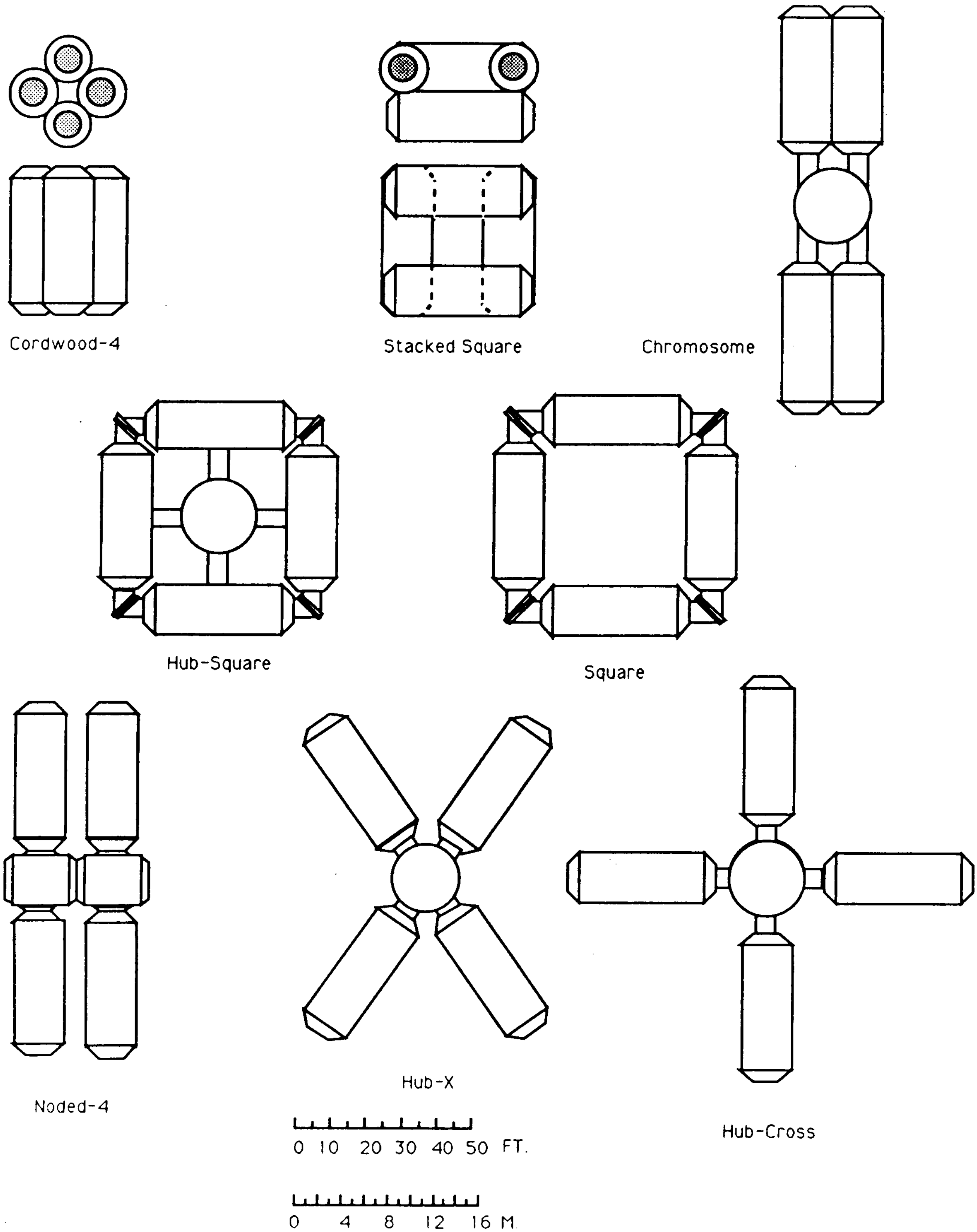


Figure 5.1-4 Zero Gravity Configurations: Five and Six Space Station Modules

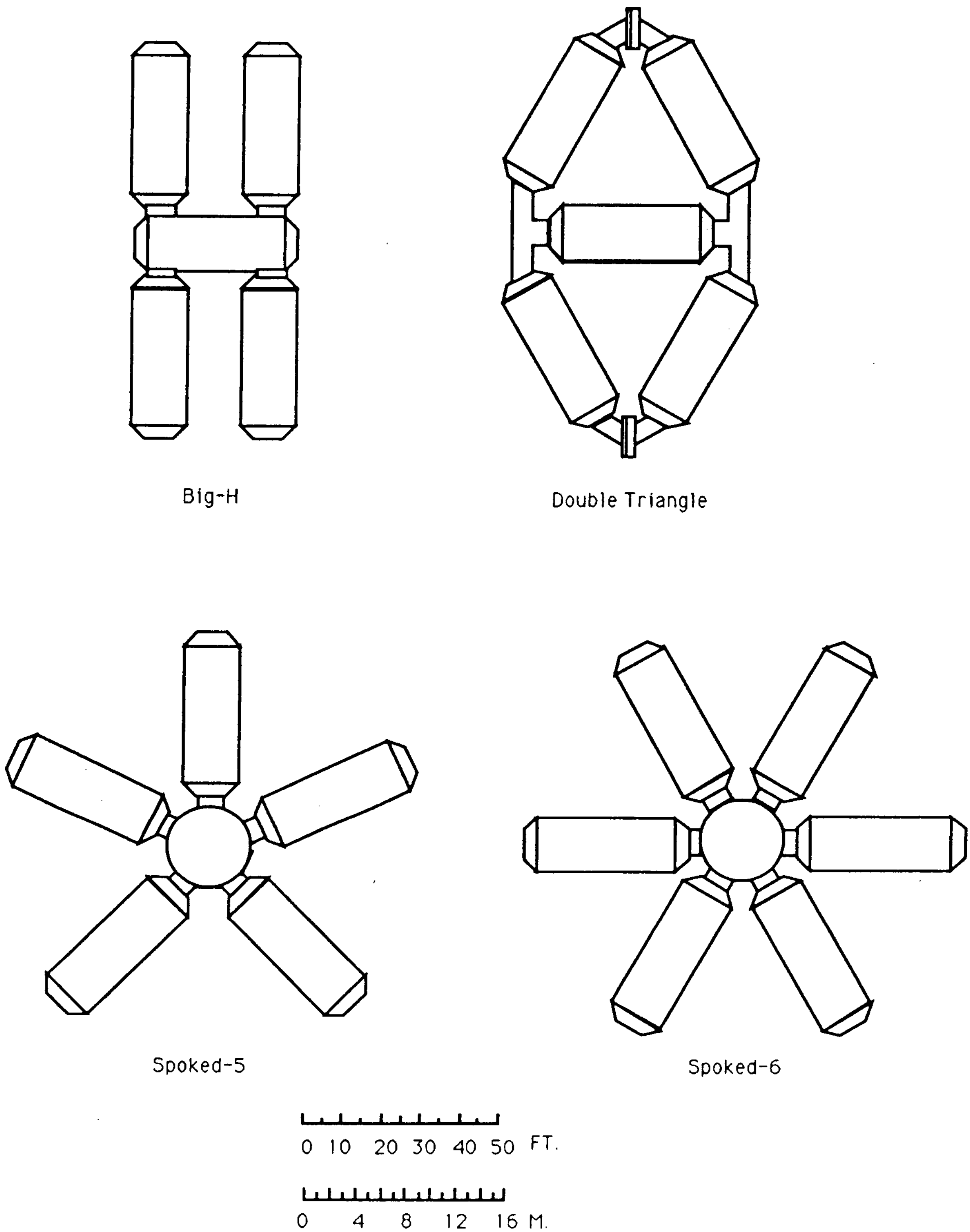
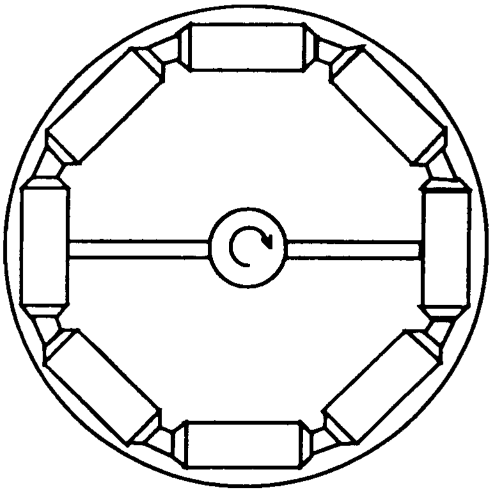
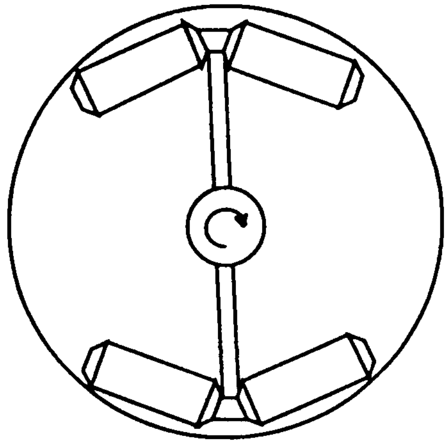


Figure 5.1-5 Artificial Gravity Configurations: Rigid Rotators

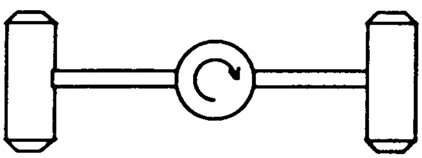


Octal Ring

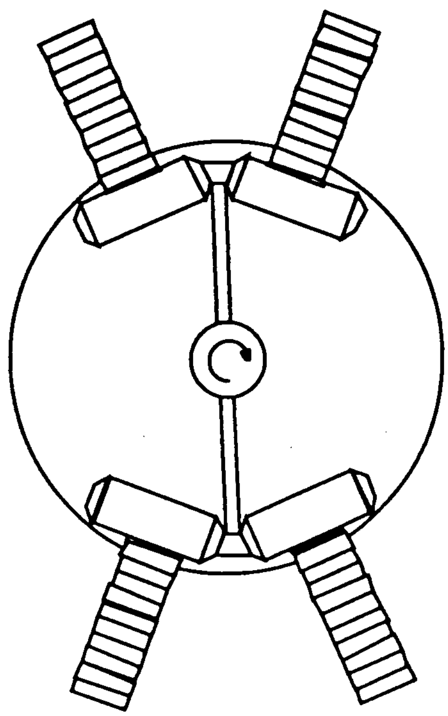


I-Beam

Dumbell A



Dumbell B



I-Beam with Photovoltaic Panels Deployed

Figure 5.1-6 Artificial Gravity Configurations: Tethered Systems

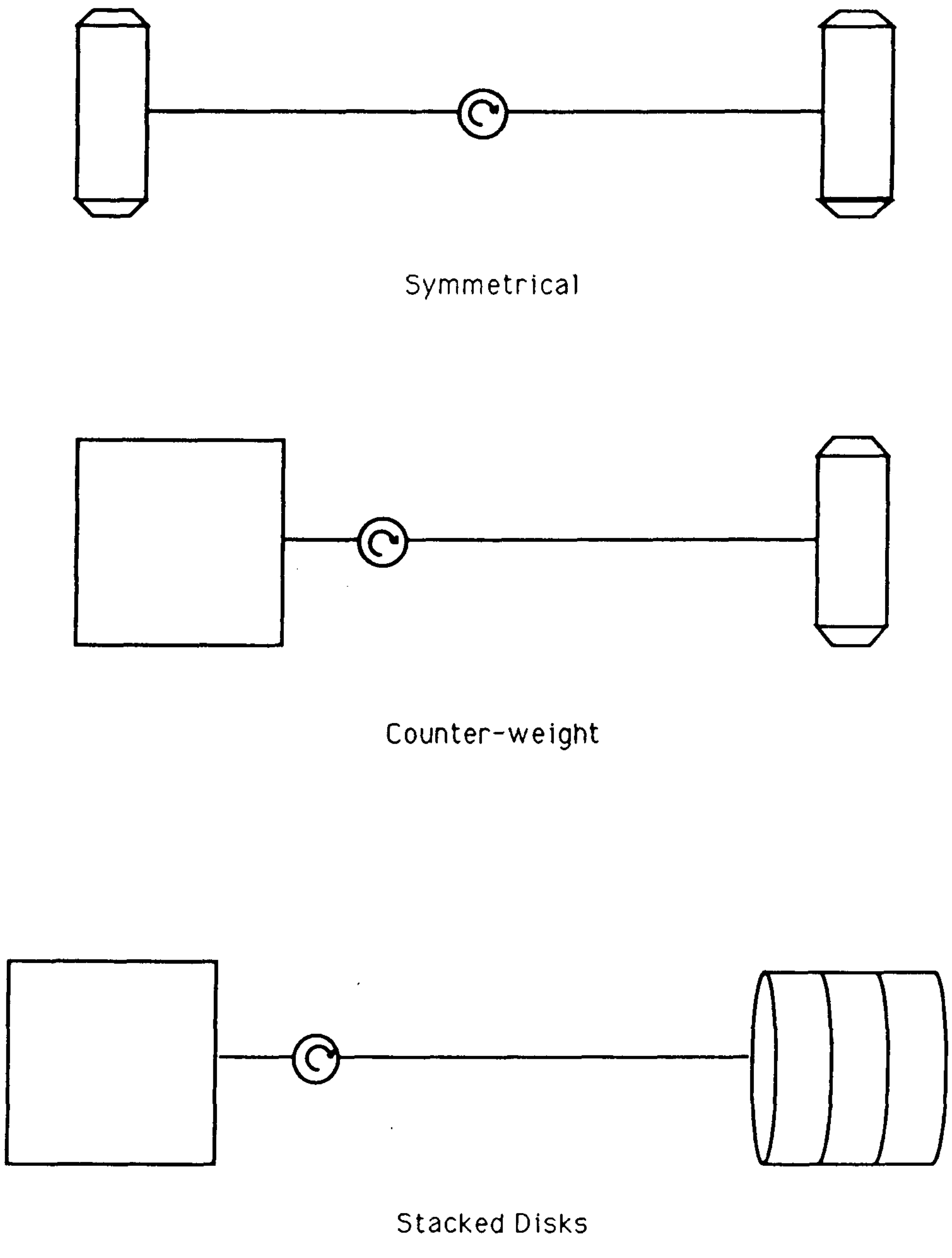


Figure 5.1-7 Pressurized Rover Storage
(Staggered Triangle-No Tilt)

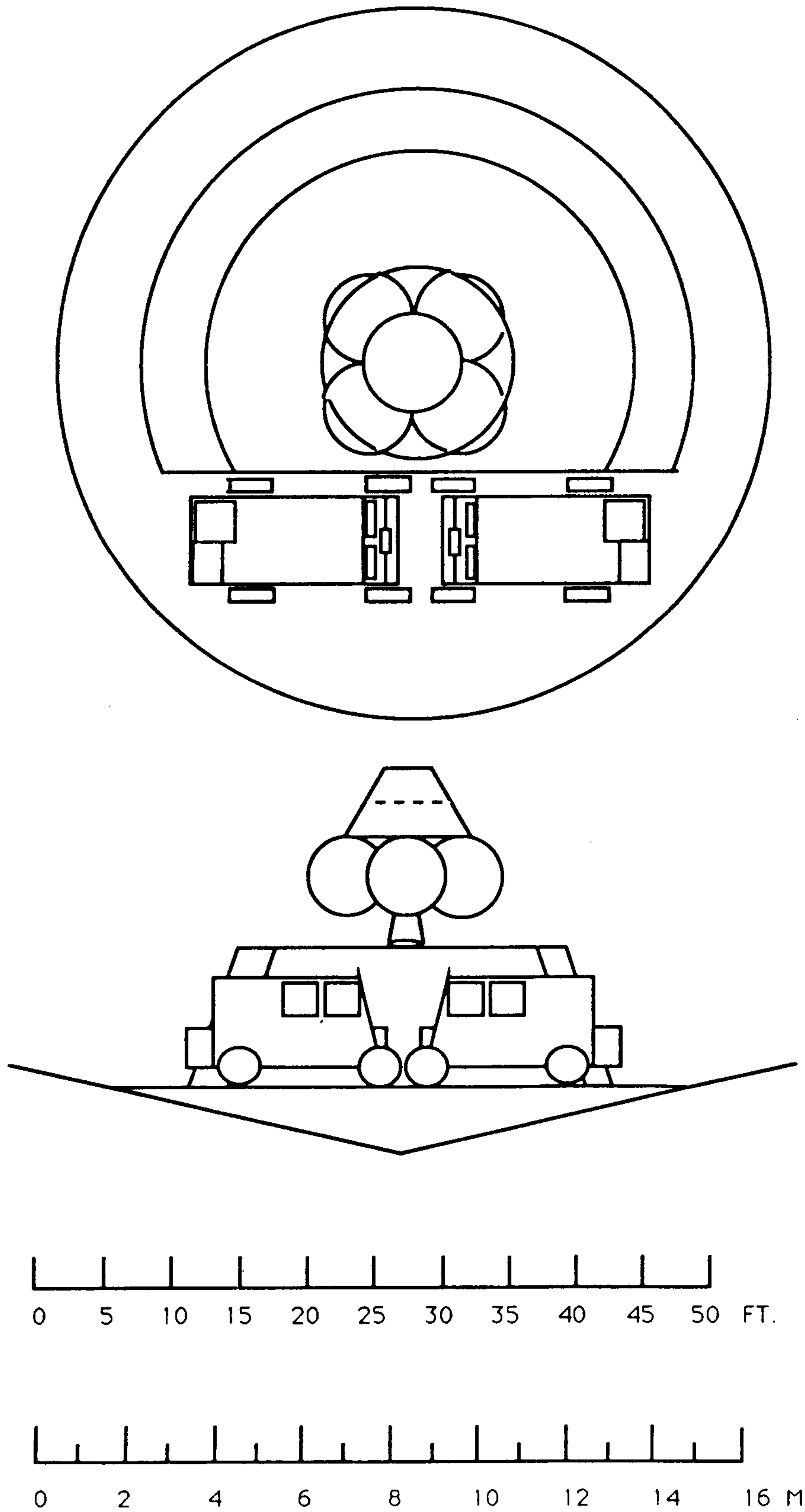


Figure 5.1-8 Pressurized Rover Storage (Staggered Triangle-10 degree tilt)

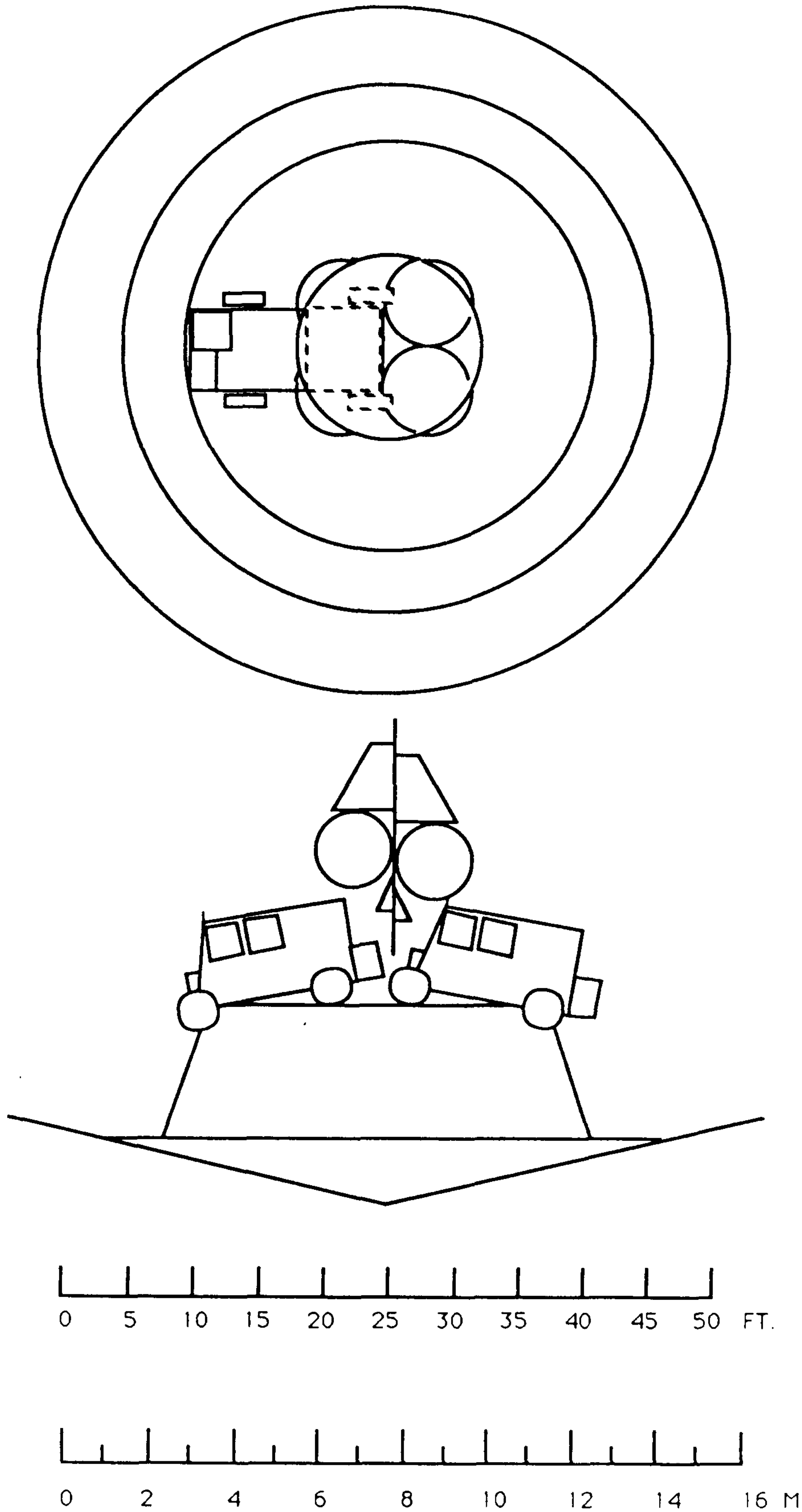


Table 5.1-1
Pressurized Volume and Atmosphere Mass

At Standard Temperature and Pressure, the mass of gas in the atmosphere the cylindrical living habitats is calculated using the Ideal Gas Law.

$$PV = nRT$$

$$P = 1 \text{ atm. (normal air)}$$

$$R = 0.08206 \text{ L} \bullet \text{atm/mol} \bullet \text{K}$$

$$T = 298 \text{ K}$$

15' Diameter Cylinder

Height(ft.)	Volume (m^3)	Atmosphere Mass(kg)	Mass(lbs.)
10	50	59	130
8	40	47	104
6	30	35	78

22' Diameter Cylinder

Height(ft.)	Volume (m^3)	Atmosphere Mass(kg)	Mass(lbs.)
10	108	127	280
8	86	102	224
6	65	76	168

25' Diameter Cylinder

Height(ft.)	Volume (m^3)	Atmosphere Mass(kg)	Mass(lbs.)
10	139	164	362
8	111	131	289
6	83	98	217

31' Diameter Cylinder

Height(ft.)	Volume (m^3)	Atmosphere Mass(kg)	Mass(lbs.)
10	214	252	556
8	171	202	445
6	128	151	334

5.2 Earth-to Orbit Evaluation

Earth-to-Orbit considerations often influenced the choice of configurations for a mission. The guidelines and assumptions which had to be worked around are listed in Table 5.2-1. In addition to the existing launch capabilities, new launch vehicles have been proposed which could decrease the number of launches necessary to put the total system into orbit, as well as decrease the number of components to be assembled. The pros and cons of such a system are listed in Table 5.2-2.

Several Transportation Node Strategies have also been considered. The pros and cons of assembling and launching spacecraft from a spacestation, GEO, HEO, lunar orbit, or Phobos/Deimos are listed in Table 5.2-3.

Table 5.2-1 Earth-to-Orbit (ETO) Guidelines and Assumptions

Launch Capabilities to Space Station LEO

	P/L Mass	P/L Size	
• HLLV	91 t [200 klb_m]	Standard: [33 x 80 ft]	10 m dia x 24.4 m
	"Special":	12.8 m dia x 38.1 m [42 x 125 ft]	
• STS	20 t [44 klb_m]	SS length: [14.5 x 45 ft]	4.4 m dia x 13.7 m
	Max length:	4.4 m dia x 18.3 m [14.5 x 60 ft]	

Multiple Payload Surcharges

Mass: 5% of total payload for structural support
Stack height: 10% of close-packed height

Launch frequency

- HLLV — nominal: one per month
  special: two per month (but only 18 per year)
- STS — as required, up to one per month

Table 5.2-2 Super HLLV -- Pros and Cons

Pros

- Reduces on-orbit assembly

    - On-orbit assembly for Mars missions cannot be totally avoided unless lift capability exceeds 1000 tonnes (10x Energia or Saturn V). In particular, assembly of propellant tanks (dry or loaded) cannot be avoided.

    - The *number* of on-orbit assembly elements can be reduced.

- Reduces the launch lead time required for on-orbit preparations

    - permits longer payload development time

- Cost reduction

    - Reduces launch operations costs

    - May reduce net launch costs

Table 5.2-2 (cont.) Super HLLV -- Pros and Cons

Cons

- Development costs may be excessive
- Development schedule risk
  - very large lift capacity greatly exceeds demonstrated state-of-the-art
- Increased program impact from a single failure
  - launch risk of a very large quantity of very expensive hardware
- On-orbit assembly will not be advanced
  - assembly capability is needed for Space Station and many other future projects, but the experience base would not be further extended by this project. This may simply result in postponement of an essential capability.

Table 5.2-3 Transportation Node Strategies

Node	Pro	Con
* Space Station	Then-Existing support services	Deep in g-well (large TMI stage)
* GEO	Stationary Orbit	OTV req'd for each transfer Circularization penalty Radiation hazards
* HEO	OTV for TMI	came as GEO, but no circ.
* Lunar Orbit/ Libration Point	LLOX (reduce ETO prop. launches)	Requires establishing and supporting Lunar Base; LLOX mfg. and transport
* Phobos/Deimos	Propellant for TEI, MDV, MAV	Requires base, mfg. plant; maroon risk

5.3 Mission Tanks and Engines

All of the engine and tanks that have been used in Case Studies 1 through 3 are tabulated in Table 5.3-1. The Advanced Space Engine and RL10-Derivative are extensions of the SSME and RL10B-2 engines, respectively. The main technique for improving engine specific impulse performance was to increase the expansion ratio by extending the nozzle length and diameter. The RL10B-2 engines and it's derivatives is shown in Figure 5.3-1, and the SSME and it's derivatives is shown in Figure 5.3-2. The mass breakdown for the SSME is shown in Table 5.3-2.

The tanks to which these engines are attached are designed to be lightweight with the structural integrity varying with mission loads. In order to accomplish this goal, a second "Siamese Twin" tank is needed to carry propellant to LEO. This tank is ruggedized to withstand launch vibrations but has none of the vapor-cooled shields or MLI that would be necessary for a mission tank. All-automatic bidirectional disconnects are used to tranfer propellant from the wet tank to the mission tank. The design approach for cryopropellant launches is given in Table 5.3-3. The assumptions and guidelines for such a system are listed in Table 5.3-4.

In addition to construction concerns, it was necessary to research the optimum dimension for an ETO launch vehicle. Figure 5.3-3 shows that three tanks each of 15 feet in diameter fit into the 32.2 foot HLLV shroud. Because several other launch vehicles of varying dimensions were also being considered, tanks of varying diameter weren't ruled out.

Figure 5.3-4 gives the volume vs. the diameter for three different cylinder heights for LH_2/LOX tanks. In Figures 5.3-5a through 5.3-5c the dimensions of liquid hydrogen tanks for a nuclear engine are explored.

Table 5.3-1 Engine Information

Element	Parameter	Case Study 1		Case Study 2	
TMIS, piloted	Number	1		1	
	Type	SSME-derivative		Advanced Space Engine	
	Thrust (total)	2,415 kN	[542.9 klb_f]	2,415 kN	[542.9 klb_f]
	I_{sp}	4.71 kN-s/kg	[480 sec]	4.76 kN-s/kg	[485 sec]
TMIS, cargo	Number	1		1	
	Type	SSME-derivative		Advanced Space Engine	
	Thrust (total)	2,415 kN	[542.9 klb_f]	2,415 kN	[542.9 klb_f]
	I_{sp}	4.71 kN-s/kg	[480 sec]	4.76 kN-s/kg	[485 sec]
TEIS	Number	3		6	
	Type	RL10B-2		RL10-derivative	
	Thrust (total)	294 kN	[66.1 klb_f]	588 kN	[132.2 klb_f]
	Isp	4.51 kN-s/kg	[460 sec]	4.61 kN-s/kg	[470 sec]
PhEV	Number	1		1	
	Type	Delta		Delta	
	Thrust (total)	44.5 kN	[10.0 klb_f]	44.5 kN	[10.0 klb_f]
	Isp	3.14 kN-s/kg	[320 sec]	3.14 kN-s/kg	[320 sec]
MOCS/MOOS	Number	4		3	
	Type	RL10B-2		RL10-derivative	
	Thrust (total)	392 kN	[88.1 klb_f]	294 kN	[66.1 klb_f]
	I_{sp}	4.51 kN-s/kg	[460 sec]	4.61 kN-s/kg	[470 sec]
MDV	Number			3	
	Type			Delta	
	Thrust (total)			134 kN	[30.0 klb_f]
	I_{sp}			3.14 kN-s/kg	[320 sec]
MAV	Number			4	
	Type			Delta	
	Thrust (total)			178 kN	[40.0 klb_f]
	I_{sp}			3.14 kN-s/kg	[320 sec]

Table 5.3-1 (cont.) Engine Information

Case Study 3

LTV	Number	4	
	Type	RL10B-2	
	Thrust (total)	392 kN	[88.1 klbf]
	I_{sp}	4.51 kN-s/kg	[460 sec]
LDV-C	Number	4	
	Type	RL10B-2	
	Thrust (total)	392 kN	[88.1 klbf]
	I_{sp}	4.51 kN-s/kg	[460 sec]
LDV-P	Number	4	
	Type	RL10B-2	
	Thrust (total)	392 kN	[88.1 klbf]
	Isp	4.51 kN-s/kg	[460 sec]
LAV	Number	3	
	Type	RL10B-2' (lightweight)	
	Thrust (total)	294 kN	[66.0 klbf]
	Isp	4.51 kN-s/kg	[460 sec]

Figure 5.3-1 RL10B-2 Engine and Derivatives

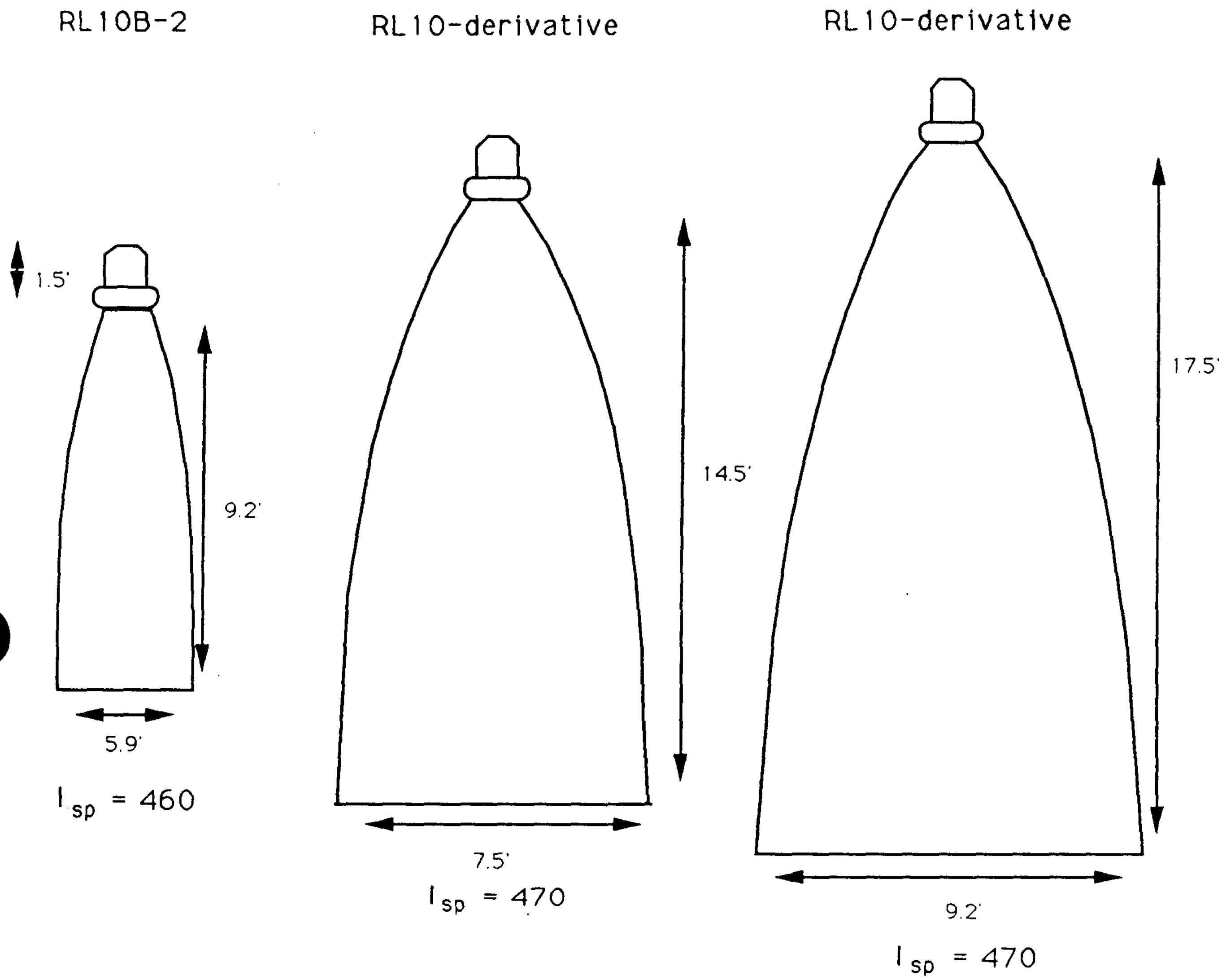
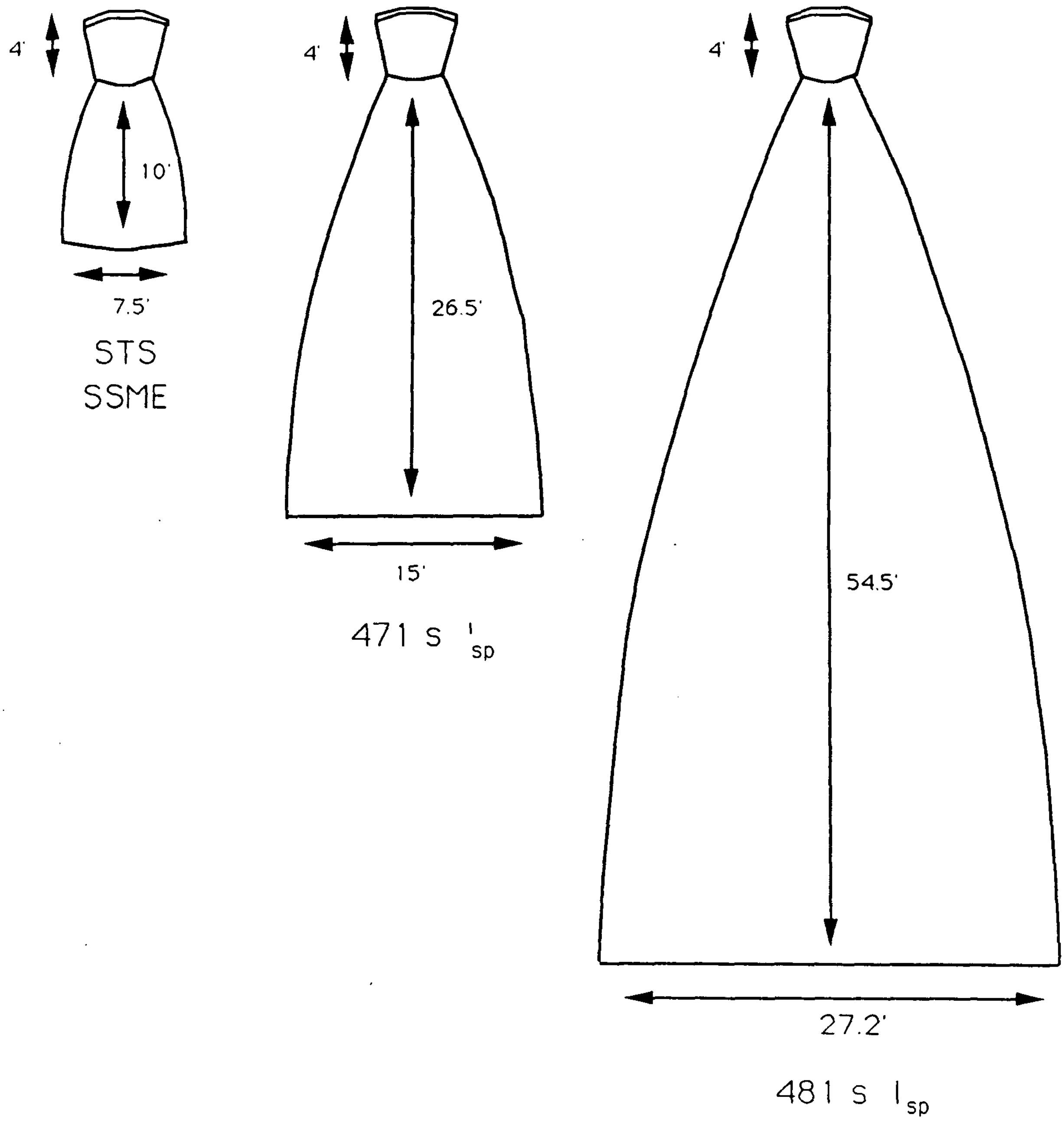


Figure 5.3-2 STS SSME and Derivatives



Relative Sizes for SSME Engines

Table 5.3-2 SSME Engine

Dry Mass	3,175.2 kg
Size	
Length	4.24 m
Diameter	
Powerhead	2.67 m by 2.4 m
Nozzle Exit	2.39 m
Propellant Type	LOX/LH2
Thrust (total)	2,366.6 kN
I_{sp} (481 sec)	4.71 kN-s/kg
Initial T/W	
Total Mass	
Full Power Level	109%
Minimum Power Level	65%
Life	
Hours	7.5
Starts	55

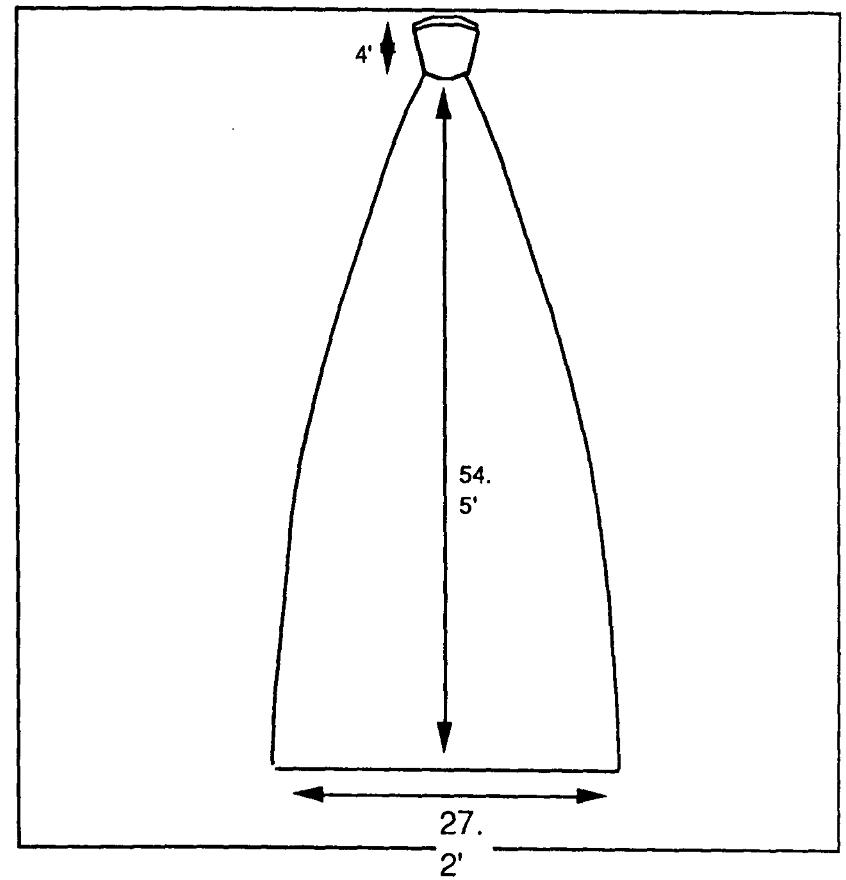


Table 5.3-3 Cryopropellant Launch Strategy

Siamese Twins Tank Design Approach

- **Wet Tank** (resupply tank)
  - Rugged construction for propellant-filled launch loads
  - Foam insulated; no vapor cooled shields or MLI
  - May draw upon ET design technology
  - 11.3 t dry weight, 69.3 t LH_2/LOX load
  - Topped off just prior to launch

- **Dry Tank** (mission tank)
  - Lightweight construction, advanced materials
  - Structural integrity consistent with mild mission loads
  - Vapor-cooled shields and MLI insulated
  - 10.4 t dry weight (15% tankage factor)
  - Partial pre-chill just prior to launch. May carry some propellant to orbit

- **Transfer of propellant**
  - On-orbit transfer from wet to dry tank through bidirectional disconnects
  - All-automatic transfer
  - Upper stage or augmentation engines provide fuel settling acceleration
  - Automated fill-line disconnects and scission of twins

Issues:

- tandem vs parallel twin configuration
- transportation of filled mission tank (upper stage, on-board engine, OMV, etc.)
- engine provisions
- wet and dry tank designs for minimum mass

Table 5.3-4 Cryoprop Tankage Assumptions and Guidelines

Tankage Factor*

very conservative (LTCSF**)	0.256***
nominal (feasible technology)	0.15
advanced technology	0.075

ETO Propellant loads**

Each 91 t HLLV P/L to SS orbit contains:

79.7 t of propellant+tankage equivalent (nominal tank factor of 0.15)

(69.3 t [152.7 klb] LH_2/LOX propellant; 10.4 t of TMIS tank; 11.3 t of resupply tank)

(Note: propellant is loaded into the resupply tank for launch, then is transferred on-orbit into the dry tank that is launch together with this load)

* Tankage factor = ratio of tank dry mass to propellant mass

** Source: "Long Term Cryogenic Storage Facility (LTCSF) Systems Study", NAS8-36612.

*** Wet launched, 200 klb.

Figure 5.3-3 Three 15' Diameter Cylinders in a Larger Cylinder

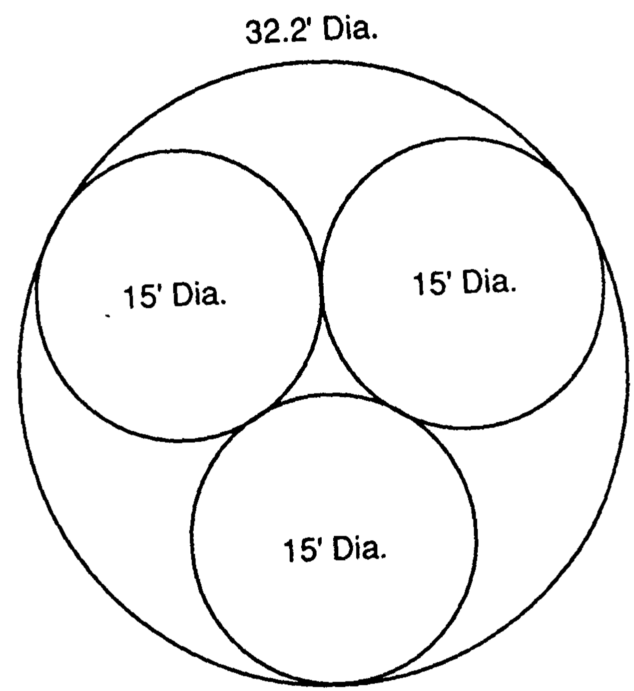


Figure 5.3-4 Volume vs. Diameter for Three Cylinder Heights

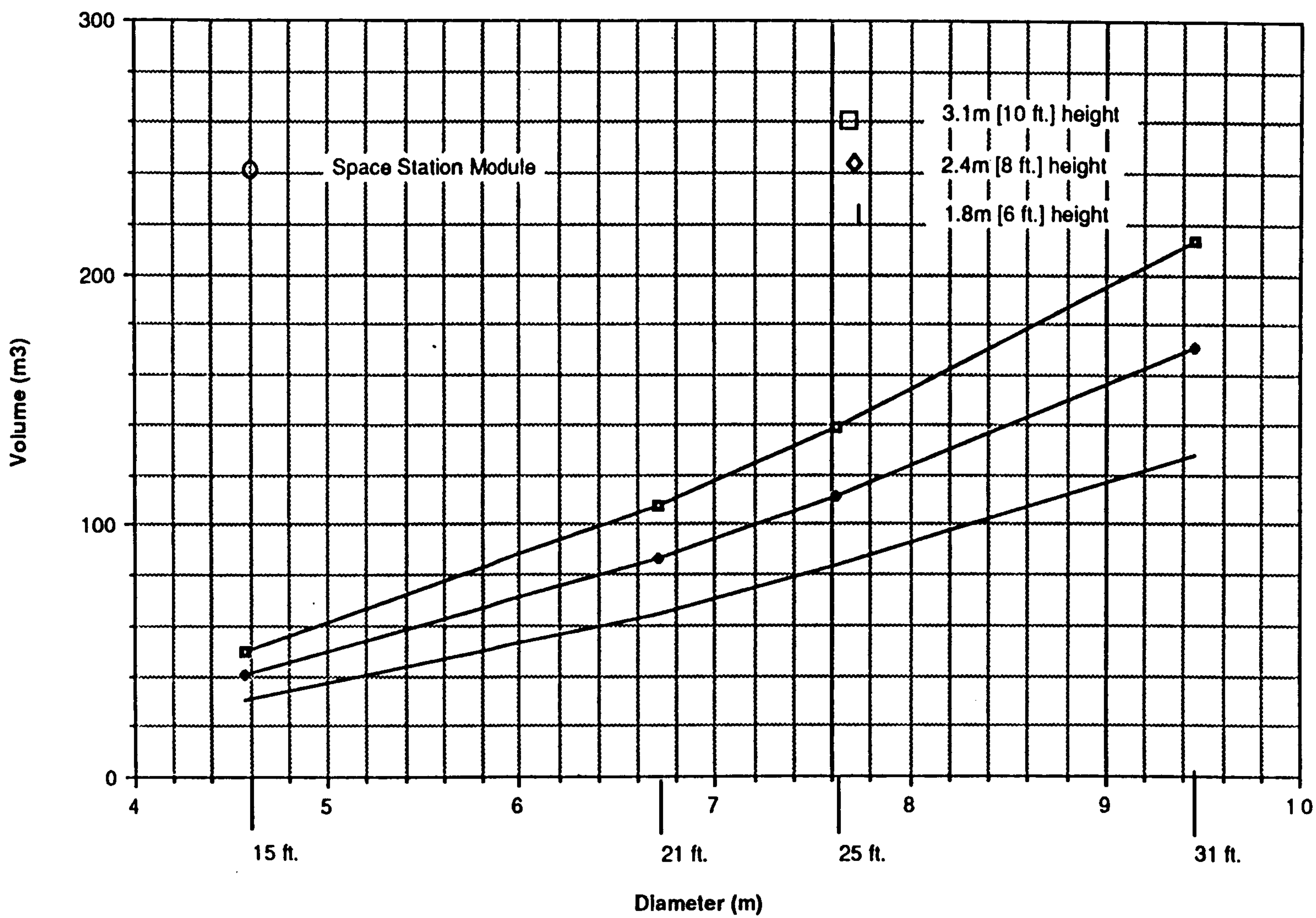


Figure 5.3-5a Tank Dimensions for Liquid Hydrogen Propellant Mass

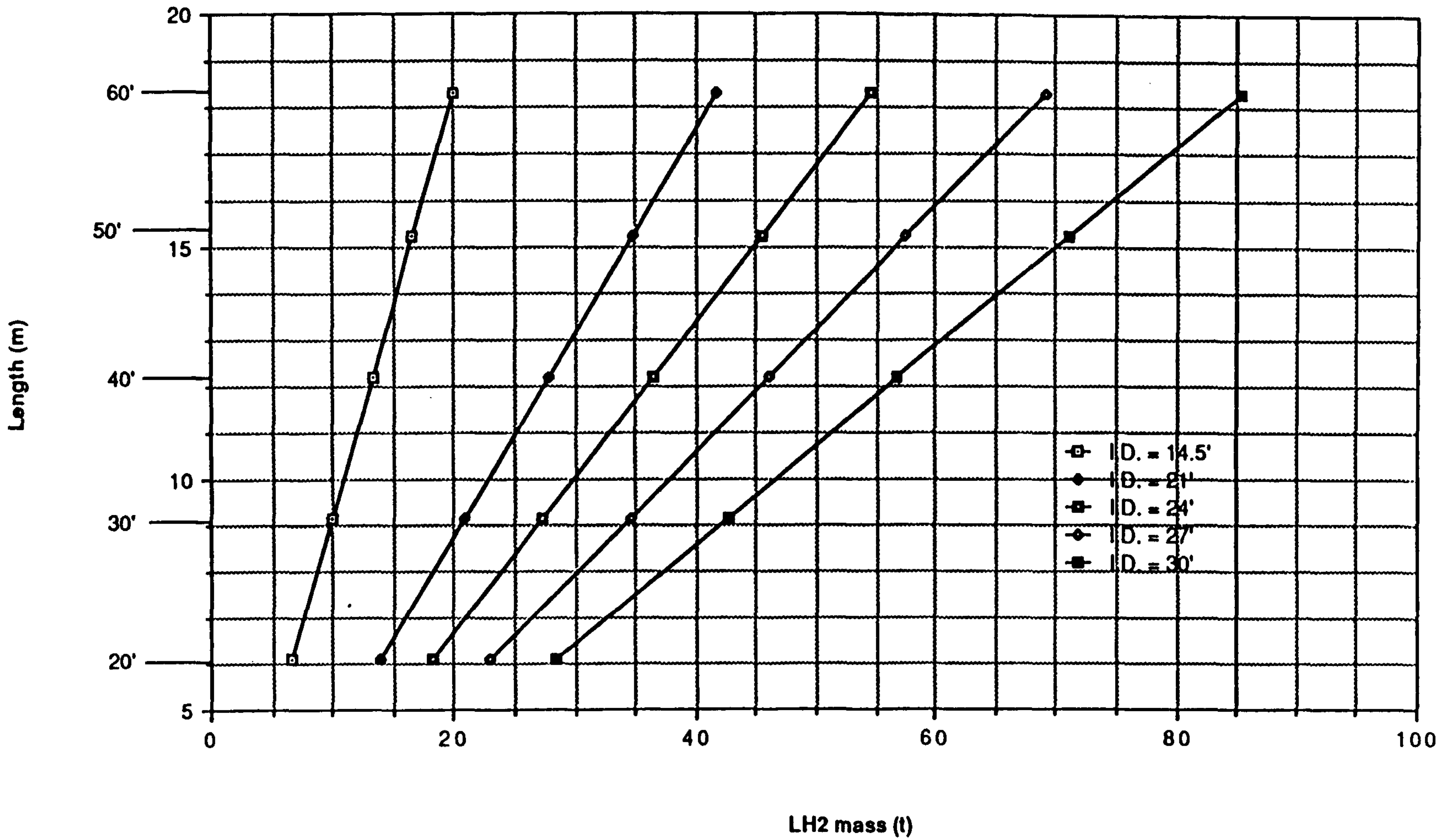


Figure 5.3-5b Tankage Factor vs. LH2 Mass (I.D. = 24')

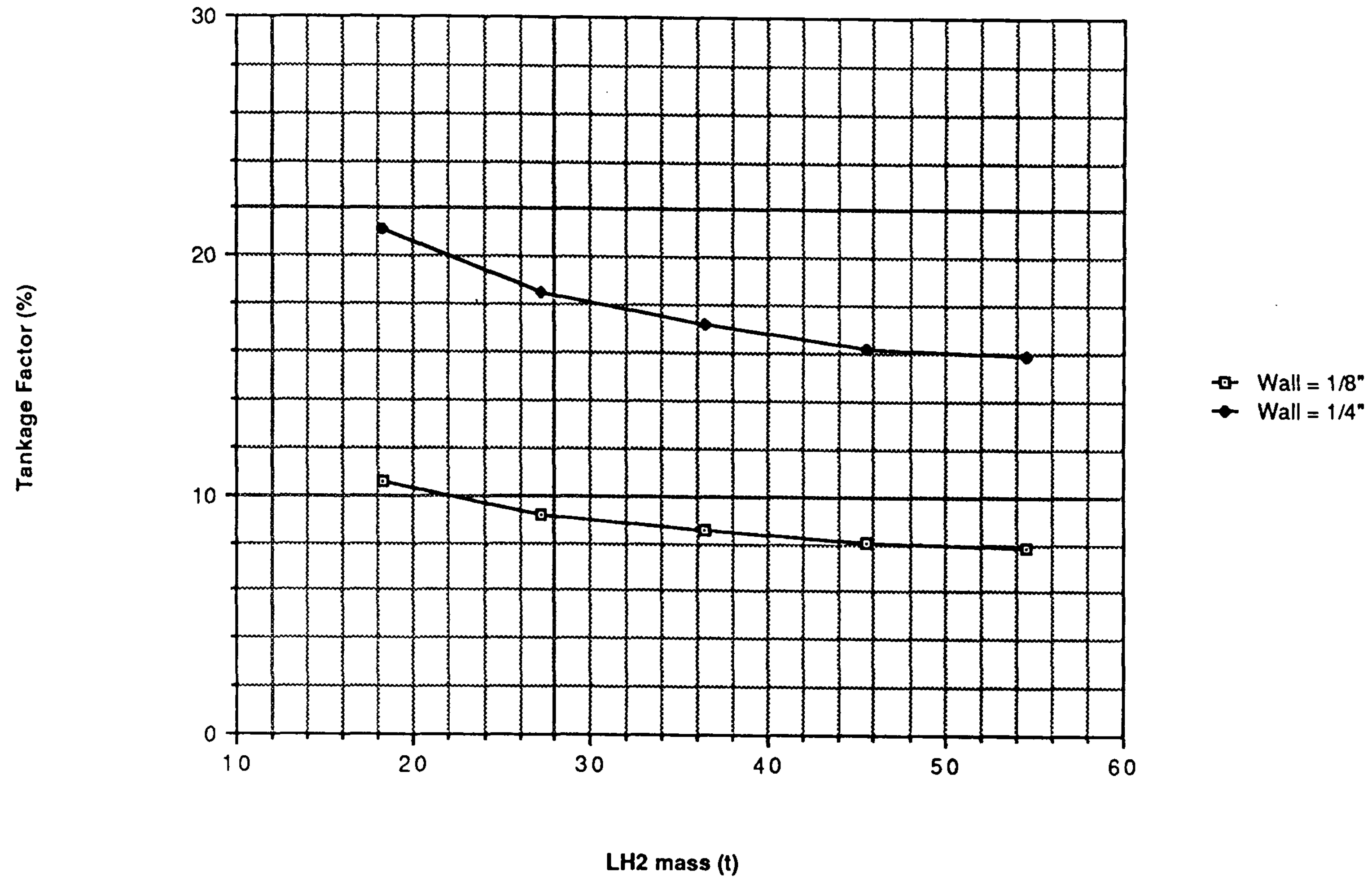
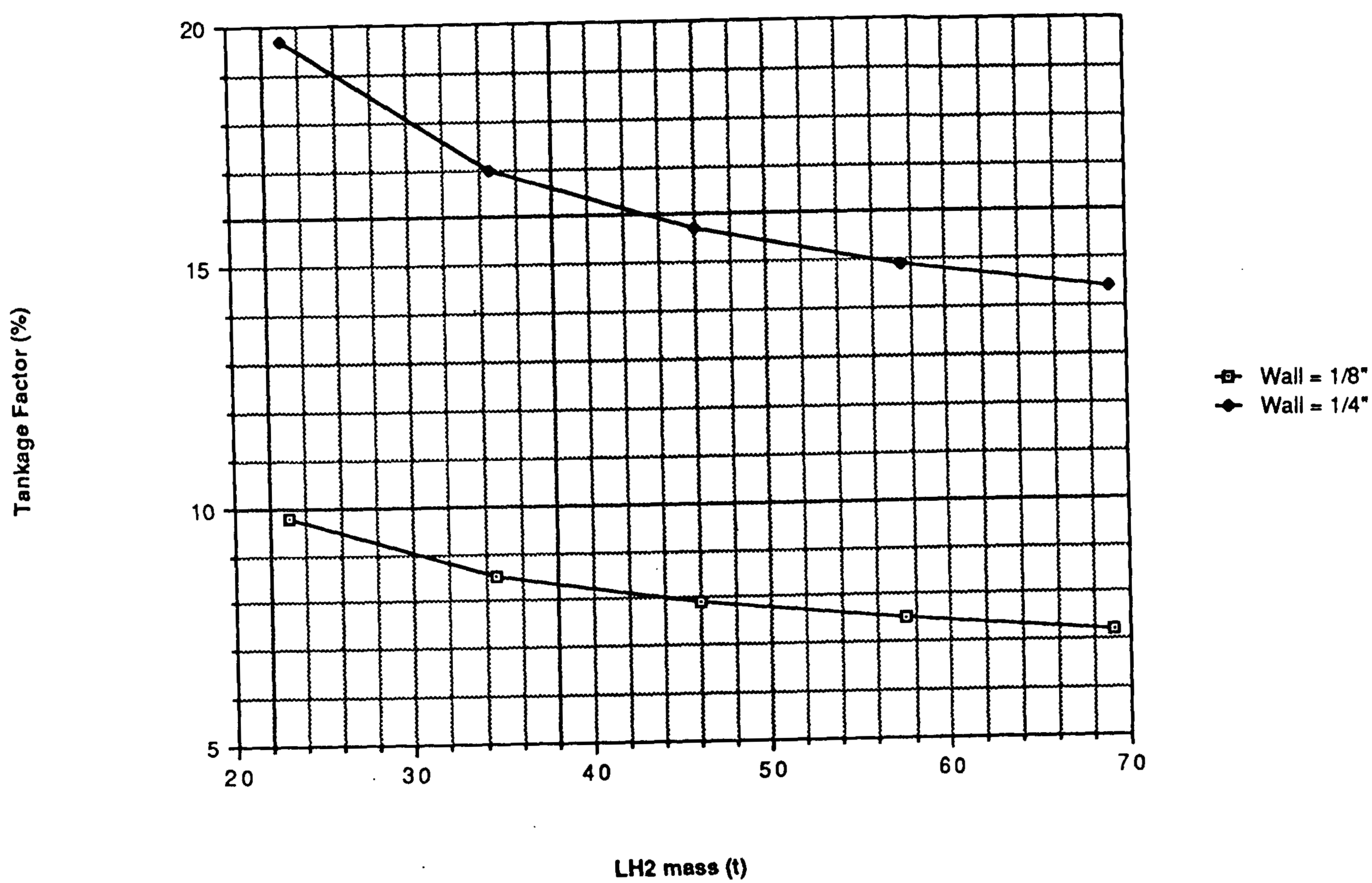


Figure 5.3-5c Tankage Factor vs. LH2 Mass (I.D. = 27')



5.4 Radiation Storm Shelter Variation

The concept for a radiation storm shelter was developed to be utilized in the event of an Anomalously Large Solar Particle Event (ALSPE) occurring during a manned mission to Mars. Data from three large historical solar flares were used as a basis for computations of radiation doses resulting from various shelter geometries and crew sizes. The three historical Solar Particle Events (SPE) used have been named for the dates of occurrence: (1) February 1956, (2) November 1960, and (3) August 1972. Integrated angular dose equivalents for each of the three flares calculated for a spherical storm shelter geometry, along with the associated shielding mass and volume parameters are shown in Table 5.4-1. The concept of a "radiation vest", which provides 5g/cm^2 additional shielding to the blood forming organs, was also introduced (Figure 5.4-1). The radiation vest covers blood forming organs located from above the knees to below the shoulders, and protects vital organs in addition to the shielding offered by the storm shelter walls and spacecraft instrumentation.

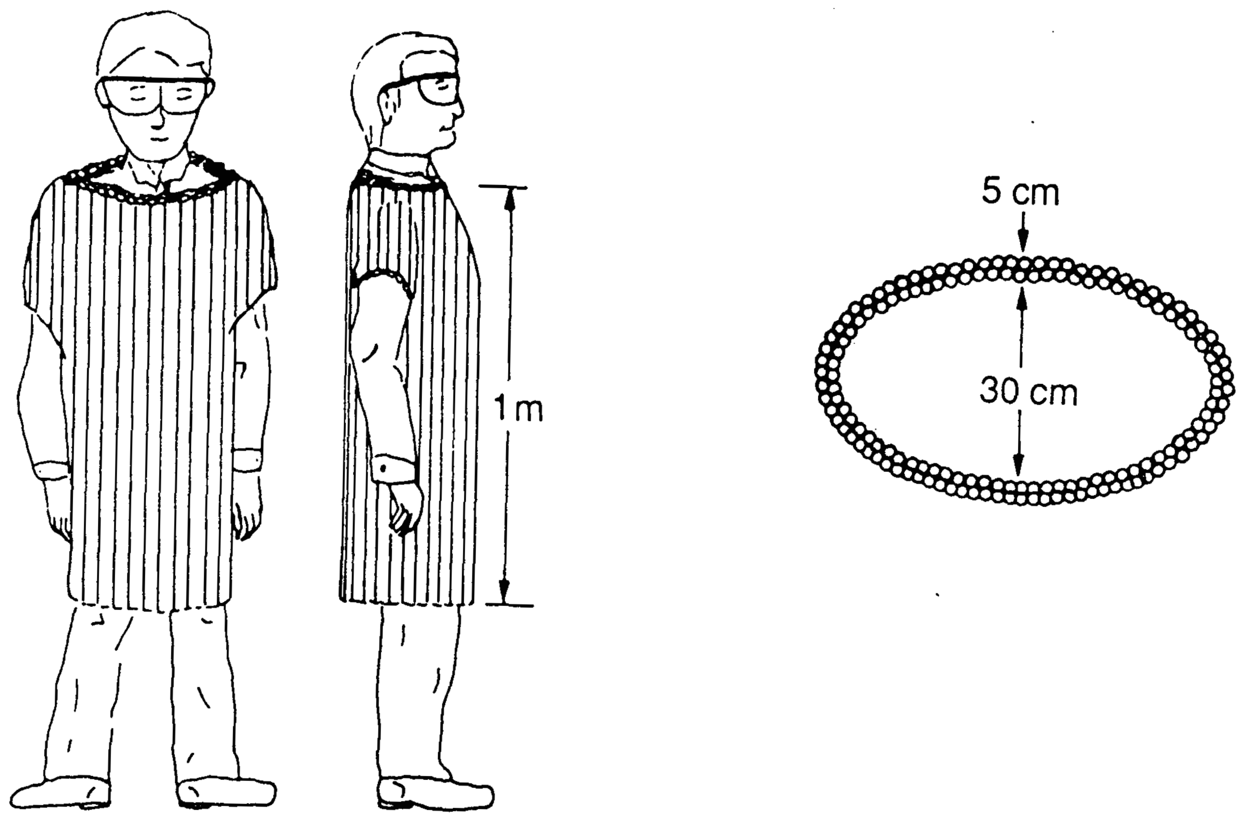
A sketch showing how a cubical radiation storm shelter could be implemented within the end of a cylindrical space station type module is depicted in Figure 5.4-2. Equipment racks are shown which hinge in to form the end doors of the shelter, and multiple compartments similar to safety deposit boxes within the shelter provide storage of consumables and waste products during the mission.

The total dose equivalent resulting from each of the historical SPE's is plotted as a function of storm shelter shielding mass, both with and without the radiation vest implementation in Figures 5.4-3 and 5.4-4. Parametric curves relating shield mass and shield volume for a spherical radiation storm shelter sized for a crew of 4 have also been generated (Figures 5.4-5 and 5.4-6). Finally, Table 5.4-2 shows the crew of 6 mass requirements for 20 g/cm^2 and 30 g/cm^2 shielding for various storm shelter geometries and summarizes the parametric evaluation of radiation shelters which were investigated.

Table 5.4-1 Spherical Geometry—Total SPE Dose (rem)

				Shield-Flare Dose		
	S Tkns (g/cm2)	Mass (kg)	Mass w/vest	Sph-Aug72	Sph-Nov60	Sph-Feb56
1	5	481	220	27.7	29	27.8
2	10	1019	701	12.9	20.8	24.4
3	15	1616	1239	6.5	15.9	22.1
4	20	2277	1836	3.4	12.1	20
5	25	3003	2497	2.1	9.4	18.2
6	30	3798	3223	1.3	7.3	16.6

Figure 5.4-1 Radiation Vest



<u>Water Filled</u> Radiation Vest - provides additional 5 g/cm^2 shielding around the blood forming organs.

Figure 5.4-2 Radiation Shelter Situated at Habitat Module End

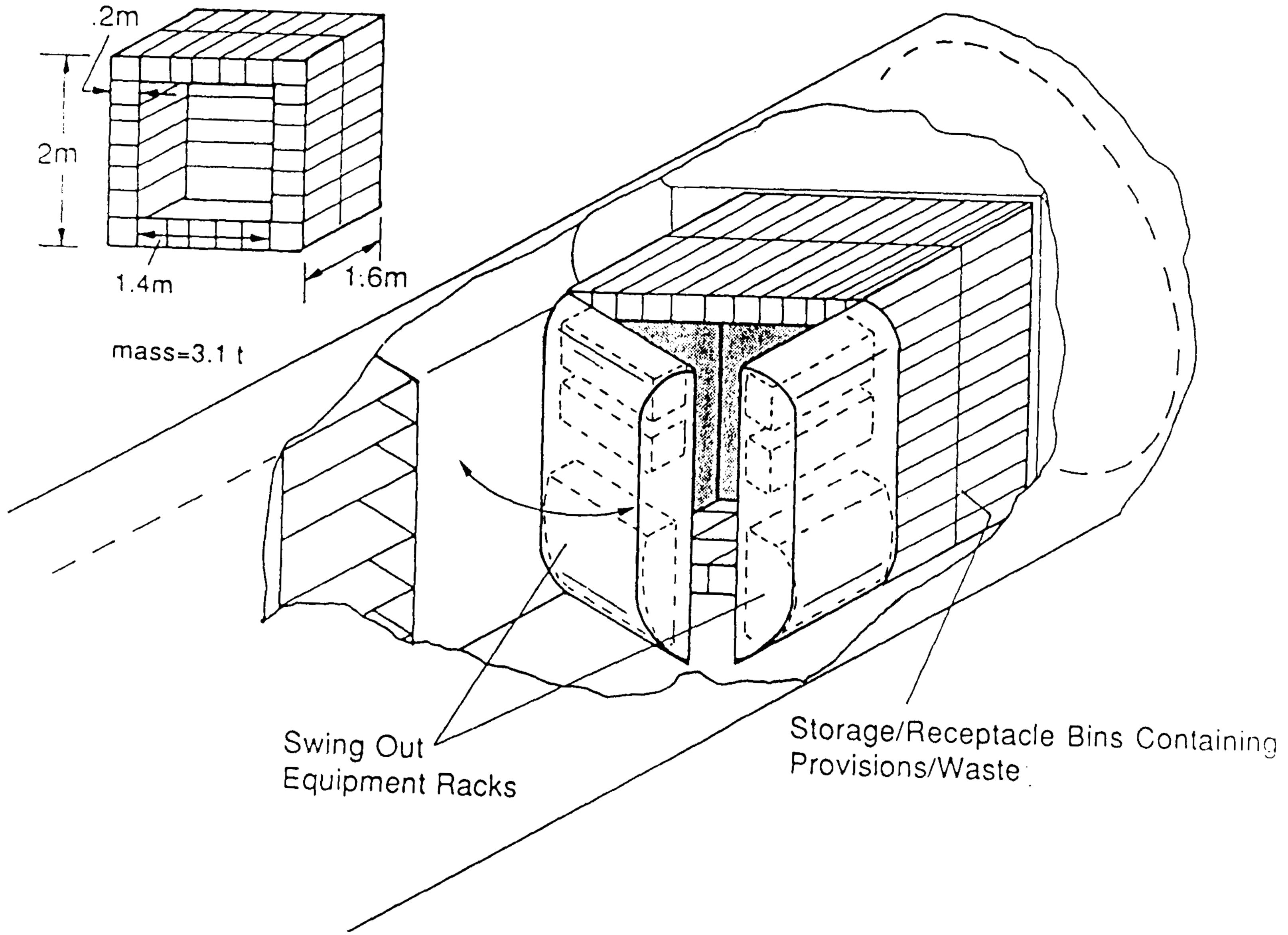
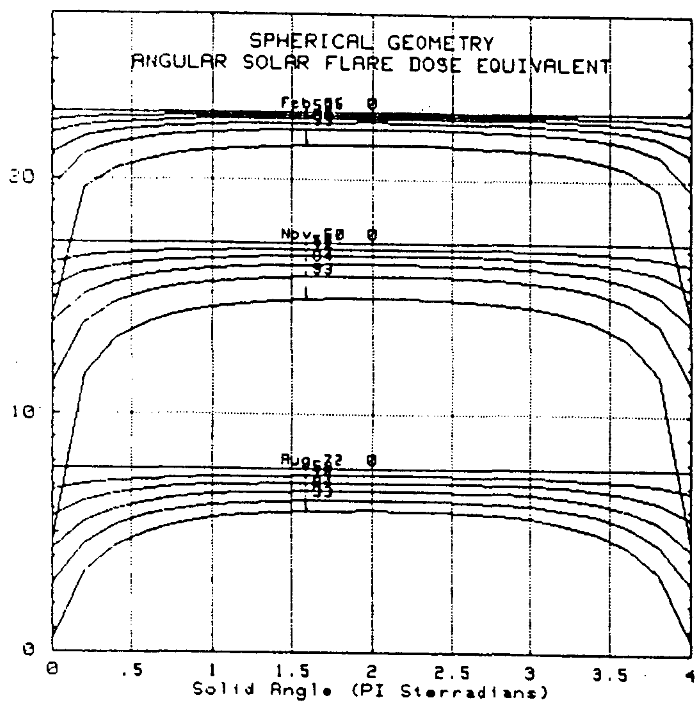


Figure 5.4-3 Parametric Curves Relating Shield Mass and Shield Volume

*** RADIATION SHIELDING CALCULATION *** 17 June 1988

MASS SUMMARY: SPHERICAL geometry of 1.7 meters inside diameter.
15 cm shielding thickness of 1 g/cm3 material
TOTAL SHIELD VOLUME: 1.62 cubic meters
TOTAL SHIELD MASS: 1616.3 kilograms (15 g/cm2)
This is a 4 person shelter, with an average person density of .11 g/cm3.

RADIATION INTEGRAL PARAMETERS:
Radial integration computed in 5 equal .514 m3 steps
Constant .2 pi sterradian steps used in Theta integration.
15 degree steps used in Phi integration.
Integral averaged radiation dose: Aug 72: 6.4 rem
Nov 60: 15.8 rem
Feb 56: 22 rem



*** RADIATION SHIELDING CALCULATION *** 17 June 1988

MASS SUMMARY: SPHERICAL geometry of 1.7 meters inside diameter.
20 cm shielding thickness of 1 g/cm3 material
TOTAL SHIELD VOLUME: 2.28 cubic meters
TOTAL SHIELD MASS: 2276.6 kilograms (20 g/cm2)
This is a 4 person shelter, with an average person density of .11 g/cm3.

RADIATION INTEGRAL PARAMETERS:
Radial integration computed in 5 equal .514 m3 steps
Constant .2 pi sterradian steps used in Theta integration.
15 degree steps used in Phi integration.
Integral averaged radiation dose: Aug 72: 3.4 rem
Nov 60: 12.1 rem
Feb 56: 20 rem

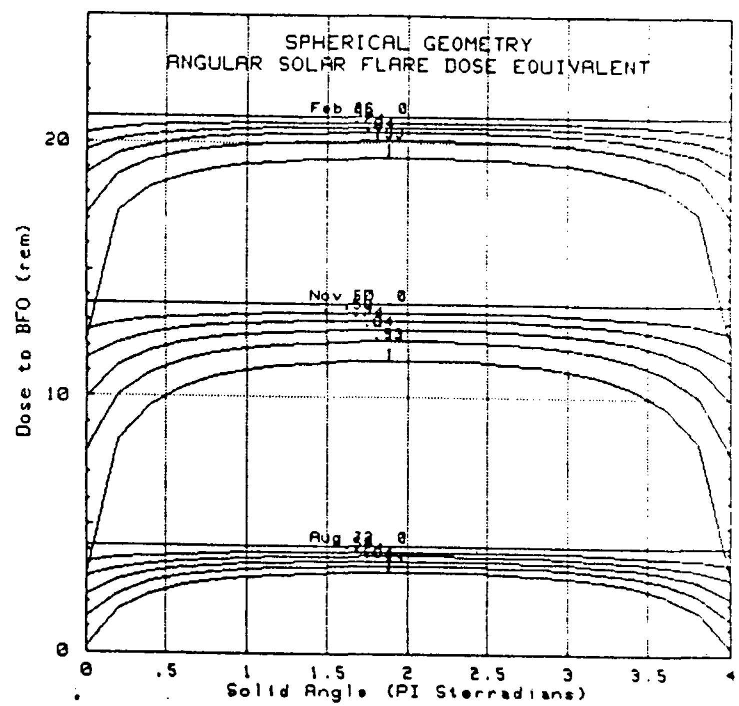


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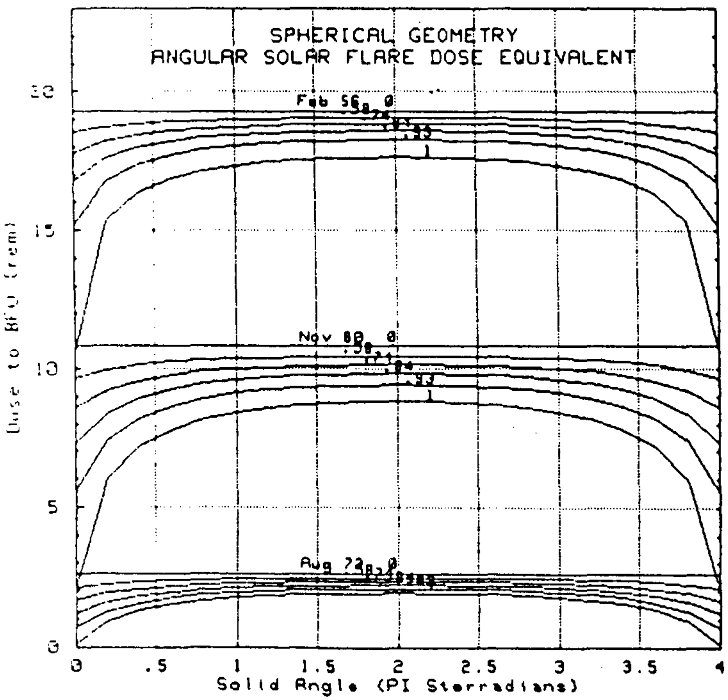
Figure 5.4-4 Parametric Curves Relating Shield Mass and Shield Volume

```
***     RADIATION SHIELDING CALCULATION     ***     17 June 1988

MASS SUMMARY:     SPHERICAL geometry of 1.7 meters inside diameter.
      25 cm shielding thickness of 1 g/cm3 material
TOTAL SHIELD VOLUME:  3 cubic meters
TOTAL SHIELD MASS:  3002.8 kilograms          ( 25 g/cm2 )
This is a 4 person shelter, with an average person density of .11 g/cm3.

RADIATION INTEGRAL PARAMETERS:
Radial integration computed in 5 equal .514 m3 steps
Constant .2 pi sterradian steps used in Theta integration.
 15 degree steps used in Phi integration.
Integral averaged radiation dose:          Aug 72:    2.1 rem
                                           Nov 60:    9.4 rem
                                           Feb 56:   18.2 rem
```

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***     RADIATION SHIELDING CALCULATION     ***     17 June

MASS SUMMARY:     SPHERICAL geometry of 1.7 meters inside diameter.
      30 cm shielding thickness of 1 g/cm3 material
TOTAL SHIELD VOLUME:  3.8 cubic meters
TOTAL SHIELD MASS:  3798.2 kilograms          ( 30 g/cm2 )
This is a 4 person shelter, with an average person density of .11 g

RADIATION INTEGRAL PARAMETERS:
Radial integration computed in 5 equal .514 m3 steps
Constant .2 pi sterradian steps used in Theta integration.
 15 degree steps used in Phi integration.
Integral averaged radiation dose:          Aug 72:    1.3 rem
                                           Nov 60:    7.3 rem
                                           Feb 56:   16.6 rem
```

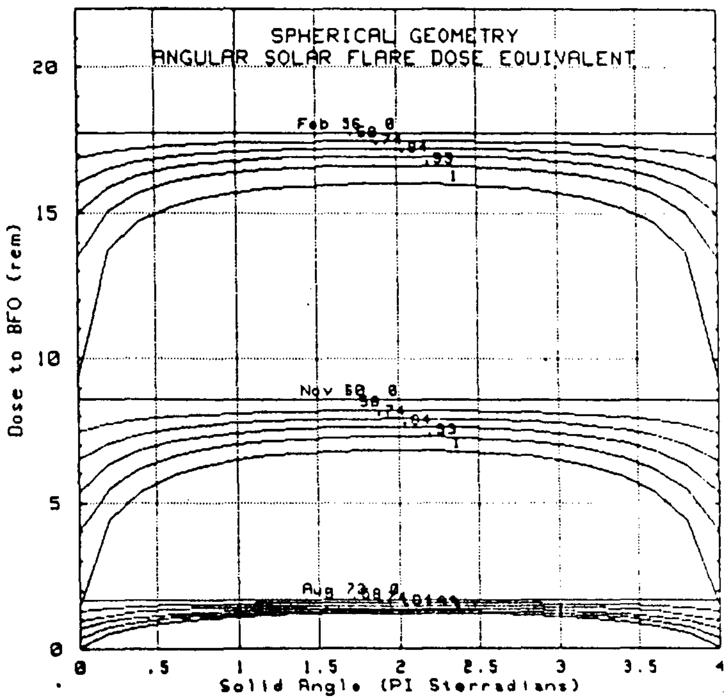

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Figure 5.4-5 Spherical Geometry–Crew of 4

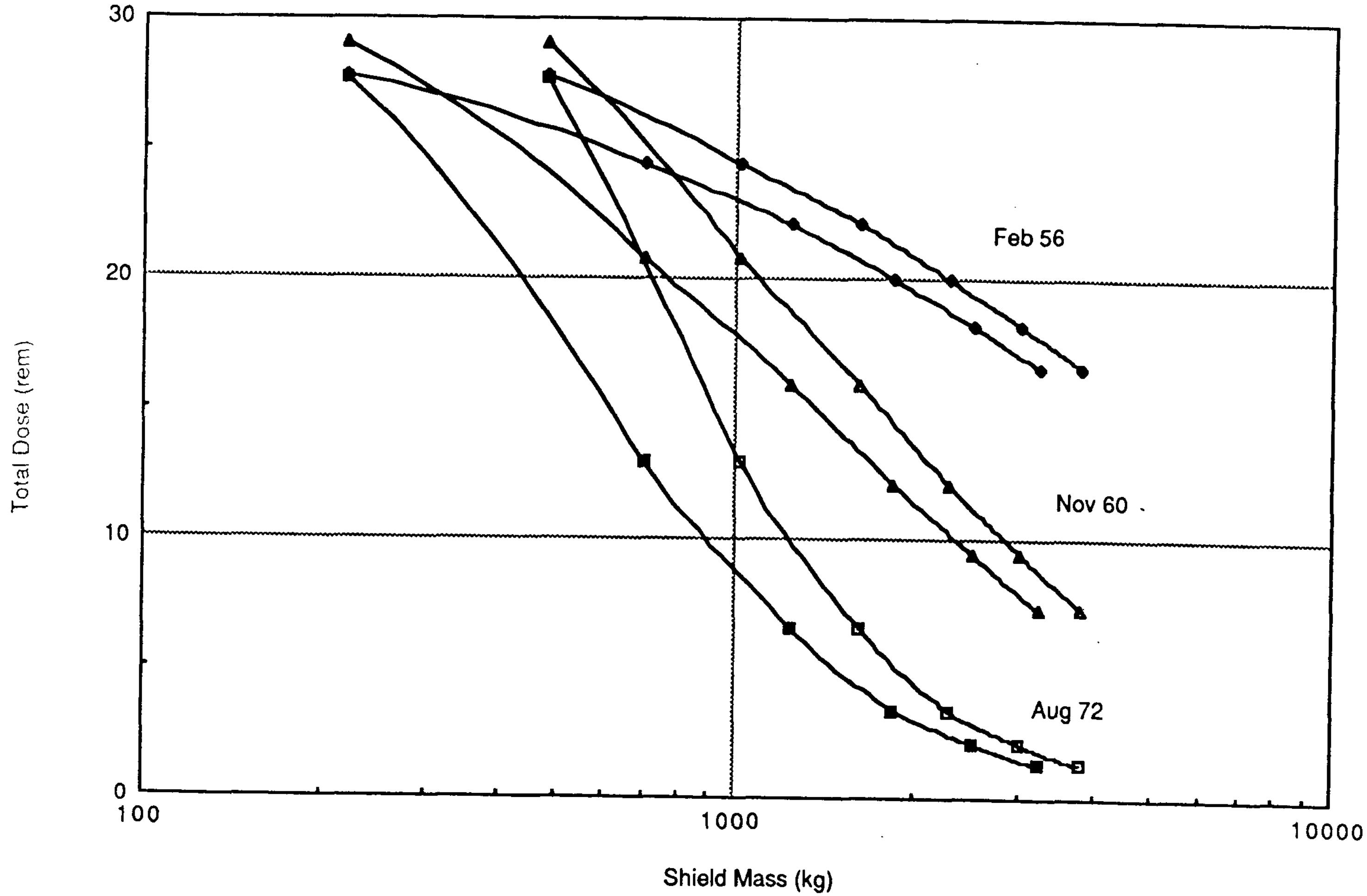


Figure 5.4-6 Spherical Geometry--Crew of 4

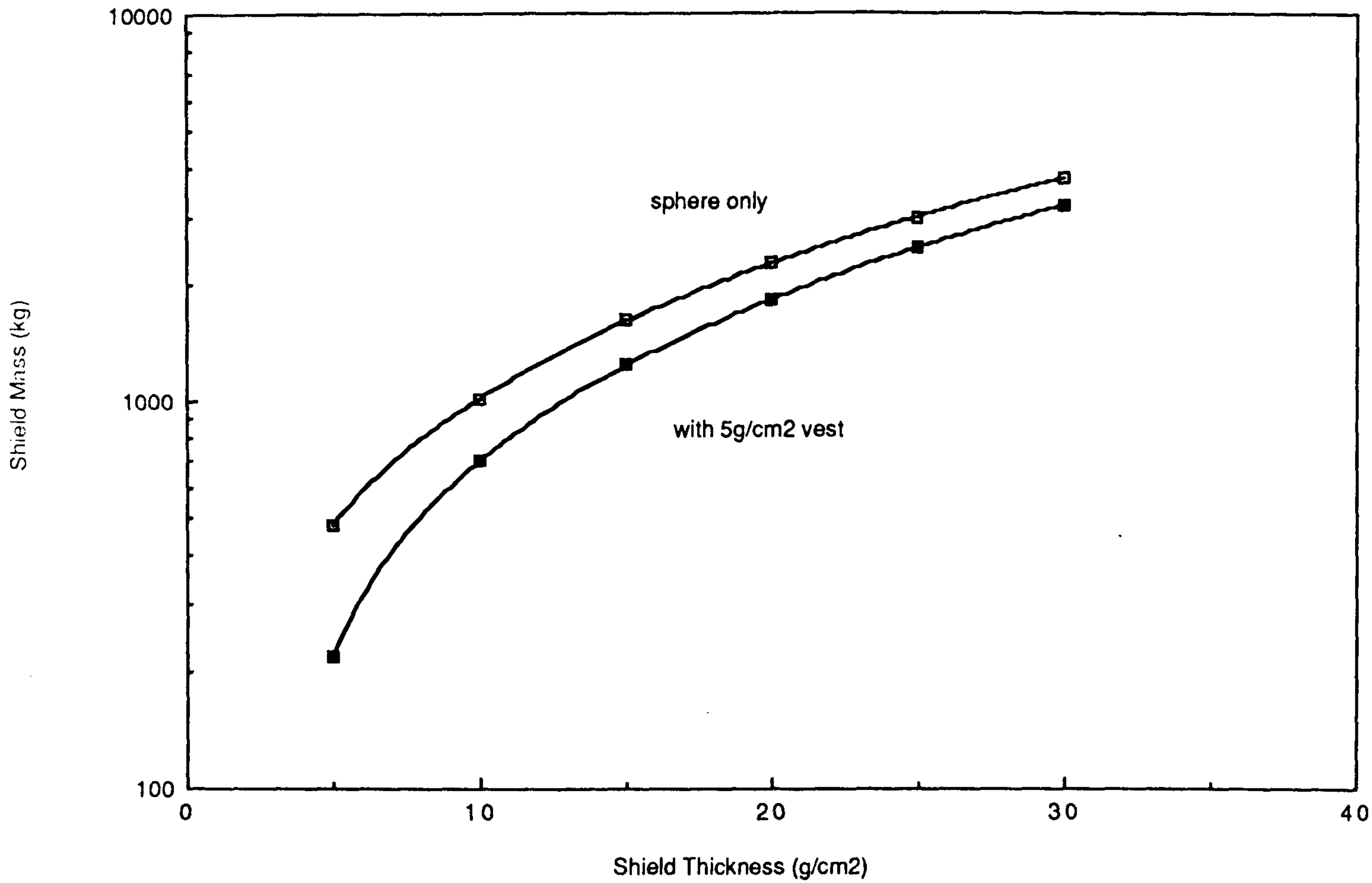


Table 5.4-2 Radiation Shelter for Crew of 6

Geometry	Mass* (kg) $20 g/cm^2$	Mass* (kg) $30 g/cm^2$
6 individual spheres	7,652	13,230
One spherical shell	4,934	7,955
Domed cylinder	2,930	4,852
Capped cylinder	2,417	4,114
Partial cylinder (knee-shoulder)	943	1,507
One person radiation vest**	314	566
Space Station module	43,536	65,305

* assumes $\rho=1.0$

**$5 g/cm^2=55$ kg, $10 g/cm^2=126$ kg

5.5 Communications Evaluation

The objective for communications during the astronauts' stay on Mars is to provide 100% lander-to-Earth communications. In order for this to be possible, one of three situations must exist. In the first the lander must be able to see either the MOV or the relay satellite at all times, and in the second the MOV and the relay satellite must both be able to see Earth at all times. Or lastly, the lander or the MOV must be able to see Earth at all times. One solution to the commumications problem involves placing the MOV and the relay satellite in mirror-symmetric, eccentric, areosynchronous orbit about Mars. Table 5.5-1 outlines the problem and solutions, and Figures 5.5-1 through 5.5-3 depict various communications satellites and MOV orbits. In addition to this, Figure 5.5-4 depict the lander to comsat visibility at different elevations above the horizon and Figure 5.5-5 shows the MOV to relay satellite visibility at different ranges.

Table 5.5-1 COMMUNICATIONS

OBJECTIVE:

Provide 100% Lander to Earth Communications

- Lander must be able to see either Mars Orbiting Vehicle (MOV) or Relay Satellite at all times.
- MOV and Relay Satellite must have constant mutual visibility.
- Lander or MOV must be able to see Earth 100% of time.

REQUIREMENTS

- Mars Lander located between 0° and 45° Latitude
- MOV in 45° inclination AreoSynchronous Orbit
  - circular and low inclination orbits discouraged because of high propellant cost
- Relay Comsat can be in any orbit
- Elevation Masking assumed to be 10°

SOLUTION

- MOV and Relay Satellite in eccentric, areosynchronous orbits of 45° inclination, with ascending nodes offset by 180°.

Figure 5.5-1 1-SOL ORBIT for COMMUNICATION SATELLITES (Plane

Semi-Major Axis:	20442 km
Eccentricity:	0.821842
Inclination:	45°
Apoapsis Altitude:	33840 km
Periapsis Altitude	250 km
Period:	24.66 hrs

SCALE

1 inch = 6409.8 km

12819.62 km

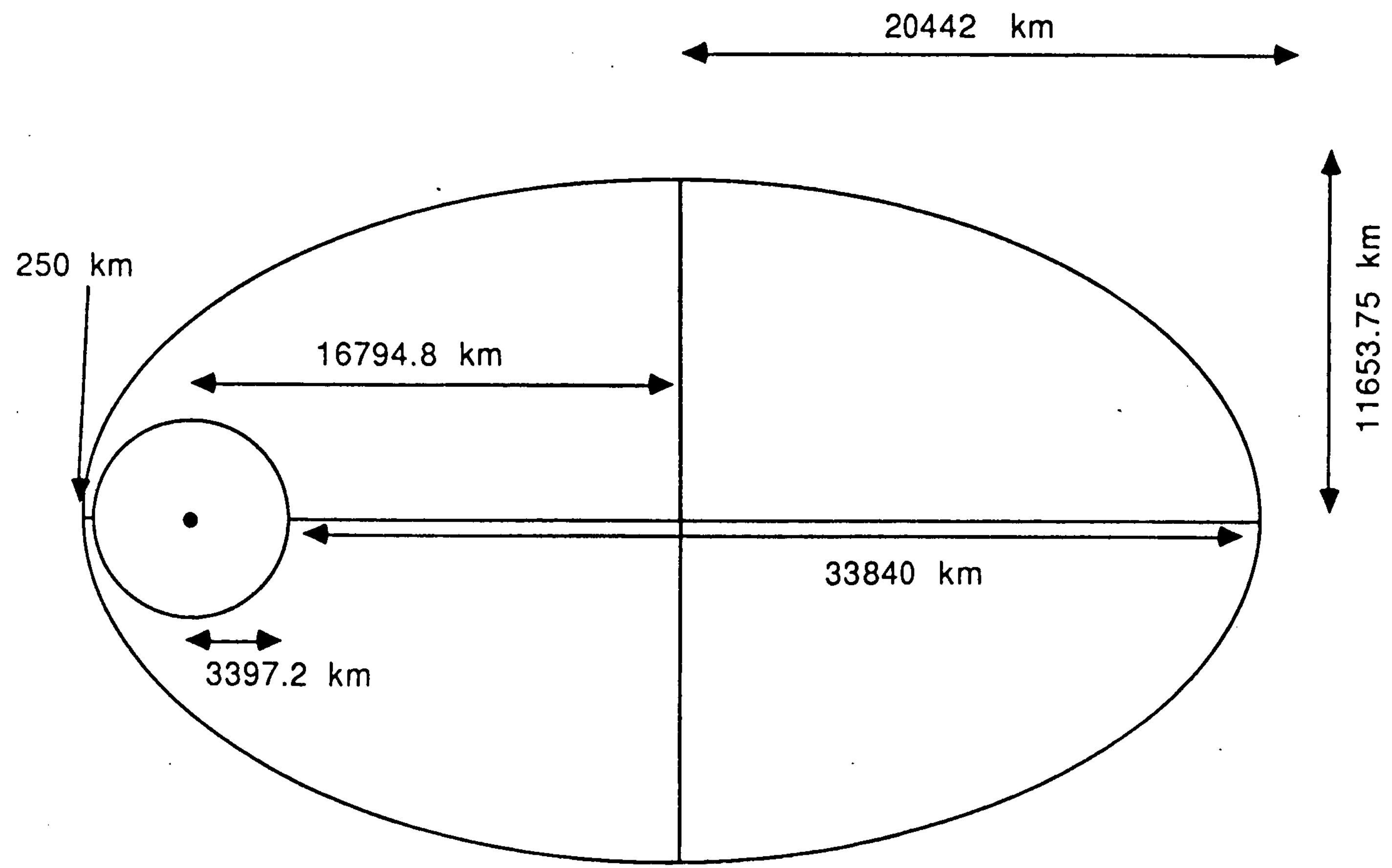


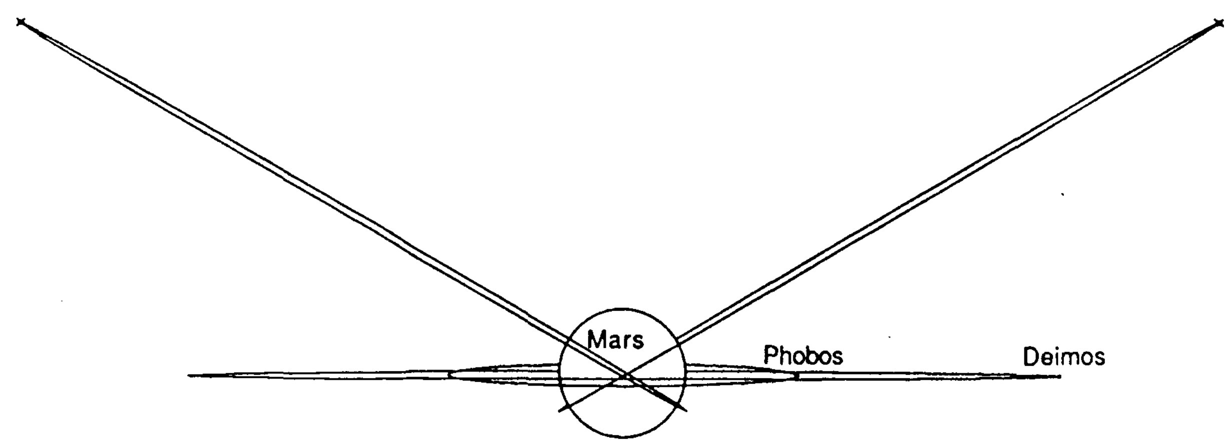
Figure 5.5-2 Comsat Orbits (side view)

Relay Satellite

Semi-Major Axis:	20442 km
Eccentricity:	0.821842
Inclination:	135 degrees
Long. of Ascending Node:	270 degrees
Argument of Periapsis:	270 degrees
True Anomaly at 0.00:	180 degrees
Period:	24.66 hours

MOV

Semi-Major Axis:	20442 km
Eccentricity:	0.821842
Inclination:	45 degrees
Long. of Ascending Node:	90 degrees
Argument of Periapsis:	270 degrees
True Anomaly at 0.00:	0 degrees
Period:	24.66 hours



Mars Diameter = 6,794 km
Phobos Altitude = 5,981 km
Deimos Altitude = 20.062 km

Figure 5.5-3 COMSAT ORBITS (TOP VIEW)

(Orbits projected onto MARS equatorial plane.)

MOV	
Semi-Major Axis:	20442 km
Eccentricity:	0.821842
Inclination:	45°
Long. of Ascending Node:	90°
Argument of Periapsis:	270°
True Anomaly at 0:00:	0°
Period:	24.66 hrs

Relay Satellite	
Semi-Major Axis:	20442 km
Eccentricity:	0.821842
Inclination:	45°
Long. of Ascending Node:	270°
Argument of Periapsis:	270°
True Anomaly at 0:00:	180°
Period:	24.66 hrs

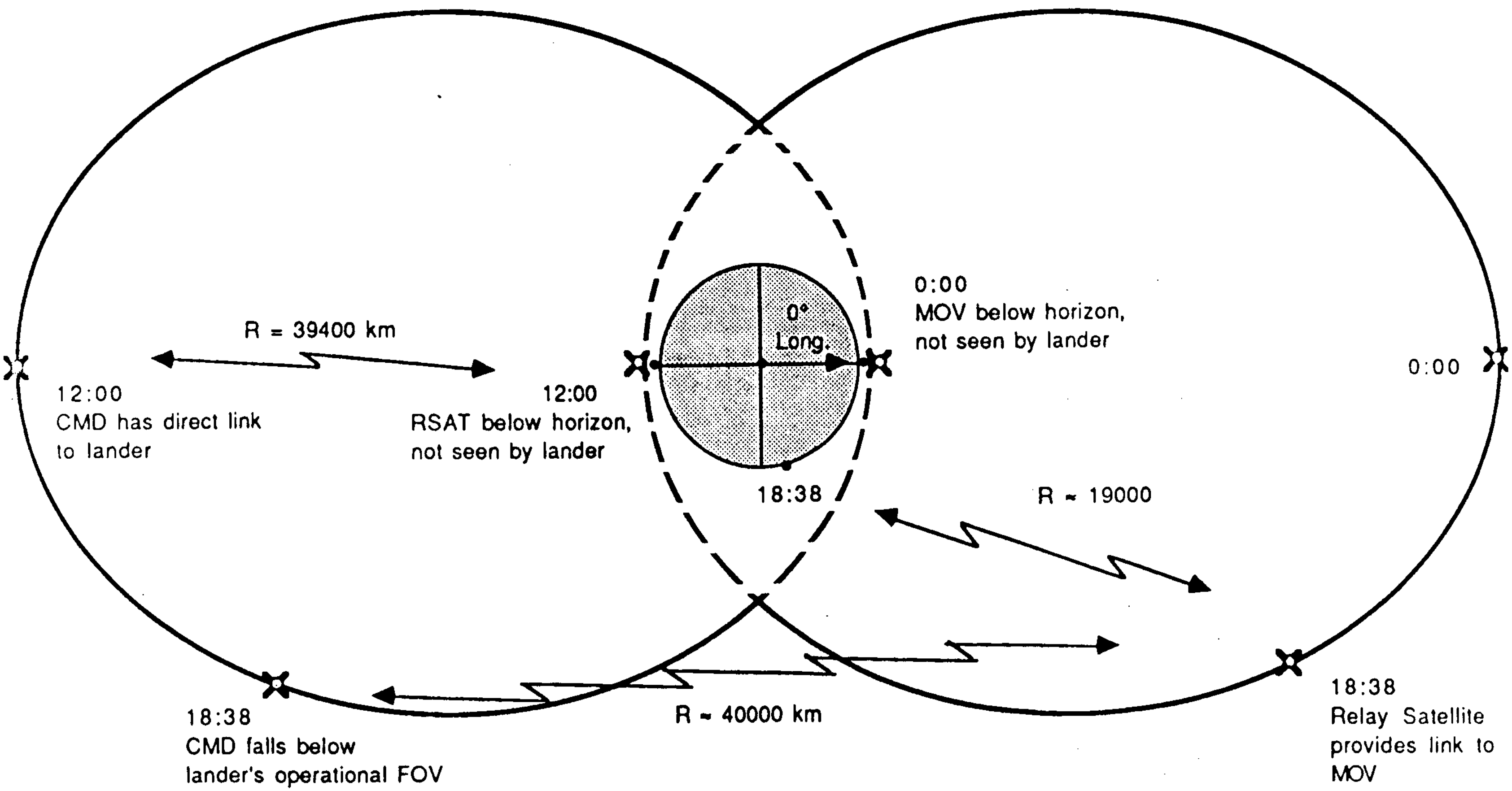


Figure 5.5-4 Lander to COMSAT Visibility

- Lander Latitude: 0°
- COMSAT Minimum Elevation : 10°

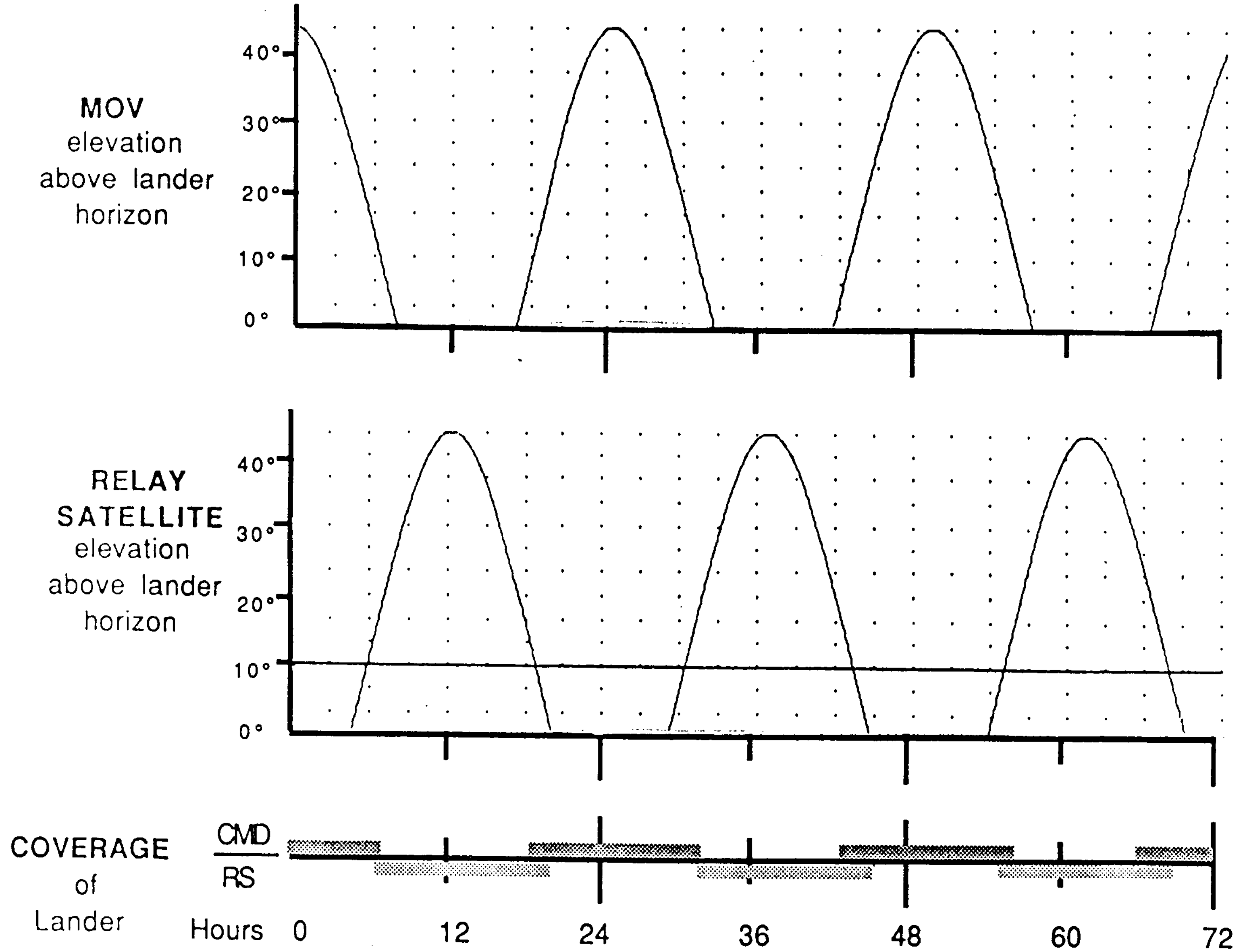
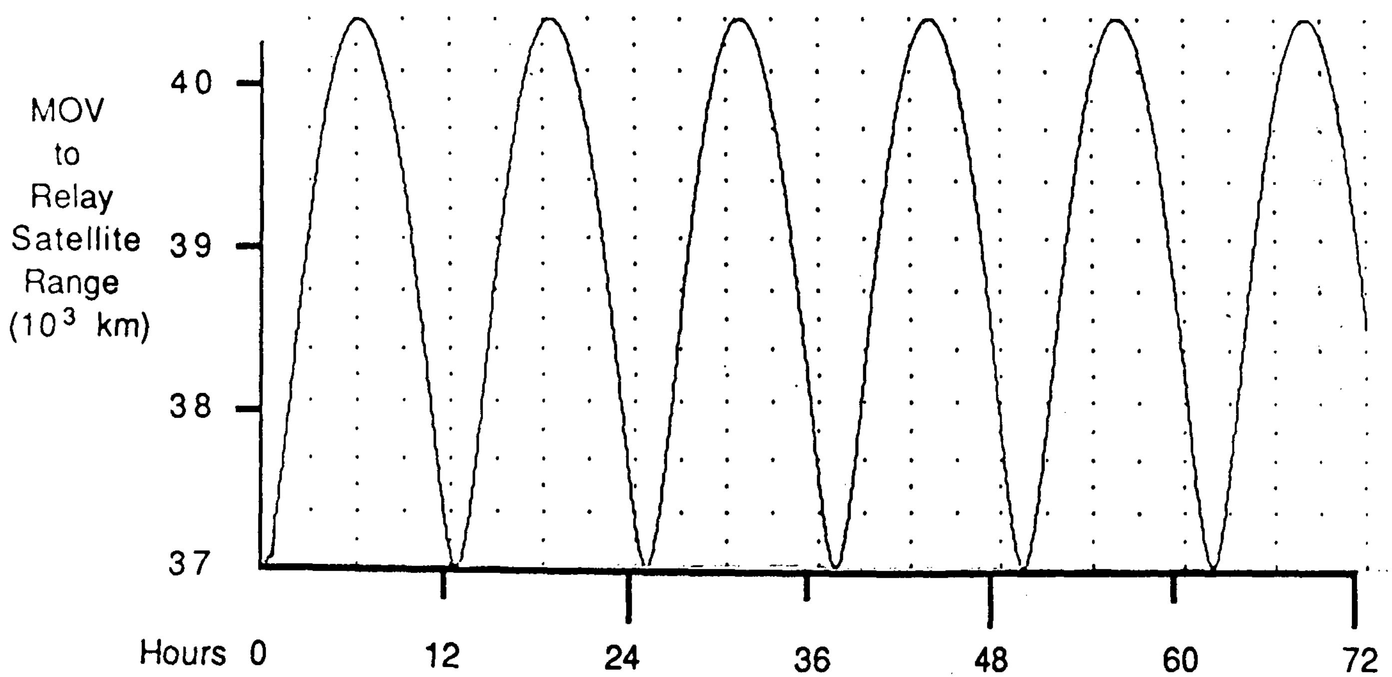


Figure 5.5-5 MOV - Relay Satellite Visibility

- Areosynchronous orbits provide continuous mutual visibility between MOV and Relay Satellite
- Range between MOV and Relay Satellite varies from 37040 km to 40400 km with a period of 12.33 hours.



5.6 Mass Allocations and Acronyms

This section includes the mass breakdowns not only for each mission, but also for each individual spacecraft and its major subsystems. Figure 5.6-1 schematically protrays vehicle nomenclature and Table 5.6-1 lists acronyms used in MMSS. Mass Allocations Summary Sheets (MASS) and Mission Mass Allocation Reports for each vehicle and mission of Case Study 1 and Case Study 2 follow.

Figure 5.6-1 Vehicle Nomenclature

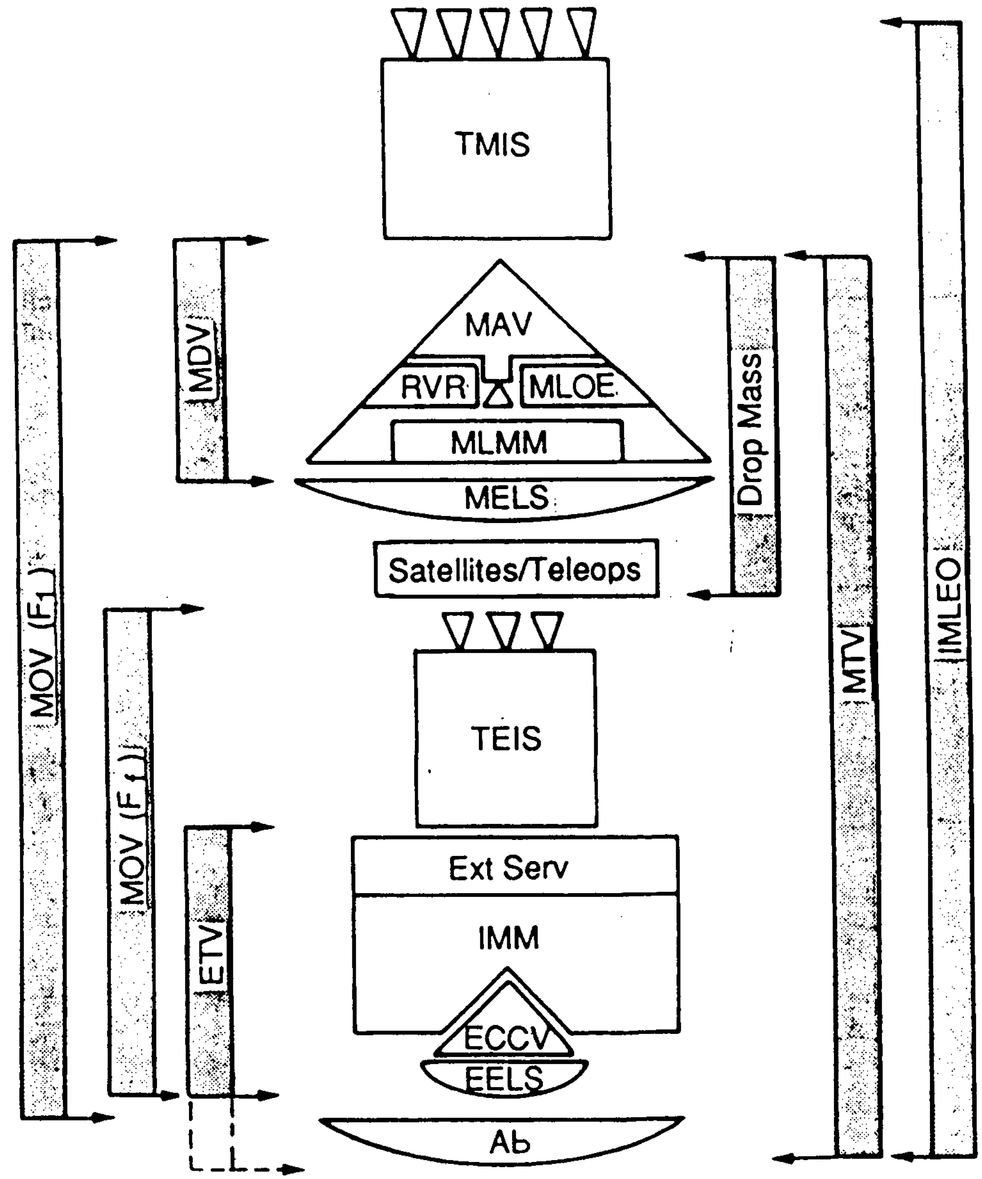


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Table 5.6-1 Acronyms, Abbreviations, and Definitions

Acronym	Definition	Comments
Ab	Aerobrake	
AC or A/C	Aerocapture	
CELSS	Controlled Ecological Life Support System	
CERV	Crew Emergency Return Vehicle	
ComSat	Communications satellite	
ComSciSat	Communication/Science satellite	
DeEV	Deimos Excursion Vehicle	
DSM	Deep Space Maneuver	(major interplanetary propulsive maneuver)
DSN	Deep Space Net	(NASA earthbased interplanetary com system)
ECCV	Earth Crew Capture Vehicle	(small vehicle for crew EOC)
ECLSS	Environmental Control and Life Support System	
EOC	Earth Orbital Capture	
EOCS	Earth Orbital Capture System	(Earth aerobrake plus retro-propulsion plus G&C)
EPS	Electrical Power System	
ET	External Tank	
ETO	Earth-to-Orbit	(vehicles such as STS, HLLV, etc.)
ETX	Earth Transfer Expendables	(propellant, other consumables during flight to Earth)
EVA	Extra-vehicular Activity	(any human activity outside protective shirtsleeve environment and requiring a spacesuit)
FTS	Flight Telerobotic Servicer	(teleoperated robot for SS)
gee	acceleration of gravity at the surface of the Earth	
GCR	Galactic Cosmic Rays	(cosmic rays, from outside the solar system)
HAL	Hyperbaric Airlock	(see AL)
Hc	Hydrocarbon	(propellant; methane (CH4) or other)
HLLV	Heavy-Lift Launch Vehicle	(SDVs and other advanced launchers)
IMLEO	Initial Mass in Low Earth Orbit	
IMM	Interplanetary Mission Modules	(Hab/Lab/Log modules for crew in space)
I_{sp}	Specific impulse	(units of Ns/kg or lb_f-s/lb_m)
ISRU	*in situ* Resources Utilization	
ISXP	*in situ* X Production	(e.g., X is: P=propellant, W=water, F=food, R=resources, C=consumables)
LEO	Low Earth Orbit	
LH2	Liquid hydrogen	
LLO	Low Lunar Orbit	
LMO	Low Mars Orbit	
LSS	Life Support System	
m	meter	(Note: "m" as a prefix indicates "milli")
MAV	Mars Ascent Vehicle	(the vehicle which is launched to Mars orbit)
MCV	Mars Cargo Vehicle	(logistics vehicle sent for cargo staging)
MDV	Mars Descent Vehicle	(the vehicle which de-orbits to land on Mars)
MELS	Mars Entry & Landing System	(de-orbit propulsion + aerobrake + parachute + terminal propulsion + G & C)
MLI	Multi-layer Insulation	
MLMM	Mars Landed Mission Module(s)	(Hab/Lab/Log modules for the surface of Mars)
MLOE	Mars Landed Operations Equipment	(Science, Transportation, Construction, Manufacturing equipment -- substitute S, T, C, M for O)
MMH	Monomethyl hydrazine	(propellant)
MMU	Manned Maneuvering Unit	

Table 5.6-1 Acronyms, Abbreviations, and Definitions (cont.)

MO	Mars Observer	(polar orbiter mission to Mars, planned for 1992 launc!
MO	Mars Orbit	
MOC	Mars Orbital Capture	
MOCS	Mars Orbital Capture System	(Mars aerobrake + retro-propulsion, if required
G&C)		
MOO	Mars Orbital Operations	
MOOS	Mars Orbital Operations System	(propulsion for Mars orbital maneuvers)
MOSE	Mars Orbit Science Equipment	(Instruments for studies from Mars orbit)
MOV	Mars Orbiting Vehicle	(MSS configuration in Mars orbit)
MRSR	Mars Rover Sample Return	(combined rover and sample return mission)
MSS	Mars Spaceship	(the spaceship that is assembled in LEO;
MTV	Mars Transfer Vehicle	(MSS configuration during flight to Mars)
NEP	Nuclear Electric Propulsion	(ion drive; nuclear reactor)
NERVA	Nuclear Engine for Rocket Vehicle Application	(nuclear thermal rocket program)
NTO	Nitrogen tetroxide	(N_2O_4, biprop oxidizer)
NTR	Nuclear Thermal Rocket	
OMV	Orbital Maneuvering Vehicle	
OTV	Orbital Transfer Vehicle	
Ph-Tele	Phobos Teleoperator	(remotely operated free-flyer to Phobos)
PhSE	Phobos Science Equipment	(instruments for studies of Phobos from a PhEM)
PhEV	Phobos Excursion Vehicle	(manned vehicle for transportation to Phobos)
P/L	payload	(means different thing to different people)
PVPA	Photovoltaic Power Array	(solar cells)
RCS	Reaction Control System	
RL-10	(LH2/LOX engine, mfg. by Pratt & Whitney)	
RMS	Remote Manipulator System	(Shuttle robot arm)
RTG	Radioisotope Thermoelectric Generator	
RVR	Rover	
SEP	Solar Electric Propulsion	(ion drive; solar power)
SI	International System of Units	
SS	Space Station	(phase 1)
SSME	Space Shuttle Main Engine	
STS	Space Transportation System	(Shuttle)
t	metric ton	(tonne, 1000 kg, or 1 Mg)
TCS	Thermal Control System	
TEI	Trans-Earth Injection	(Mars orbital escape and trans-Earth maneuver)
TEIS	Trans-Earth Injection System	(propulsion and guidance system for TEI)
TMI	Trans-Mars Injection	(Earth orbital escape and trans-Mars maneuver)
TMIS	Trans-Mars Injection System	(propulsion and guidance system for TMI)
TPS	Thermal Protection System	
VGRF	Variable Gravity Research Facility	(proposed for SS)

Table 5.6-1 Hierarchical Summary of Vehicle and Facility Acronyms

MSS	Mars Spaceship	(the spaceship that is assembled in LEO)
TMIS	Trans-Mars Injection System	(propulsion and guidance system for TMI)
MTV	Mars Transfer Vehicle	(configuration during flight to Mars)
IMM	Interplanetary Mission Modules	(Hab/Lab/Log modules for crew in space)
MOCS	Mars Orbital Capture System	(Mars aerobrake+retro-propulsion+ G&C)
MCV	Mars Cargo Vehicle	(logistics vehicle sent for cargo staging)
MDV	Mars Descent Vehicle	(the vehicle which de-orbits to land)
MAV	Mars Ascent Vehicle	(the vehicle which is launched to Mars orbit)
MELS	Mars Entry & Landing System	(de-orbit propulsion + aerobrake + parachute + terminal propulsion + G & C)
MLMM	Mars Landed Mission Module(s)	(Hab/Lab/Log modules)
MLOE	Mars Landed Operations Equipment	(Science, Transportation, Construction, Manufacturing substitute S, T, C, M for O)
RVR	Rover	
MOV	Mars Orbiting Vehicle	(configuration in Mars orbit, not incl. the MDVs)
TEIS	Trans-Earth Injection System	(propulsion and guidance system for TEI)
ETV	Earth Transfer Vehicle	(config. of the MSS for Mars to Earth transfer)
MTM	Mars Transfer Modules	(Hab/Lab/Log modules for crew in space)
EOCS	Earth Orbital Capture System	(Earth aerobrake + retro-propulsion, if required)
ECCV	Earth Crew Capture Vehicle	(small vehicle for crew EOC and/or EELS)
EELS	Earth Entry & Landing System	(see MELS subsystems)

Mass Allocations Summary Sheet (MASS) file name: LTV. Scen #3.Cargo

Case Study #3 - Lunar Transfer Vehicle
(with LDV-C Payload)

Basis: E = Estimated (contingency = 10%)
C = Calculated from Drawing (contingency = 5%)
S = Manufacturer's Spec. (contingency = 1%)
A = Actual Mass (contingency = 1%)

Rev A 06/16/88
Rev B 06/30/88
Rev C 07/05/88

Item Name	Basis	Rev	Mass, ea (kg)	Qty.	Mass Totals (kg)		
A. Structure							1636
1. Framework	E	B	1636			1636	
B. Equipment							460
1. Orientation Control	E	A	120			120	
2. Electric Systems	E	A	240			240	
3. G N & C	E	A	100			100	
C. Propulsion							86766
1. Engine	E	B	191	4		764	
2. Fuel Tanks	E	C	8086			8086	
3. Propellant	E	C	77916			77916	
D. Aerobrake							2298
1. TPS						1252	
a. RSI	E	A	160		160		
b. FSI	E	A	1092		1092		
2. Structure						1046	
a. RSI Honeycomb Substrate	E	A	78		78		
b. Interface Ring	E	A	264		264		
c. Radial Beams	E	A	152		152		
d. Support Struts	E	A	282		282		
e. Attach Hardware	E	A	270		270		
E. Payload							34287
1. Cargo Vehicle	E	A	34287			34287	

Total Mass (kg) = 125447

Mass Allocations Summary Sheet (MASS) file name: LTV. Scen #3.Cargo.2

Case Study #3 - Lunar Transfer Vehicle
(with LDV-C Payload; no Earth return)

Basis: E = Estimated (contingency = 10%)
C = Calculated from Drawing (contingency = 5%)
S = Manufacturer's Spec. (contingency = 1%)
A = Actual Mass (contingency = 1%)

Rev A 06/16/88
Rev B 06/30/88
Rev C 07/12/88

Item Name	Basis	Rev	Mass, ea (kg)	Qty.	Mass Totals (kg)		
A. Structure							1636
1. Framework	E	B	1636			1636	
B. Equipment							460
1. Orientation Control	E	A	120			120	
2. Electric Systems	E	A	240			240	
3. G N & C	E	A	100			100	
C. Propulsion							78481
1. Engine	E	B	191	4		764	
2. Fuel Tanks	E	C	8086			8086	
3. Propellant	E	C	69631			69631	
D. Aerobrake							2298
1. TPS						1252	
a. RSI	E	A	160		160		
b. FSI	E	A	1092		1092		
2. Structure						1046	
a. RSI Honeycomb Substrate	E	A	78		78		
b. Interface Ring	E	A	264		264		
c. Radial Beams	E	A	152		152		
d. Support Struts	E	A	282		282		
e. Attach Hardware	E	A	270		270		
E. Payload							34287
1. Cargo Vehicle	E	A	34287			34287	

Total Mass (kg) = 117162

Mass Allocations Summary Sheet (MASS) file name: LTV. Scen #3.Piloted

Case Study #3 - Lunar Transfer Vehicle
(with LDV-P Payload)

Basis: E = Estimated (contingency = 10%)
C = Calculated from Drawing (contingency = 5%)
S = Manufacturer's Spec. (contingency = 1%)
A = Actual Mass (contingency = 1%)

Rev A 06/16/88
Rev B 06/30/88
Rev C 07/05/88

Item Name	Basis	Rev	Mass, ea (kg)	Qty.	Mass Totals (kg)		
A. Structure							1636
1. Framework	E	B	1636			1636	
B. Equipment							460
1. Orientation Control	E	A	120			120	
2. Electric Systems	E	A	240			240	
3. G N & C	E	A	100			100	
C. Propulsion							89709
1. Engine	E	B	191	4		764	
2. Fuel Tanks	E	C	8086			8086	
3. Propellant	E	C	80859			80859	
D. Aerobrake							2298
1. TPS						1252	
a. RSI	E	A	160		160		
b. FSI	E	A	1092		1092		
2. Structure						1046	
a. RSI Honeycomb Substrate	E	A	78		78		
b. Interface Ring	E	A	264		264		
c. Radial Beams	E	A	152		152		
d. Support Struts	E	A	282		282		
e. Attach Hardware	E	A	270		270		
E. Payload							34840
1. Piloted Vehicle	E	C	34840			34840	

Total Mass (kg) = 128943

Mass Allocations Summary Sheet (MASS) file name: Lunar Cargo. Scen #3

Case Study #3 - Lunar Descent Vehicle - Cargo (LDV-C) (Descent only)

Basis: E = Estimated (contingency = 10%)
C = Calculated from Drawing (contingency = 5%)
S = Manufacturer's Spec. (contingency = 1%)
A = Actual Mass (contingency = 1%)

Rev A 06/16/88
Rev B 06/30/88

Item Name	Basis	Rev	Mass, ea (kg)	Qty.	Mass Totals (kg)	
A. Payload						17500
1. User Payload	E	A	17500		17500	
B. Structure						950
1. Cargo Bay	E	A	500		500	
2. Misc. Structure	E	B	450		450	
C. Propulsion; Descent						14809
1. Engine	E	B	191	4	764	
2. Fuel Tanks	E	A	1277		1277	
3. Propellant	E	A	12768		12768	
D. Equipment						464
1. G N & C	E	A	100		100	
2. Electrical Systems	E	B	244		244	
3. Orientation Control	E	A	120		120	
E. Landing Gear						564
1. Landing Pads	E	B	141	4	564	
					Total Mass (kg) =	34287

Mass Allocations Summary Sheet (MASS) file name: Lunar Piloted. Scen #3

Case Study #3 - Lunar Descent Vehicle - Piloted (LDV-P)
(14 day stay; 55 day contingency)

Basis: E = Estimated (contingency = 10%)
C = Calculated from Drawing (contingency = 5%)
S = Manufacturer's Spec. (contingency = 1%)
A = Actual Mass (contingency = 1%)

Rev A 06/16/88
Rev B 06/30/88
Rev C 07/05/88

Item Name	Basis	Rev	Mass, ea (kg)	Qty.		Mass Totals (kg)	
A. LAV							5430
1. LAV Dry Weight	E	C	3206			3206	
2. LAV Prop. and Tanks	E	C	2224			2224	
B. Payload							6500
1. User Payload	E	A	6500			6500	
C. Hab. Module							4489
1. LSS						1114	
a. Oxygen	E	A	1.00	288	288		
b. Water (byproduct of Fuel Cells)	E	A	0.00		0		
c. Food	E	A	2.00	288	576		
d. Air Purification	E	A	50		50		
e. Waste Management	E	A	50		50		
f. Thermal Control	E	A	150		150		
2. Structure						1700	
a. Outer Shell	E	A	1200		1200		
b. Insulation	E	A	300		300		
c. Internal Walls and Supports	E	A	200		200		
3. Power System						225	
a. Batteries	E	A	125		125		
b. Fuel Cells	E	A	100		100		
4. Medical Equipment	E	A	250			250	
5. Science Equipment	E	A	250			250	
6. Airlock	E	A	750			750	
7. Misc. Crew Systems	E	A	200			200	
D. Structure							1064
1. Landing Pads	E	B	141	4		564	
2. Misc. Structure	E	A	500			500	
E. Propulsion							17357
1. Propulsion; Descent						15035	
a. Engine	E	B	191	4	764		
b. Fuel Tanks	E	C	1297		1297		
c. Propellant (LOX/LH2)	E	C	12974		12974		
2. Fuel Cells (5 KW)	E	B	2122			2122	
3. Fueled ACS	E	A	200			200	

Total Mass (kg) = 34840

Mass Allocations Summary Sheet (MASS) file name: LAV. Scen #3

Case Study # 3 -Lunar Ascent Vehicle (LAV)
(LLO Rendezvous)

Basis: E = Estimated (contingency = 10%)
C = Calculated from Drawing (contingency = 5%)
S = Manufacturer's Spec. (contingency = 1%)
A = Actual Mass (contingency = 1%)

Rev A 06/16/88
Rev B 06/30/88
Rev C 07/05/88

Item Name	Basis	Rev	Mass, ea (kg)	Qty.	Mass Totals (kg)		
A. Payload							460
1. Crew	E	A	75	4		300	
2. Spacesuit	E	A	20	4		80	
3. Couch	E	A	10	4		40	
4. Inerts	E	C	40			40	
B. LSS							1168
1. Oxygen	E	A	1.00	16		16	
2. Water (byproduct of Fuel Cells)	E	A	0.00			0	
3. Food	E	A	2.00	16		32	
4. Air Purification	E	A	30			30	
5. Thermal Control	E	A	100			100	
6. Waste Management	E	A	30			30	
7. Radiation Vests	E	C	240	4		960	
C. Structure							410
1. Outer Shell	E	A	300			300	
2. Insulation	E	A	50			50	
3. Window	E	A	10	2		20	
4. Hatch	E	A	20			20	
5. Panels and Supports	E	A	20			20	
D. Propulsion							2947
1. Ascent						2659	
a. Engine	E	B	145	3	435		
b. Fuel Tanks	E	C	202		202		
c. Propellant (LOX/LH2)	E	C	2022		2022		
2. Fuel Cells (2 KW)	E	A	88			88	
3. Fueled ACS	E	A	200			200	
E. Power							175
1. Batteries	E	A	125			125	
2. Fuel Cells	E	A	50			50	
F. Equipment							270
1. Controls and Displays	E	A	60			60	
2. Communication	E	A	50			50	
3. Guidance and Navigation	E	A	100			100	
4. Docking Provisions	E	A	60			60	

Total Mass (kg) = 5430

Mass Allocations Summary Sheet (MASS)   file name: Lunar Cargo. Scen #3.direct

Case Study #3 - Direct Alternative; LDV-C
(Direct Lunar descent)

Basis: E = Estimated (contingency = 10%)
C = Calculated from Drawing (contingency = 5%)
S = Manufacturer's Spec. (contingency = 1%)
A = Actual Mass (contingency = 1%)

Rev A 06/16/88
Rev B 06/30/88

Item Name	Basis	Rev	Mass, ea (kg)	Qty.	Mass Totals (kg)	
A. Payload						17500
1. User Payload	E	A	17500		17500	
B. Structure						950
1. Cargo Bay	E	A	500		500	
2. Misc. Structure	E	B	450		450	
C. Propulsion; Descent						76902
1. Engine	E	B	191	4	764	
2. Fuel Tanks	E	B	6922		6922	
3. Propellant	E	B	69216		69216	
D. Equipment						464
1. G N & C	E	A	100		100	
2. Electrical Systems	E	B	244		244	
3. Orientation Control	E	A	120		120	
E. Landing Gear						564
1. Landing Pads	E	B	141	4	564	
					Total Mass (kg) =	96380

Mass Allocations Summary Sheet (MASS)   file name: Lunar Piloted. Scen #3.direct

Case Study #3 - Direct Alternative; LDV-P
(Direct Lunar descent; 14 day stay; 55 day contingency)

Basis:		
	E = Estimated (contingency = 10%)	Rev A 06/16/88
	C = Calculated from Drawing (contingency = 5%)	Rev B 06/30/88
	S = Manufacturer's Spec. (contingency = 1%)	Rev C 07/05/88
	A = Actual Mass (contingency = 1%)	

Item Name	Basis	Rev	Mass, ea (kg)	Qty.	Mass Totals (kg)		
A. LAV							8034
1. LAV Dry Weight	E	B	4206			4206	
2. LAV Prop. and Tanks	E	B	3828			3828	
B. Payload							6500
1. User Payload	E	A	6500			6500	
C. Hab. Module							5489
1. LSS						1114	
a. Oxygen	E	A	1.00	288	288		
b. Water (byproduct of Fuel Cells)	E	A	0.00		0		
c. Food	E	A	2.00	288	576		
d. Air Purification	E	A	50		50		
e. Waste Management	E	A	50		50		
f. Thermal Control	E	A	150		150		
2. Structure						1700	
a. Outer Shell	E	A	1200		1200		
b. Insulation	E	A	300		300		
c. Internal Walls and Supports	E	A	200		200		
3. Power System						225	
a. Batteries	E	A	125		125		
b. Fuel Cells	E	A	100		100		
4. Medical Equipment	E	A	250			250	
5. Science Equipment	E	A	250			250	
6. Airlock	E	A	750			750	
7. Storm Shelter	E	A	1000			1000	
8. Misc. Crew Systems	E	A	200			200	
D. Structure							1064
1. Landing Pads	E	B	141	4		564	
2. Misc. Structure	E	A	500			500	
E. Propulsion							94009
1. Propulsion; Descent &TLI						91687	
a. Engine	E	B	191	4	764		
b. Fuel Tanks	E	C	8266		8266		
c. Propellant (LOX/LH2)	E	C	82657		82657		
2. Fuel Cells (5 KW)	E	B	2122			2122	
3. Fueled ACS	E	A	200			200	

Total Mass (kg) = 115096

Mass Allocations Summary Sheet (MASS) file name: LAV. Scen #3.direct

Case Study # 3 - Direct Alternative; LAV (Direct Earth entry)

Basis: E = Estimated (contingency = 10%)
C = Calculated from Drawing (contingency = 5%)
S = Manufacturer's Spec. (contingency = 1%)
A = Actual Mass (contingency = 1%)

Rev A 06/16/88
Rev B 06/30/88
Rev C 07/05/88

Item Name	Basis	Rev	Mass, ea (kg)	Qty.		Mass Totals (kg)	
A. Payload							460
1. Crew	E	A	75	4		300	
2. Spacesuit	E	A	20	4		80	
3. Couch	E	A	10	4		40	
4. Inerts	E	A	40			40	
B. LSS							1168
1. Oxygen	E	A	1.00	16		16	
2. Water (byproduct of Fuel Cells)	E	A	0.00			0	
3. Food	E	A	2.00	16		32	
4. Air Purification	E	A	30			30	
5. Thermal Control	E	A	100			100	
6. Waste Management	E	A	30			30	
7. Radiation Vests	E	C	240	4		960	
C. Structure							410
1. Outer Shell	E	A	300			300	
2. Insulation	E	A	50			50	
3. Window	E	A	10	2		20	
4. Hatch	E	A	20			20	
5. Panels and Supports	E	A	20			20	
D. Propulsion							4551
1. Ascent						4263	
a. Engine	E	B	145	3	435		
b. Fuel Tanks	E	C	348		348		
c. Propellant (LOX/LH2)	E	C	3480		3480		
2. Fuel Cells (2 KW)	E	A	88			88	
3. Fueled ACS	E	A	200			200	
E. Power							175
1. Batteries	E	A	125			125	
2. Fuel Cells	E	A	50			50	
F. Equipment							270
1. Controls and Displays	E	A	60			60	
2. Communication	E	A	50			50	
3. Guidance and Navigation	E	A	100			100	
4. Docking Provisions	E	A	60			60	
G. Thermal Protection System							800
1. Heat Shield	E	B	800			800	
H. Parachute							200
1. Parachute System	E	B	200			200	

Total Mass (kg) = 8034

Mass Allocations Summary Sheet (MASS) file name: MLL. Scen #4

Case Study #4 - Mars Logistic Lander (MLL)
Descent from Phobos Circular

Basis: E = Estimated (contingency = 10%) Rev A 07/09/88
C = Calculated from Drawing (contingency = 5%)
S = Manufacturer's Spec. (contingency = 1%)
A = Actual Mass (contingency = 1%)

Item Name	Basis	Rev	Mass, ea (kg)	Qty.		Mass Totals (kg)	
A. MLSE							8320
1. Rover	E	A	1020			1020	
2. Teleoperators	E	A	3000			3000	
3. ISRU demo's	E	A	2000			2000	
4. Misc. Science Equipment	E	A	2300			2300	
B. Hab. Module							40789
1. Crew	E	A	75	8		600	
2. Structure						8700	
a. Outer Shell	E	A	6000		6000		
b. Insulation & TCS	E	A	1400		1400		
c. Internal Walls and Supports	E	A	1300		1300		
3. ECLSS						1825	
a. Atm. Revitalization	E	A	545		545		
b. Atm. Control and Supply	E	A	300		300		
c. Fire Detection and Suppression	E	A	150		150		
d. Temp. and Humidity Control	E	A	300		300		
e. Water Recovery and Management	E	A	400		400		
f. Waste Management	E	A	130		130		
4. Power System (10 KW)	E	A	3640			3640	
5. Data & Communications	E	A	450			450	
6. Consumables						23624	
a. Solids	E	A	1.42	5762	8182		
b. Food (Dry)	E	A	0.93	5762	5359		
c. Nitrogen Leakage	E	A	0.45	5762	2593		
d. Liquids	E	A	0.17	5762	980		
e. Oxygen Leakage	E	A	0.12	5762	691		
f. Water	E	A	1.01	5762	5820		
7. Medical Equipment	E	A	250			250	
8. Airlock	E	A	700			700	
9. Misc. Crew Systems	E	A	1000			1000	
C. Structure							2700
1. Truss	E	A	1500			1500	
2. Landing Pads	E	A	200	6		1200	
D. Propulsion							5935
1. Engines	E	A	100	6		600	
2. Fuel Tanks	E	A	485			485	
3. Propellant	E	A	4850			4850	
E. Parachute							4000
1. Parachute	E	A	4000			4000	
F. Aerobrake							4080
1. Aerobrake	E	A	4080			4080	

Total Mass (kg) = 65824

Mass Allocations Summary Sheet (MASS) file name: P-OTV. Scen #4

Case Study #4 - Piloted Orbital Transfer Vehicle (P-OTV)
Aerobrake at Earth

Basis: E = Estimated (contingency = 10%) Rev A 07/09/88
C = Calculated from Drawing (contingency = 5%)
S = Manufacturer's Spec. (contingency = 1%)
A = Actual Mass (contingency = 1%)

Item Name	Basis	Rev	Mass, ea (kg)	Qty.	Mass Totals (kg)		
A. Structure							1636
1. Framework	E	A	1636			1636	
B. Equipment							460
1. Orientation Control	E	A	120			120	
2. Electric Systems	E	A	240			240	
3. G N & C	E	A	100			100	
C. Propulsion							23699
1. Engine	E	A	191	4		764	
2. Fuel Tanks	E	A	2085			2085	
3. Propellant	E	A	20850			20850	
D. Aerobrake							2298
1. TPS						1252	
a. RSI	E	A	160		160		
b. FSI	E	A	1092		1092		
2. Structure						1046	
a. RSI Honeycomb Substrate	E	A	78		78		
b. Interface Ring	E	A	264		264		
c. Radial Beams	E	A	152		152		
d. Support Struts	E	A	282		282		
e. Attach Hardware	E	A	270		270		
E. Hab Module							2729
1. Payload	E	A	470			470	
2. LSS	E	A	1228			1228	
3. Structure	E	A	410			410	
4. Power	E	A	351			351	
5. Equipment	E	A	270			270	

Total Mass (kg) = 30822

Mass Allocations Summary Sheet (MASS) file name: LPL. Scen #4

Case Study # 4 - Lunar Piloted Lander (LPL)
(Ascent & Descent from LLO)

Basis: E = Estimated (contingency = 10%) Rev A 07/09/88
C = Calculated from Drawing (contingency = 5%)
S = Manufacturer's Spec. (contingency = 1%)
A = Actual Mass (contingency = 1%)

Item Name	Basis	Rev	Mass, ea (kg)	Qty.	Mass Totals (kg)		
A. Payload							470
1. Crew	E	A	75	4		300	
2. Spacesuit	E	A	20	4		80	
3. Couch	E	A	10	4		40	
4. Inerts	E	A	50			50	
B. LSS							184
1. Oxygen	E	A	1.00	8		8	
2. Water (byproduct of Fuel Cells)	E	A	0.00			0	
3. Food	E	A	2.00	8		16	
4. Air Purification	E	A	30			30	
5. Thermal Control	E	A	100			100	
6. Waste Management	E	A	30			30	
C. Structure							1174
1. Outer Shell	E	A	300			300	
2. Insulation	E	A	50			50	
3. Window	E	A	10	2		20	
4. Hatch	E	A	20			20	
5. Panels and Supports	E	A	20			20	
6. Landing Legs	E	A	141	4		564	
7. Misc. Structure	E	A	200			200	
D. Propulsion							6585
1. Ascent						6341	
a. Engine	E	A	145	3	435		
b. Fuel Tanks	E	A	537		537		
c. Propellant (LOX/LH2)	E	A	5369		5369		
2. Fuel Cells (2 KW)	E	A	44			44	
3. Fueled ACS	E	A	200			200	
E. Power							175
1. Batteries	E	A	125			125	
2. Fuel Cells	E	A	50			50	
F. Equipment							270
1. Controls and Displays	E	A	60			60	
2. Communication	E	A	50			50	
3. Guidance and Navigation	E	A	100			100	
4. Docking Provisions	E	A	60			60	
					Total Mass (kg) =		8858

Mass Allocations Summary Sheet (MASS) file name: LCL. Scen #4

Case Study # 4 - Lunar Cargo Lander (LCL)
(Ascent & Descent from LLO)

Basis: E = Estimated (contingency = 10%) Rev A 07/09/88
C = Calculated from Drawing (contingency = 5%)
S = Manufacturer's Spec. (contingency = 1%)
A = Actual Mass (contingency = 1%)

Item Name	Basis	Rev	Mass, ea (kg)	Qty.		Mass Totals (kg)	
A. Payload							80000
1. LLOX (to LLO)	E	A	80000			80000	
2. Cargo (from LLO to LS; 40000 kg)			0.00			0	
B. Structure							1564
1. Cargo Bay	E	A	500			500	
2. Landing Legs	E	A	141	4		564	
3. Misc. Structure	E	A	500			500	
C. Propulsion							118721
1. Ascent & descent						118521	
a. Engine	E	A	600	4	2400		
b. Propellant (LOX/LH2)	E	A	106782		106782		
c. Payload LLOX Tank	E	A	4000		4000		
d. Ascent & Descent Tanks	E	A	5339		5339		
2. Fueled ACS	E	A	200			200	
D. Equipment							464
1. G N & C	E	A	100			100	
2. Electrical Systems	E	A	244			244	
3. Orientation Control	E	A	120			120	

Total Mass (kg) = 200749

Mass Allocations Summary Sheet (MASS)   file name: MPEM. Scen #4

Case Study # 4 - Mars Phobos Excursion Module (MPEM)
Descent and Ascent from Phobos Circular

Basis: E = Estimated (contingency = 10%)   Rev A 07/10/88
C = Calculated from Drawing (contingency = 5%)
S = Manufacturer's Spec. (contingency = 1%)
A = Actual Mass (contingency = 1%)

Item Name	Basis	Rev	Mass, ea (kg)	Qty.		Mass Totals (kg)	
A. Payload							470
1. Crew	E	A	75	4		300	
2. Spacesuit	E	A	20	4		80	
3. Couch	E	A	10	4		40	
4. Inerts	E	A	50			50	
B. LSS							176
1. Oxygen	E	A	1.00	4		4	
2. Water	E	A	4.00			4	
3. Food	E	A	2.00	4		8	
4. Air Purification	E	A	30			30	
5. Thermal Control	E	A	100			100	
6. Waste Management	E	A	30			30	
C. Structure							1256
1. Outer Shell	E	A	300			300	
2. Insulation	E	A	50			50	
3. Window	E	A	10	2		20	
4. Hatch	E	A	20			20	
5. Panels and Supports	E	A	20			20	
6. Landing Legs	E	A	141	4		564	
7. Misc. Structure for Legs	E	A	282			282	
D. Propulsion							38296
1. Ascent & Descent						38096	
a. Engine	E	A	100	6	600		
b. Fuel Tanks	E	A	3409		3409		
c. Propellant	E	A	34087		34087		
2. Fueled ACS	E	A	200			200	
E. Power							150
1. Batteries	E	A	150			150	
F. Equipment							270
1. Controls and Displays	E	A	60			60	
2. Communication	E	A	50			50	
3. Guidance and Navigation	E	A	100			100	
4. Docking Provisions	E	A	60			60	
G. MELS							4760
1. Aerobrake	E	A	1889			1889	
2. Parachute	E	A	2871			2871	

Total Mass (kg) = 45378

Human Mission
Mass Allocation Report

Mission: JBM-FD
L02.Hum4c.SpVs.ChHO.MOCP.0D.ChHO.ECCVd
L01.Car.Cn.ChHO.MOCP

Reference mission: JBM-FD *Trajectory file:* TRJ.6/30.02.Sp.Prop

Mission purpose:
CS-1 TIC-1R. All-prop, (480/460/320)
H-config (2 SS mod). Separate PhEV (9.8 t); 7t MRSR; ECCV direct entry

	Mass (t)		
MSS (IMLEO)			1311.34
TMIS		934.31	
Stage(s) to be dropped post-TMI	122.82		
TMI Propellant	811.49		
MTV		377.03	
MTV			377.03
Mars transfer expendables (MTX)		18.49	
Crew consumables	6.32		
MCC propellant	6.04		
RCS ETM propellant	6.14		
Venus swingby probes(s)		0.00	
MOCS		252.60	
Prop. sys. (dry)	33.90		
Propellant	218.70		
MOV (F1)		105.94	
MOV (F1)			105.94
Mars orbit expendables (MOX)		40.11	
Crew consumables	0.66		
MOO 1 propellant	16.23		
MOO 2 propellant	21.50		
RCS MOO propellant	1.72		
MOO prop. sys. (dry)		6.75	
Satellites		0.00	
RelayComSat(s)	0.00		
MarsSciSat(s)	0.00		
Ph/D teleoperator(s)	0.00		
Teleoperated MRSR		0.00	
MOSE		0.00	
MOV (Ff)		119.85	
TEIS (received from MCV)		-60.78	
MOV (Ff)			119.85
TEIS		60.78	
Stage to be dropped post-TEI	8.88		
TEI propellant	51.89		
ETV		59.07	
ETV			59.07
IMM		43.25	
External Services (power, com, thermal)		1.05	

Earth transfer expendables (ETX)		4.64	
Crew consumables	2.73		
Flyaround propellant	0.00		
MCC propellant	0.95		
RCS MTE propellant	0.96		
ETM/MTE MCC prop. sys. (dry)		1.33	
RCS ETM/MOO/MTE prop. sys. (dry)		1.01	
Spacesuits		0.42	
ISE		0.50	
Solar/SPE monitoring	0.40		
Astro/Planetary	0.10		
ECCV		6.87	
ECCV			6.87
Payload		5.97	
Crew+returnables+consum+suits	0.61		
Inert module	5.36		
EELS		0.90	
Earth entry Ab	0.60		
Propulsion	0.00		
Other EELS (parachutes, avionics)	0.30		
IMM			43.25
Cylindrical Module(s)		34.00	
Disk Module(s)		0.00	
Tunnel(s)		0.90	
Resource Nodes (docking, prox ops)		0.00	
Airlock(s)		0.65	
Radiation shelter shielding		2.00	
Life support system (LSS)		2.80	
Data management system (DMS)		0.30	
Internal Com/EPS/TCS		2.60	
External Services			1.05
Electrical power system (EPS), external		0.40	
Thermal control system (TCS), external		0.50	
Communications system, external		0.15	

Technology Status

Propulsion system masses	
LH2/LOX	Nominal
Stored biprop	Nominal
Mars aerobrake scaling	Nominal

Human Mission
Crew Consumables* Report

Mission: JBM-FD
L02.Hum4c.SpVs.ChHO.MOCP.0D.ChHO.ECCVd
L01.Car.Cn.ChHO.MOCP

Reference mission: JBM-FD *Trajectory file:* TRJ.6/30.02.Sp.Prop

Mission purpose:
CS-1 TIC-1R. All-prop, (480/460/320)
H-config (2 SS mod). Separate PhEV (9.8 t); 7t MRSR; ECCV direct entry

Crew composition: Nominal U. S. male crew

Period	Mission phase	# of crew	Time	Person-days	Margin	Total mass (t)
LEO Checkout	A	4	21 day	84	20 %	0.46
MTV	D	4	286 day	1145	20 %	6.32
MOV	F		30 day	120	20 %	0.66
ETV	H	4	124 day	495	20 %	2.73
ECCV	I	4	1 day	4	200 a %	0.07
					Total (incl. margin)	10.24
					Total (w/o margin)	8.50

Total supply = 5.06 person-years = 1848 person-days
Average supply = 5.54 kg/person-day

Consumables Baseline (nominal U. S. gender-mixed crew, kg/person-day):

	Food	Water	Other	Total
Spaceborne	1.5	1.0	2.1	4.6
Surface	1.5	2.0	2.0	5.5
MAV, ECCV	1.5	2.0	2.0	5.5

* Consumables includes LSS + Food
(a) To provide interplanetary safe-haven capability.

Human Mission
Dry* Masses Deployed or Jettisoned Report

Mission: JBM-FD
L02.Hum4c.SpVs.ChHO.MOCP.0D.ChHO.ECCVd
L01.Car.Cn.ChHO.MOCP

Reference mission: JBM-FD *Trajectory file:* TRJ.6/30.02.Sp.Prop

Mission purpose:
CS-1 TIC-1R. All-prop, (480/460/320)
H-config (2 SS mod). Separate PhEV (9.8 t); 7t MRSR; ECCV direct entry

What	When	Mission phase	Mass (t)
TMIS stage	during TMI	C	122.82
Venus probes	Venus swingby	D	0.00
MOC prop. sys.	post-MOC	E	33.90
RelayComSat(s)	Mars orbit	F1	0.00
MarsSciSat(s)	Mars orbit	F1	0.00
Ph/D teleoperator(s)	Mars orbit	F1	0.00
Teleoperated MRSR(s)	Mars orbit	F1	0.00
MOO prop. sys.	Mars orbit	F5	6.75
TEI stage	Post TEI	G	8.88
ECCV	pre-EOC	I	6.26
ETV	pre-EOC	I	47.14
		Total	225.77

*Not included:
propellants, consumables, crew mass, spacesuits, other expendables

Human Mission
Total Propellant* Report

Mission: JBM-FD
L02.Hum4c.SpVs.ChHO.MOCP.0D.ChHO.ECCVd
L01.Car.Cn.ChHO.MOCP

Reference mission: JBM-FD *Trajectory file:* TRJ.6/30.02.Sp.Prop

Mission purpose:
CS-1 TIC-1R. All-prop, (480/460/320)
H-config (2 SS mod). Separate PhEV (9.8 t); 7t MRSR; ECCV direct entry

		Propellant Mass Expended (t)			
		Main		ACS/RCS	
Period	Mission phase	Cryo	Biprop	Cryo	Biprop
TMI	C	811.489	0.000	0.000	0.000
Mars transfer (MCC, RCS)	D	0.000	6.038	0.000	6.138
MOC	E	218.698	0.000	0.000	0.000
Mars orbit (RCS)	F	0.000	0.000	0.000	1.725
Mars orbit (MOO 1)	F	16.230	0.000	0.000	0.000
Mars orbit (MOO 2)	F	21.496	0.000	0.000	0.000
RelayComSat(s)	F	0.000	0.000	0.000	0.000
MarsSciSat(s)	F	0.000	0.000	0.000	0.000
Ph/D teleoperator(s)	F	0.000	0.000	0.000	0.000
Teleoperated MRSR(s)	F	0.000	0.000	0.000	0.000
TEI	G	51.895	0.000	0.000	0.000
Earth transfer (MCC, RCS)	H	0.000	0.946	0.000	0.962
ECCV (EELS)	I	0.000	0.000	0.000	0.000
	Totals	1119.807	6.984	0.000	8.824

*Includes boiloff and reserves for Isp, delta V, and bulk propellant margins.

Human Mission
Propellant Reserves* Report

Mission: JBM-FD
L02.Hum4c.SpVs.ChHO.MOCP.0D.ChHO.ECCVd
L01.Car.Cn.ChHO.MOCP

Reference mission: JBM-FD *Trajectory file:* TRJ.6/30.02.Sp.Prop

Mission purpose:
CS-1 TIC-1R. All-prop, (480/460/320)
H-config (2 SS mod). Separate PhEV (9.8 t); 7t MRSR; ECCV direct entry

		Propellant Reserve (t)			
		Main		ACS/RCS	
Period	Mission phase	Cryo	Biprop	Cryo	Biprop
TMI	C	54.902	0.000	0.000	0.000
Mars transfer (MCC, RCS)	D	0.000	0.178	0.000	0.181
MOC	E	13.006	0.000	0.000	0.000
Mars orbit (RCS)	F	0.000	0.000	0.000	0.051
Mars orbit (MOO 1)	F	0.509	0.000	0.000	0.000
Mars orbit (MOO 2)	F	0.666	0.000	0.000	0.000
RelayComSat(s)	F	0.000	0.000	0.000	0.000
MarsSciSat(s)	F	0.000	0.000	0.000	0.000
Ph/D teleoperator(s)	F	0.000	0.000	0.000	0.000
Teleoperated MRSR(s)	F	0.000	0.000	0.000	0.000
TEI	G	3.023	0.000	0.000	0.000
Earth transfer (MCC, RCS)	H	0.000	0.028	0.000	0.028
ECCV (EELS)	I	0.000	0.000	0.000	0.000
	Totals	72.107	0.206	0.000	0.261

*Reserves for Isp, delta V, and bulk propellant margins.

Human Mission
Cryo-propellant Boiloff Report

Mission: JBM-FD
L02.Hum4c.SpVs.ChHO.MOCP.0D.ChHO.ECCVd
L01.Car.Cn.ChHO.MOCP

Reference mission: JBM-FD *Trajectory file:* TRJ.6/30.02.Sp.Prop

Mission purpose:
CS-1 TIC-1R. All-prop, (480/460/320)
H-config (2 SS mod). Separate PhEV (9.8 t); 7t MRSR; ECCV direct entry

Period	Mission phase	Cryo-propellant Boiloff (t) Main	ACS/RCS
TMI	C	11.583	0.000
Mars transfer (MCC, RCS)	D	0.000	0.000
MOC	E	17.706	0.000
Mars orbit (RCS)	F	0.000	0.000
Mars orbit (MOO 1)	F	0.519	0.000
Mars orbit (MOO 2)	F	0.672	0.000
RelayComSat(s)	F	0.000	0.000
MarsSciSat(s)	F	0.000	0.000
Ph/D teleoperator(s)	F	0.000	0.000
Teleoperated MRSR(s)	F	0.000	0.000
TEI	G	3.570	0.000
Earth transfer (MCC, RCS)	H	0.000	0.000
ECCV (EELS)	I	0.000	0.000
	Totals	34.048	0.000

Human Mission Astrodynamics Report

Mission: JBM-FD
L02.Hum4c.SpVs.ChHO.MOCP.0D.ChHO.ECCVd
L01.Car.Cn.ChHO.MOCP

Reference mission: JBM-FD *Trajectory file:* TRJ.6/30.02.Sp.Prop

Mission purpose:
CS-1 TIC-1R. All-prop, (480/460/320)
H-config (2 SS mod). Separate PhEV (9.8 t); 7t MRSR; ECCV direct entry

Trajectory Type: Sprint with Venus swingby(s)

	Date	Apoapsis (km)	Periapsis (km)	Inclination (deg)	Vinf (km/s)	C3 (km^2/s^2)
Earth departure (TMI)	15-Aug-02	500.0	500.0	28.50	5.216	27.21
Venus swingby	3-Nov-02					
Mars arrival (MOC)	28-May-03	33840.0	250.0	24.50	7.057	49.80
Mars departure (TEI)	27-Jun-03	33840.0	250.0	24.50	4.995	24.95
Earth arrival (EOC)	29-Oct-03	500.0	500.0	28.50	3.942	15.54

Duration

	Days	Sols	Months
Marsbound (ETM)	286.1		9.40
Mars orbit	30.0	29.2	0.99
Earthbound (MTE)	123.9		4.07
Total trip	440.0		14.46

Delta V Summary

Item	Delta V (km/s)
TMI	4.352
ETM DSM	0.000
ETM MCC	0.050
RCS ETM	0.050
MOC	3.931
MOO 1	0.719
MOO 2	0.619
RCS MOO	0.050
TEI	2.331
MTE DSM	0.000
MTE MCC	0.050
RCS MTE	0.050
EOC	3.854

Cargo Mission
Mass Allocation Report

Mission: JBM-FD
L02.Hum4c.SpVs.ChHO.MOCP.0D.ChHO.ECCVd
L01.Car.Cn.ChHO.MOCP

Reference mission: JBM-FD *Trajectory file:* TRJ.6/30.02.Sp.Prop

Mission purpose:
CS-1 TIC-1R. All-prop, (480/460/320)
H-config (2 SS mod). Separate PhEV (9.8 t); 7t MRSR; ECCV direct entry

		Mass (t)		
MCV (IMLEO)				466.97
TMIS			303.36	
Stage(s) to be dropped post-TMI		40.53		
TMI Propellant		262.83		
MTV			163.61	
MTV				163.61
Mars transfer expendables (MTX)			5.28	
MCC propellant		2.62		
RCS ETM propellant		2.66		
ETM MCC prop. sys. (dry)			0.00	
MOCS			57.88	
Prop. sys. (dry)		8.51		
Propellant		49.38		
MOV (F1)			100.44	
MOV (F1)				100.44
Structure			0.50	
Mars orbit expendables (MOX)			15.64	
MOO propellant		14.15		
RCS MOO propellant		1.49		
MOO prop. sys. (dry)			3.22	
RCS ETM/MOO prop. sys. (dry)			0.74	
Support Services			0.32	
Data management system (DMS)		0.05		
Electrical power system (EPS)		0.05		
Thermal control system (TCS)		0.15		
Communications system		0.07		
Payload			80.02	
Satellites		2.00		
RelayComSat(s)	1.00			
MarsSciSat(s)	1.00			
Ph/D teleoperator(s)	0.00			
Teleoperated MRSR		7.00		
MOSE		0.15		
ISE		0.30		
Solar/SPE monitoring	0.20			
Astro/Planetary	0.10			
Phobos EV		9.79		
TEIS		60.78		
Stage	8.88			
Propellant	51.89			

Technology Status

Propulsion system masses	
LH2/LOX	Nominal
Stored biprop	Nominal
Mars aerobrake scaling	Nominal

Cargo Mission
Dry* Masses Deployed or Jettisoned Report

Mission: JBM-FD
L02.Hum4c.SpVs.ChHO.MOCP.0D.ChHO.ECCVd
L01.Car.Cn.ChHO.MOCP

Reference mission: JBM-FD *Trajectory file:* TRJ.6/30.02.Sp.Prop

Mission purpose:
CS-1 TIC-1R. All-prop, (480/460/320)
H-config (2 SS mod). Separate PhEV (9.8 t); 7t MRSR; ECCV direct entry

What	When	Mission phase	Mass (t)
TMIS stage	during TMI	C	40.53
ETM MCC prop. sys.	pre-MOC	E	0.00
Mars Ab	post-MOC	E	0.00
MOC prop. sys.	post-MOC	E	8.51
Other MOCS	post-MOC	E	0.00
RelayComSat(s)	Mars orbit	F1	1.00
MarsSciSat(s)	Mars orbit	F1	1.00
Ph/D teleoperator(s)	Mars orbit	F1	0.00
Teleoperated MRSR(s)	Mars orbit	F1	7.00
Phobos EV	Mars orbit	F1	9.79
MOO prop. sys.	Mars orbit	F5	3.22
MOV, pre-human TEI	Mars orbit	F5	2.01
		Total	73.06

*Not included:
propellants, consumables, crew mass, spacesuits, other expendables

Cargo Mission
Total Propellant* Report

Mission: JBM-FD
L02.Hum4c.SpVs.ChHO.MOCP.0D.ChHO.ECCVd
L01.Car.Cn.ChHO.MOCP

Reference mission: JBM-FD *Trajectory file:* TRJ.6/30.02.Sp.Prop

Mission purpose:
CS-1 TIC-1R. All-prop, (480/460/320)
H-config (2 SS mod). Separate PhEV (9.8 t); 7t MRSR; ECCV direct entry

		Propellant Mass Expended (t)			
		Main		ACS/RCS	
Period	Mission phase	Cryo	Biprop	Cryo	Biprop
TMI	C	262.834	0.000	0.000	0.000
Mars transfer	D	0.000	2.620	0.000	2.664
MOC	E	49.375	0.000	0.000	0.000
Mars orbit operations	F	14.148	0.000	0.000	1.489
RelayComSat(s)	F	0.000	0.000	0.000	0.000
MarsSciSat(s)	F	0.000	0.000	0.000	0.000
Ph/D teleoperator(s)	F	0.000	0.000	0.000	0.000
Teleoperated MRSR(s)	F	0.000	0.000	0.000	0.000
	Totals	326.357	2.620	0.000	4.152

*Includes boiloff and reserves for Isp, delta V, and bulk propellant margins.

Cargo Mission
Propellant Reserves* Report

Mission: JBM-FD
L02.Hum4c.SpVs.ChHO.MOCP.0D.ChHO.ECCVd
L01.Car.Cn.ChHO.MOCP

Reference mission: JBM-FD *Trajectory file:* TRJ.6/30.02.Sp.Prop

Mission purpose:
CS-1 TIC-1R. All-prop, (480/460/320)
H-config (2 SS mod). Separate PhEV (9.8 t); 7t MRSR; ECCV direct entry

		Propellant Reserve (t)			
		Main		ACS/RCS	
Period	Mission phase	Cryo	Biprop	Cryo	Biprop
TMI	C	15.484	0.000	0.000	0.000
Mars transfer	D	0.000	0.077	0.000	0.079
MOC	E	1.860	0.000	0.000	0.000
Mars orbit operations	F	0.450	0.000	0.000	0.044
RelayComSat(s)	F	0.000	0.000	0.000	0.000
MarsSciSat(s)	F	0.000	0.000	0.000	0.000
Ph/D teleoperator(s)	F	0.000	0.000	0.000	0.000
Teleoperated MRSR(s)	F	0.000	0.000	0.000	0.000
	Totals	17.794	0.077	0.000	0.123

*Reserves for Isp, delta V, and bulk propellant margins.

Cargo Mission
Cryo-propellant Boiloff Report

Mission: JBM-FD
L02.Hum4c.SpVs.ChHO.MOCP.0D.ChHO.ECCVd
L01.Car.Cn.ChHO.MOCP

Reference mission: JBM-FD *Trajectory file:* TRJ.6/30.02.Sp.Prop

Mission purpose:
CS-1 TIC-1R. All-prop, (480/460/320)
H-config (2 SS mod). Separate PhEV (9.8 t); 7t MRSR; ECCV direct entry

Period	Mission phase	Cryo-propellant Boiloff (t) Main	ACS/RCS
TMI	C	3.102	0.000
Mars transfer	D	0.000	0.000
MOC	E	1.874	0.000
Mars orbit operations	F	0.627	0.000
RelayComSat(s)	F	0.000	0.000
MarsSciSat(s)	F	0.000	0.000
Ph/D teleoperator(s)	F	0.000	0.000
Teleoperated MRSR(s)	F	0.000	0.000
	Totals	5.602	0.000

Cargo Mission Astrodynamics Report

Mission: JBM-FD
L02.Hum4c.SpVs.ChHO.MOCP.0D.ChHO.ECCVd
L01.Car.Cn.ChHO.MOCP

Reference mission: JBM-FD *Trajectory file:* TRJ.6/30.02.Sp.Prop

Mission purpose:
CS-1 TIC-1R. All-prop, (480/460/320)
H-config (2 SS mod). Separate PhEV (9.8 t); 7t MRSR; ECCV direct entry

Trajectory Type: Conjunction

	Date	Apoapsis (km)	Periapsis (km)	Inclination (deg)	Vinf (km/s)	C3 (km^2/s^2)
Earth departure (TMI)	4-Feb-01	500.0	500.0	28.50	3.593	12.91
Mars arrival (MOC)	30-Oct-01	33840.0	250.0	24.50	3.888	15.12

Duration	Days	Sols	Months
Marsbound (ETM)	268.0		8.80
Mars orbit	606.5	589.5	19.93

Delta V Summary

Item	Delta V (km/s)
TMI	3.739
ETM MCC	0.050
RCS ETM	0.050
MOC	1.585
MOO	0.719
RCS MOO	0.050

Human Mission
Mass Allocation Report

Mission: BCC-HG
L04.Hum8c.SpDsm.ChHO.MAb.1D4cP.20s2R.ChHO.ECCVo
L03.Car.Cn.ChHO.MAb

Reference mission: BCC-HG *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-1R Reference. (Sprint '04, 485/470/320, MAb, 2t rad shield)
PhTele, but no MOV to Phobos. ECCV into LEO.

	------------------ Mass (t)	-----------------	
MSS (IMLEO)			1124.32
TMIS		790.23	
Stage(s) to be dropped post-TMI	104.99		
TMI Propellant	685.25		
MTV		334.09	
MTV			334.09
Mars transfer expendables (MTX)		131.76	
Crew consumables	10.46		
DSM propellant	114.81		
MCC propellant	3.22		
RCS ETM propellant	3.27		
ETM DSM prop. sys. (dry)		18.32	
MOCS		27.16	
Mars capture Ab	23.53		
Prop. sys. (dry)	1.17		
Propellant	0.46		
other MOCS	2.00		
MOV (F1)		156.85	
MOV (F1)			156.85
Mars orbit expendables (MOX)		60.95	
Crew consumables	0.78		
MOO 1 propellant	23.36		
MOO 2 propellant	33.25		
RCS MOO propellant	3.56		
MOO prop. sys. (dry)		9.59	
Satellites		0.00	
RelayComSat(s)	0.00		
MarsSciSat(s)	0.00		
Ph/D teleoperator(s)	0.00		
Teleoperated MRSR		0.00	
MOSE		0.00	
MOV (Ff)		155.80	
TEIS (received from MCV)		-69.48	
MOV (Ff)			155.80
TEIS		69.48	
Stage to be dropped post-TEI	10.02		
TEI propellant	59.46		
ETV		86.32	

ETV			86.32
IMM		64.85	
External Services (power, com, thermal)		1.05	
Earth transfer expendables (ETX)		7.92	
Crew consumables	5.68		
Flyaround propellant	0.00		
MCC propellant	1.39		
RCS MTE propellant	0.85		
ETM/MTE MCC prop. sys. (dry)		1.18	
RCS ETM/MOO/MTE prop. sys. (dry)		0.95	
Spacesuits		0.70	
ISE		0.50	
Solar/SPE monitoring	0.40		
Astro/Planetary	0.10		
ECCV		9.18	
ECCV			9.18
Payload		6.82	
Crew+returnables+consum+suits	1.11		
Inert module	5.71		
EOCS		2.36	
Ab	0.68		
Propulsion	1.33		
Other EOCS	0.34		
IMM			64.85
Cylindrical Module(s)		51.00	
Disk Module(s)		2.50	
Tunnel(s)		2.70	
Resource Nodes (docking, prox ops)		0.00	
Airlock(s)		0.95	
Radiation shelter shielding		2.00	
Life support system (LSS)		2.80	
Data management system (DMS)		0.30	
Internal Com/EPS/TCS		2.60	
External Services			1.05
Electrical power system (EPS), external		0.40	
Thermal control system (TCS), external		0.50	
Communications system, external		0.15	

Technology Status

Propulsion system masses	
LH2/LOX	Nominal
Stored biprop	Nominal
Mars aerobrake scaling	V. conservative

Human Mission
Crew Consumables* Report

Mission: BCC-HG
L04.Hum8c.SpDsm.ChHO.MAb.1D4cP.20s2R.ChHO.ECCVo
L03.Car.Cn.ChHO.MAb

Reference mission: BCC-HG *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-1R Reference. (Sprint '04, 485/470/320, MAb, 2t rad shield)
PhTele, but no MOV to Phobos. ECCV into LEO.

Crew composition: Nominal U. S. male crew

Period	Mission phase	# of crew	Time	Person-days	Margin	Total mass (t)
LEO Checkout	A	8	21 day	168	20 %	0.83
MTV	D	8	266 day	2125	20 %	10.46
MOV	F		30 day	158	20 %	0.78
MDV 1	F3	4	20 sol	82	5 %	0.40
MAV 1	F4	4	1 day	4	200 %	0.07
ETV	H	8	144 day	1155	20 %	5.68
ECCV	I	8	1 day	8	200 a %	0.13
					Total (incl. margin)	18.34
					Total (w/o margin)	15.23

Total supply = 10.13 person-years = 3700 person-days
Average supply = 4.96 kg/person-day

Consumables Baseline (nominal U. S. gender-mixed crew, kg/person-day):

	Food	Water	Other	Total
Spaceborne	1.5	0.5	2.1	4.1
Surface	1.5	2.5	0.7	4.7
MAV, ECCV	1.5	2.0	2.0	5.5

* Consumables includes LSS + Food
(a) To provide interplanetary safe-haven capability.

Human Mission
Dry* Masses Deployed or Jettisoned Report

Mission: BCC-HG
L04.Hum8c.SpDsm.ChHO.MAb.1D4cP.20s2R.ChHO.ECCVo
L03.Car.Cn.ChHO.MAb

Reference mission: BCC-HG *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-1R Reference. (Sprint '04, 485/470/320, MAb, 2t rad shield)
PhTele, but no MOV to Phobos. ECCV into LEO.

What	When	Mission phase	Mass (t)
TMIS stage 1	during TMI	C	67.61
TMIS stage 2	during TMI	C	37.38
ETM DSM prop. sys.	Mars transfer	D	18.32
Mars Ab	post-MOC	E	23.53
MOC prop. sys.	post-MOC	E	1.17
Other MOCS	post-MOC	E	2.00
RelayComSat(s)	Mars orbit	F1	0.00
MarsSciSat(s)	Mars orbit	F1	0.00
Ph/D teleoperator(s)	Mars orbit	F1	0.00
Teleoperated MRSR(s)	Mars orbit	F1	0.00
MOO prop. sys.	Mars orbit	F5	9.59
TEI stage	Post TEI	G	10.02
ECCV	pre-EOC	I	7.66
ETV	pre-EOC	I	68.52
		Total	245.80

*Not included:
propellants, consumables, crew mass, spacesuits, other expendables

Human Mission
Total Propellant* Report

Mission: BCC-HG
L04.Hum8c.SpDsm.ChHO.MAb.1D4cP.20s2R.ChHO.ECCVo
L03.Car.Cn.ChHO.MAb

Reference mission: BCC-HG *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-1R Reference. (Sprint '04, 485/470/320, MAb, 2t rad shield)
PhTele, but no MOV to Phobos. ECCV into LEO.

		Propellant Mass Expended (t)			
		Main		ACS/RCS	
Period	Mission phase	Cryo	Biprop	Cryo	Biprop
TMI stage 1	C	443.375	0.000	0.000	0.000
TMI stage 2	C	241.870	0.000	0.000	0.000
Mars transfer (DSM)	D	114.807	0.000	0.000	0.000
Mars transfer (MCC, RCS)	D	0.000	3.218	0.000	3.274
MOC	E	0.462	0.000	0.000	0.000
Mars orbit (RCS)	F	0.000	0.000	0.000	3.563
Mars orbit (MOO 1)	F	23.361	0.000	0.000	0.000
Mars orbit (MOO 2)	F	33.246	0.000	0.000	0.000
RelayComSat(s)	F	0.000	0.000	0.000	0.000
MarsSciSat(s)	F	0.000	0.000	0.000	0.000
Ph/D teleoperator(s)	F	0.000	0.000	0.000	0.000
Teleoperated MRSR(s)	F	0.000	0.000	0.000	0.000
TEI	G	59.463	0.000	0.000	0.000
Earth transfer (MCC, RCS)	H	0.000	1.391	0.000	0.846
ECCV (EOCS)	I	0.000	0.401	0.000	0.000
	Totals	916.584	5.011	0.000	7.683

*Includes boiloff and reserves for Isp, delta V, and bulk propellant margins.

Human Mission Propellant Reserves* Report

Mission: BCC-HG
L04.Hum8c.SpDsm.ChHO.MAb.1D4cP.20s2R.ChHO.ECCVo
L03.Car.Cn.ChHO.MAb

Reference mission: BCC-HG *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-1R Reference. (Sprint '04, 485/470/320, MAb, 2t rad shield)
PhTele, but no MOV to Phobos. ECCV into LEO.

		Propellant Reserve (t)			
		Main		ACS/RCS	
Period	Mission phase	Cryo	Biprop	Cryo	Biprop
TMI stage 1	C	19.372	0.000	0.000	0.000
TMI stage 2	C	10.568	0.000	0.000	0.000
Mars transfer (DSM)	D	4.624	0.000	0.000	0.000
Mars transfer (MCC, RCS)	D	0.000	0.095	0.000	0.096
MOC	E	0.013	0.000	0.000	0.000
Mars orbit (RCS)	F	0.000	0.000	0.000	0.106
Mars orbit (MOO 1)	F	0.733	0.000	0.000	0.000
Mars orbit (MOO 2)	F	1.030	0.000	0.000	0.000
RelayComSat(s)	F	0.000	0.000	0.000	0.000
MarsSciSat(s)	F	0.000	0.000	0.000	0.000
Ph/D teleoperator(s)	F	0.000	0.000	0.000	0.000
Teleoperated MRSR(s)	F	0.000	0.000	0.000	0.000
TEI	G	3.277	0.000	0.000	0.000
Earth transfer (MCC, RCS)	H	0.000	0.041	0.000	0.025
ECCV (EOCS)	I	0.000	0.012	0.000	0.000
	Totals	39.617	0.148	0.000	0.227

*Reserves for Isp, delta V, and bulk propellant margins.

Human Mission
Cryo-propellant Boiloff Report

Mission: BCC-HG
L04.Hum8c.SpDsm.ChHO.MAb.1D4cP.20s2R.ChHO.ECCVo
L03.Car.Cn.ChHO.MAb

Reference mission: BCC-HG *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-1R Reference. (Sprint '04, 485/470/320, MAb, 2t rad shield)
PhTele, but no MOV to Phobos. ECCV into LEO.

Period	Mission phase	Cryo-propellant Boiloff (t) Main	ACS/RCS
TMI stage 1	C	3.448	0.000
TMI stage 2	C	1.881	0.000
Mars transfer (DSM)	D	2.341	0.000
Mars transfer (MCC, RCS)	D	0.000	0.000
MOC	E	0.011	0.000
Mars orbit (RCS)	F	0.000	0.000
Mars orbit (MOO 1)	F	0.693	0.000
Mars orbit (MOO 2)	F	0.965	0.000
RelayComSat(s)	F	0.000	0.000
MarsSciSat(s)	F	0.000	0.000
Ph/D teleoperator(s)	F	0.000	0.000
Teleoperated MRSR(s)	F	0.000	0.000
TEI	G	3.071	0.000
Earth transfer (MCC, RCS)	H	0.000	0.000
ECCV (EOCS)	I	0.000	0.000
	Totals	12.411	0.000

Human Mission Astrodynamics Report

Mission: BCC-HG
L04.Hum8c.SpDsm.ChHO.MAb.1D4cP.20s2R.ChHO.ECCVo
L03.Car.Cn.ChHO.MAb

Reference mission: BCC-HG *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-1R Reference. (Sprint '04, 485/470/320, MAb, 2t rad shield)
PhTele, but no MOV to Phobos. ECCV into LEO.

Trajectory Type: Sprint with deep space maneuver(s)

	Date	Apoapsis (km)	Periapsis (km)	Inclination (deg)	Vinf (km/s)	C3 (km^2/s^2)
Earth departure (TMI)	9-Oct-04	500.0	500.0	28.50	5.748	33.04
Mars arrival (MOC)	2-Jul-05	33840.0	250.0	45.00	6.569	43.15
Mars departure (TEI)	1-Aug-05	33840.0	250.0	45.00	4.581	20.99
Earth arrival (EOC)	23-Dec-05	500.0	500.0	28.50	5.000	25.00

Duration

	Days	Sols	Months
Marsbound (ETM)	265.7		8.73
Mars orbit	30.0	29.2	0.99
Mars surface			
MDV1		20.0	
Earthbound (MTE)	144.3		4.74
Total trip	440.0		14.46

Delta V Summary

Item	Delta V (km/s)
TMI	4.593
ETM DSM	1.874
ETM MCC	0.050
RCS ETM	0.030
MOC	3.533
MOO 1	0.719
MOO 2	0.619
RCS MOO	0.070
MDV 1 deorbit	0.000
MDV 1 terminal descent	0.150
MAV 1 ascent	5.408
MAV 1 RCS	0.050
TEI	2.040
MTE DSM	0.000
MTE MCC	0.050
RCS MTE	0.030
EOC	4.260

Mars Descent and Ascent Vehicles Mass Allocation Report

Mission: BCC-HG
L04.Hum8c.SpDsm.ChHO.MAb.1D4cP.20s2R.ChHO.ECCVo
L03.Car.Cn.ChHO.MAb

Reference mission: BCC-HG *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-1R Reference. (Sprint '04, 485/470/320, MAb, 2t rad shield)
PhTele, but no MOV to Phobos. ECCV into LEO.

	Mass (t)			
MDV 1 (w/o crew)				51.25
MELS			10.89	
Mars entry Ab		2.39		
Propulsion, deorbit+descent		4.64		
Parachute		3.87		
Adapter structure (incl. landing legs)			3.70	
Landed P/L (w/o crew)			13.60	
MLMM		5.63		
MLOE		6.57		
MLSE	2.30			
MLTE (RVRs, suits)	1.02			
Teleoperated equip.	3.00			
MLCE	0.15			
MLME (ISRU demo's)	0.10			
Crew*		(0.32)		
Mars landed expendables (MLX)		1.40		
Crew consumables	0.40			
other	1.00			
MAV (w/o crew)			23.06	
MAV (w/o crew)				23.06
Payload			1.24	
(Crew+returnables)*		(0.47)		
MAV Crew consumables + suits		0.19		
Inert module		0.97		
RCS propulsion		0.08		
Ascent propulsion			21.82	
MLMM				5.63
Structure			2.80	
Pressure shell, support structure		2.80		
Partitions, equipment acceptors		0.00		
Windows		0.00		
Man-systems			0.50	
Living quarters		0.30		
Galley		0.10		
Personal hygiene		0.10		
ECLSS			0.79	
DMS			0.05	
EPS			0.54	
TCS			0.70	
Communications			0.25	

* Used only to size propulsion. Crew+returnables allocated in ECCV.

Technology	Status
Propulsion system masses	
LH2/LOX	Nominal
Stored biprop	Nominal
MELS aerobrake scaling	Nominal
MDV adapter scaling	Nominal
MAV inert module mass	Nominal

Cargo Mission
Mass Allocation Report

Mission: BCC-HG
L04.Hum8c.SpDsm.ChHO.MAb.1D4cP.20s2R.ChHO.ECCVo
L03.Car.Cn.ChHO.MAb

Reference mission: BCC-HG *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-1R Reference. (Sprint '04, 485/470/320, MAb, 2t rad shield)
PhTele, but no MOV to Phobos. ECCV into LEO.

	Mass (t)			
MCV (IMLEO)				503.25
TMIS			314.20	
Stage(s) to be dropped post-TMI		41.94		
TMI Propellant		272.26		
MTV			189.05	
MTV				189.05
Mars transfer expendables (MTX)			6.11	
MCC propellant		3.03		
RCS ETM propellant		3.08		
ETM MCC prop. sys. (dry)			0.00	
MOCS			19.95	
Mars capture Ab		16.30		
Prop. sys. (dry)		1.17		
Propellant		0.48		
other MOCS		2.00		
MOV (F1)			163.00	
MOV (F1)				163.00
Structure			0.50	
Mars orbit expendables (MOX)			28.07	
MOO propellant		23.05		
RCS MOO propellant		5.02		
MOO prop. sys. (dry)			4.56	
RCS ETM/MOO prop. sys. (dry)			0.97	
Support Services			0.21	
Data management system (DMS)		0.03		
Electrical power system (EPS)		0.05		
Thermal control system (TCS)		0.10		
Communications system		0.03		
Payload			128.69	
Satellites		4.00		
RelayComSat(s)	1.00			
MarsSciSat(s)	1.00			
Ph/D teleoperator(s)	2.00			
Teleoperated MRSR		3.50		
MOSE		0.15		
ISE		0.30		
Solar/SPE monitoring	0.20			
Astro/Planetary	0.10			
MDV(s)		51.25		
TEIS		69.48		

Stage	10.02
Propellant	59.46

Technology Status

Propulsion system masses	
LH2/LOX	Nominal
Stored biprop	Nominal
Mars aerobrake scaling	Nominal

Cargo Mission
Dry* Masses Deployed or Jettisoned Report

Mission: BCC-HG
L04.Hum8c.SpDsm.ChHO.MAb.1D4cP.20s2R.ChHO.ECCVo
L03.Car.Cn.ChHO.MAb

Reference mission: BCC-HG *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-1R Reference. (Sprint '04, 485/470/320, MAb, 2t rad shield)
PhTele, but no MOV to Phobos. ECCV into LEO.

What	When	Mission phase	Mass (t)
TMIS stage	during TMI	C	41.94
ETM MCC prop. sys.	pre-MOC	E	0.00
Mars Ab	post-MOC	E	16.30
MOC prop. sys.	post-MOC	E	1.17
Other MOCS	post-MOC	E	2.00
RelayComSat(s)	Mars orbit	F1	1.00
MarsSciSat(s)	Mars orbit	F1	1.00
Ph/D teleoperator(s)	Mars orbit	F1	2.00
Teleoperated MRSR(s)	Mars orbit	F1	3.50
MDV 1 (without MAV 1)	Mars orbit	F2	22.95
MAV 1	Mars orbit	F5	3.08
MOO prop. sys.	Mars orbit	F5	4.56
MOV, pre-human TEI	Mars orbit	F5	2.13
		Total	101.63

*Not included:
propellants, consumables, crew mass, spacesuits, other expendables

Cargo Mission
Total Propellant* Report

Mission: BCC-HG
L04.Hum8c.SpDsm.ChHO.MAb.1D4cP.20s2R.ChHO.ECCVo
L03.Car.Cn.ChHO.MAb

Reference mission: BCC-HG *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-1R Reference. (Sprint '04, 485/470/320, MAb, 2t rad shield)
PhTele, but no MOV to Phobos. ECCV into LEO.

		Propellant Mass Expended (t)			
		Main		ACS/RCS	
Period	Mission phase	Cryo	Biprop	Cryo	Biprop
TMI	C	272.259	0.000	0.000	0.000
Mars transfer	D	0.000	3.028	0.000	3.078
MOC	E	0.477	0.000	0.000	0.000
Mars orbit operations	F	23.050	0.000	0.000	5.022
RelayComSat(s)	F	0.000	0.000	0.000	0.000
MarsSciSat(s)	F	0.000	0.000	0.000	0.000
Ph/D teleoperator(s)	F	0.000	0.000	0.000	0.000
Teleoperated MRSR(s)	F	0.000	0.000	0.000	0.000
MDV 1 deorbit	F2a	0.000	0.248	0.000	0.000
MDV 1 terminal descent	F2b	0.000	3.274	0.000	0.000
MAV 1	F4	0.000	19.764	0.000	0.028
	Totals	295.785	26.313	0.000	8.128

*Includes boiloff and reserves for Isp, delta V, and bulk propellant margins.

Cargo Mission
Propellant Reserves* Report

Mission: BCC-HG
L04.Hum8c.SpDsm.ChHO.MAb.1D4cP.20s2R.ChHO.ECCVo
L03.Car.Cn.ChHO.MAb

Reference mission: BCC-HG *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-1R Reference. (Sprint '04, 485/470/320, MAb, 2t rad shield)
PhTele, but no MOV to Phobos. ECCV into LEO.

		Propellant Reserve (t)			
		Main		ACS/RCS	
Period	Mission phase	Cryo	Biprop	Cryo	Biprop
TMI	C	15.300	0.000	0.000	0.000
Mars transfer	D	0.000	0.089	0.000	0.091
MOC	E	0.014	0.000	0.000	0.000
Mars orbit operations	F	0.737	0.000	0.000	0.150
RelayComSat(s)	F	0.000	0.000	0.000	0.000
MarsSciSat(s)	F	0.000	0.000	0.000	0.000
Ph/D teleoperator(s)	F	0.000	0.000	0.000	0.000
Teleoperated MRSR(s)	F	0.000	0.000	0.000	0.000
MDV 1 deorbit	F2a	0.000	0.007	0.000	0.000
MDV 1 terminal descent	F2b	0.000	0.062	0.000	0.000
MAV 1	F4	0.000	3.314	0.000	0.001
	Totals	16.050	3.473	0.000	0.242

*Reserves for Isp, delta V, and bulk propellant margins.

Cargo Mission
Cryo-propellant Boiloff Report

Mission: BCC-HG
L04.Hum8c.SpDsm.ChHO.MAb.1D4cP.20s2R.ChHO.ECCVo
L03.Car.Cn.ChHO.MAb

Reference mission: BCC-HG *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-1R Reference. (Sprint '04, 485/470/320, MAb, 2t rad shield)
PhTele, but no MOV to Phobos. ECCV into LEO.

Period	Mission phase	Cryo-propellant Boiloff (t) Main	ACS/RCS
TMI	C	3.010	0.000
Mars transfer	D	0.000	0.000
MOC	E	0.009	0.000
Mars orbit operations	F	0.841	0.000
RelayComSat(s)	F	0.000	0.000
MarsSciSat(s)	F	0.000	0.000
Ph/D teleoperator(s)	F	0.000	0.000
Teleoperated MRSR(s)	F	0.000	0.000
MDV 1 deorbit	F2a	0.000	0.000
MDV 1 terminal descent	F2b	0.000	0.000
MAV 1	F4	0.000	0.000
	Totals	3.861	0.000

Cargo Mission
Astrodynamics Report

Mission: BCC-HG
L04.Hum8c.SpDsm.ChHO.MAb.1D4cP.20s2R.ChHO.ECCVo
L03.Car.Cn.ChHO.MAb

Reference mission: BCC-HG *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-1R Reference. (Sprint '04, 485/470/320, MAb, 2t rad shield)
PhTele, but no MOV to Phobos. ECCV into LEO.

Trajectory Type: Conjunction

	Date	Apoapsis (km)	Periapsis (km)	Inclination (deg)	Vinf (km/s)	C3 (km^2/s^2)
Earth departure (TMI)	7-Jun-03	500.0	500.0	28.50	2.968	8.81
Mars arrival (MOC)	25-Dec-03	33840.0	250.0	45.00	2.705	7.32

Duration	Days	Sols	Months
Marsbound (ETM)	201.0		6.60
Mars orbit	586.5	570.1	19.27

Delta V Summary

Item	Delta V (km/s)
TMI	3.557
ETM MCC	0.050
RCS ETM	0.050
MOC	0.923
MOO	0.719
RCS MOO	0.100

Human Mission
Mass Allocation Report

Mission: BCC-HQ
L06.Hum8c.SpDsm.ChHO.MAb.1D4cP.15s2R.ChHO.ECCVo
L05.Car.Cn.ChHO.MAb

Reference mission: BCC-HQ *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-2R Ref., Miss 2. (Sprint '06, 485/470/320, MAb, 2t rad shield)
PhEV to Phobos, 1 MRSR. ECCV into LEO.

	------------------ Mass (t)	------------------	
MSS (IMLEO)			1769.66
TMIS		1423.64	
Stage(s) to be dropped post-TMI	187.60		
TMI Propellant	1236.03		
MTV		346.02	
MTV			346.02
Mars transfer expendables (MTX)		129.43	
Crew consumables	9.01		
DSM propellant	113.59		
MCC propellant	3.43		
RCS ETM propellant	3.39		
ETM DSM prop. sys. (dry)		18.14	
MOCS		29.08	
Mars capture Ab	25.41		
Prop. sys. (dry)	1.17		
Propellant	0.50		
other MOCS	2.00		
MOV (F1)		169.38	
MOV (F1)			169.38
Mars orbit expendables (MOX)		70.56	
Crew consumables	0.88		
MOO 1 propellant	25.14		
MOO 2 propellant	40.70		
RCS MOO propellant	3.85		
MOO prop. sys. (dry)		10.97	
Satellites		0.00	
RelayComSat(s)	0.00		
MarsSciSat(s)	0.00		
Ph/D teleoperator(s)	0.00		
Teleoperated MRSR		0.00	
MOSE		0.00	
MOV (Ff)		194.45	
TEIS (received from MCV)		-106.61	
MOV (Ff)			194.45
TEIS		106.61	
Stage to be dropped post-TEI	14.86		
TEI propellant	91.75		
ETV		87.84	

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ETV			87.84
IMM		64.85	
External Services (power, com, thermal)		1.05	
Earth transfer expendables (ETX)		9.40	
Crew consumables	7.12		
Flyaround propellant	0.00		
MCC propellant	1.42		
RCS MTE propellant	0.86		
ETM/MTE MCC prop. sys. (dry)		1.19	
RCS ETM/MOO/MTE prop. sys. (dry)		0.97	
Spacesuits		0.70	
ISE		0.50	
Solar/SPE monitoring	0.40		
Astro/Planetary	0.10		
ECCV		9.18	
ECCV			9.18
Payload		6.82	
Crew+returnables+consum+suits	1.11		
Inert module	5.71		
EOCS		2.36	
Ab	0.68		
Propulsion	1.33		
Other EOCS	0.34		
IMM			64.85
Cylindrical Module(s)		51.00	
Disk Module(s)		2.50	
Tunnel(s)		2.70	
Resource Nodes (docking, prox ops)		0.00	
Airlock(s)		0.95	
Radiation shelter shielding		2.00	
Life support system (LSS)		2.80	
Data management system (DMS)		0.30	
Internal Com/EPS/TCS		2.60	
External Services			1.05
Electrical power system (EPS), external		0.40	
Thermal control system (TCS), external		0.50	
Communications system, external		0.15	

Technology Status

Propulsion system masses	
LH2/LOX	Nominal
Stored biprop	Nominal
Mars aerobrake scaling	V. conservative

Human Mission
Crew Consumables* Report

Mission: BCC-HQ
L06.Hum8c.SpDsm.ChHO.MAb.1D4cP.15s2R.ChHO.ECCVo
L05.Car.Cn.ChHO.MAb

Reference mission: BCC-HQ *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-2R Ref., Miss 2. (Sprint '06, 485/470/320, MAb, 2t rad shield)
PhEV to Phobos, 1 MRSR. ECCV into LEO.

Crew composition: Nominal U. S. male crew

Period	Mission phase	# of crew	Time	Person-days	Margin	Total mass (t)
LEO Checkout	A	8	21 day	168	20 %	0.83
MTV	D	8	229 day	1832	20 %	9.01
MOV	F		30 day	178	20 %	0.88
MDV 1	F3	4	15 sol	62	5 %	0.30
MAV 1	F4	4	1 day	4	200 %	0.07
ETV	H	8	181 day	1448	20 %	7.12
ECCV	I	8	1 day	8	200 a %	0.13
					Total (incl. margin)	18.34
					Total (w/o margin)	15.22

Total supply = 10.13 person-years = 3700 person-days
Average supply = 4.96 kg/person-day

Consumables Baseline (nominal U. S. gender-mixed crew, kg/person-day):

	Food	Water	Other	Total
Spaceborne	1.5	0.5	2.1	4.1
Surface	1.5	2.5	0.7	4.7
MAV, ECCV	1.5	2.0	2.0	5.5

* Consumables includes LSS + Food
(a) To provide interplanetary safe-haven capability.

Human Mission
Dry* Masses Deployed or Jettisoned Report

Mission: BCC-HQ
L06.Hum8c.SpDsm.ChHO.MAb.1D4cP.15s2R.ChHO.ECCVo
L05.Car.Cn.ChHO.MAb

Reference mission: BCC-HQ *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-2R Ref., Miss 2. (Sprint '06, 485/470/320, MAb, 2t rad shield)
PhEV to Phobos, 1 MRSR. ECCV into LEO.

What	When	Mission phase	Mass (t)
TMIS stage 1	during TMI	C	129.62
TMIS stage 2	during TMI	C	57.98
ETM DSM prop. sys.	Mars transfer	D	18.14
Mars Ab	post-MOC	E	25.41
MOC prop. sys.	post-MOC	E	1.17
Other MOCS	post-MOC	E	2.00
RelayComSat(s)	Mars orbit	F1	0.00
MarsSciSat(s)	Mars orbit	F1	0.00
Ph/D teleoperator(s)	Mars orbit	F1	0.00
Teleoperated MRSR(s)	Mars orbit	F1	0.00
MOO prop. sys.	Mars orbit	F5	10.97
TEI stage	Post TEI	G	14.86
ECCV	pre-EOC	I	7.66
ETV	pre-EOC	I	68.56
		Total	336.38

*Not included:
propellants, consumables, crew mass, spacesuits, other expendables

Human Mission
Total Propellant* Report

Mission: BCC-HQ
L06.Hum8c.SpDsm.ChHO.MAb.1D4cP.15s2R.ChHO.ECCVo
L05.Car.Cn.ChHO.MAb

Reference mission: BCC-HQ *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-2R Ref., Miss 2. (Sprint '06, 485/470/320, MAb, 2t rad shield)
PhEV to Phobos, 1 MRSR. ECCV into LEO.

		Propellant Mass Expended (t)			
		Main		ACS/RCS	
Period	Mission phase	Cryo	Biprop	Cryo	Biprop
TMI stage 1	C	856.820	0.000	0.000	0.000
TMI stage 2	C	379.211	0.000	0.000	0.000
Mars transfer (DSM)	D	113.591	0.000	0.000	0.000
Mars transfer (MCC, RCS)	D	0.000	3.433	0.000	3.391
MOC	E	0.497	0.000	0.000	0.000
Mars orbit (RCS)	F	0.000	0.000	0.000	3.848
Mars orbit (MOO 1)	F	25.140	0.000	0.000	0.000
Mars orbit (MOO 2)	F	40.698	0.000	0.000	0.000
RelayComSat(s)	F	0.000	0.000	0.000	0.000
MarsSciSat(s)	F	0.000	0.000	0.000	0.000
Ph/D teleoperator(s)	F	0.000	0.000	0.000	0.000
Teleoperated MRSR(s)	F	0.000	0.000	0.000	0.000
TEI	G	91.746	0.000	0.000	0.000
Earth transfer (MCC, RCS)	H	0.000	1.416	0.000	0.861
ECCV (EOCS)	I	0.000	0.401	0.000	0.000
	Totals	1507.704	5.251	0.000	8.100

*Includes boiloff and reserves for Isp, delta V, and bulk propellant margins.

Human Mission
Propellant Reserves* Report

Mission: BCC-HQ
L06.Hum8c.SpDsm.ChHO.MAb.1D4cP.15s2R.ChHO.ECCVo
L05.Car.Cn.ChHO.MAb

Reference mission: BCC-HQ *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-2R Ref., Miss 2. (Sprint '06, 485/470/320, MAb, 2t rad shield)
PhEV to Phobos, 1 MRSR. ECCV into LEO.

		Propellant Reserve (t)			
		Main		ACS/RCS	
Period	Mission phase	Cryo	Biprop	Cryo	Biprop
TMI stage 1	C	43.173	0.000	0.000	0.000
TMI stage 2	C	19.107	0.000	0.000	0.000
Mars transfer (DSM)	D	4.504	0.000	0.000	0.000
Mars transfer (MCC, RCS)	D	0.000	0.101	0.000	0.100
MOC	E	0.014	0.000	0.000	0.000
Mars orbit (RCS)	F	0.000	0.000	0.000	0.114
Mars orbit (MOO 1)	F	0.792	0.000	0.000	0.000
Mars orbit (MOO 2)	F	1.265	0.000	0.000	0.000
RelayComSat(s)	F	0.000	0.000	0.000	0.000
MarsSciSat(s)	F	0.000	0.000	0.000	0.000
Ph/D teleoperator(s)	F	0.000	0.000	0.000	0.000
Teleoperated MRSR(s)	F	0.000	0.000	0.000	0.000
TEI	G	5.552	0.000	0.000	0.000
Earth transfer (MCC, RCS)	H	0.000	0.042	0.000	0.025
ECCV (EOCS)	I	0.000	0.012	0.000	0.000
	Totals	74.408	0.155	0.000	0.239

*Reserves for Isp, delta V, and bulk propellant margins.

Human Mission
Cryo-propellant Boiloff Report

Mission: BCC-HQ
L06.Hum8c.SpDsm.ChHO.MAb.1D4cP.15s2R.ChHO.ECCVo
L05.Car.Cn.ChHO.MAb

Reference mission: BCC-HQ *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-2R Ref., Miss 2. (Sprint '06, 485/470/320, MAb, 2t rad shield)
PhEV to Phobos, 1 MRSR. ECCV into LEO.

		Cryo-propellant Boiloff (t)	
Period	Mission phase	Main	ACS/RCS
TMI stage 1	C	8.140	0.000
TMI stage 2	C	3.603	0.000
Mars transfer (DSM)	D	1.947	0.000
Mars transfer (MCC, RCS)	D	0.000	0.000
MOC	E	0.011	0.000
Mars orbit (RCS)	F	0.000	0.000
Mars orbit (MOO 1)	F	0.648	0.000
Mars orbit (MOO 2)	F	1.026	0.000
RelayComSat(s)	F	0.000	0.000
MarsSciSat(s)	F	0.000	0.000
Ph/D teleoperator(s)	F	0.000	0.000
Teleoperated MRSR(s)	F	0.000	0.000
TEI	G	8.075	0.000
Earth transfer (MCC, RCS)	H	0.000	0.000
ECCV (EOCS)	I	0.000	0.000
	Totals	23.449	0.000

Human Mission Astrodynamics Report

Mission: BCC-HQ
L06.Hum8c.SpDsm.ChHO.MAb.1D4cP.15s2R.ChHO.ECCVo
L05.Car.Cn.ChHO.MAb

Reference mission: BCC-HQ *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-2R Ref., Miss 2. (Sprint '06, 485/470/320, MAb, 2t rad shield)
PhEV to Phobos, 1 MRSR. ECCV into LEO.

Trajectory Type: Sprint with deep space maneuver(s)

	Date	Apoapsis (km)	Periapsis (km)	Inclination (deg)	Vinf (km/s)	C3 (km^2/s^2)
Earth departure (TMI)	31-Dec-06	500.0	500.0	28.50	8.419	70.88
Mars arrival (MOC)	17-Aug-07	33840.0	250.0	45.00	6.822	46.54
Mars departure (TEI)	16-Sep-07	33840.0	250.0	45.00	5.428	29.46
Earth arrival (EOC)	15-Mar-08	500.0	500.0	28.50	3.477	12.09

Duration

	Days	Sols	Months
Marsbound (ETM)	229.0		7.52
Mars orbit	30.0	29.2	0.99
Mars surface			
MDV1		15.0	
Earthbound (MTE)	181.0		5.95
Total trip	440.0		14.46

Delta V Summary

Item	Delta V (km/s)
TMI	6.055
ETM DSM	1.775
ETM MCC	0.050
RCS ETM	0.030
MOC	3.738
MOO 1	0.719
MOO 2	0.619
RCS MOO	0.070
MDV 1 deorbit	0.000
MDV 1 terminal descent	0.150
MAV 1 ascent	5.408
MAV 1 RCS	0.050
TEI	2.648
MTE DSM	0.000
MTE MCC	0.050
RCS MTE	0.030
EOC	3.703

Mars Descent and Ascent Vehicles Mass Allocation Report

Mission: BCC-HQ
L06.Hum8c.SpDsm.ChHO.MAb.1D4cP.15s2R.ChHO.ECCVo
L05.Car.Cn.ChHO.MAb

Reference mission: BCC-HQ *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-2R Ref., Miss 2. (Sprint '06, 485/470/320, MAb, 2t rad shield)
PhEV to Phobos, 1 MRSR. ECCV into LEO.

	Mass (t)			
MDV 1 (w/o crew)				51.12
MELS			10.87	
Mars entry Ab		2.38		
Propulsion, deorbit+descent		4.63		
Parachute		3.86		
Adapter structure (incl. landing legs)			3.69	
Landed P/L (w/o crew)			13.50	
MLMM		5.63		
MLOE		6.57		
MLSE	2.30			
MLTE (RVRs, suits)	1.02			
Teleoperated equip.	3.00			
MLCE	0.15			
MLME (ISRU demo's)	0.10			
Crew*		(0.32)		
Mars landed expendables (MLX)		1.30		
Crew consumables	0.30			
other	1.00			
MAV (w/o crew)			23.06	
MAV (w/o crew)				23.06
Payload			1.24	
(Crew+returnables)*		(0.47)		
MAV Crew consumables + suits		0.19		
Inert module		0.97		
RCS propulsion		0.08		
Ascent propulsion			21.82	
MLMM				5.63
Structure			2.80	
Pressure shell, support structure		2.80		
Partitions, equipment acceptors		0.00		
Windows		0.00		
Man-systems			0.50	
Living quarters		0.30		
Galley		0.10		
Personal hygiene		0.10		
ECLSS			0.79	
DMS			0.05	
EPS			0.54	
TCS			0.70	
Communications			0.25	

* Used only to size propulsion. Crew+returnables allocated in ECCV.

Technology Status

Propulsion system masses	
LH2/LOX	Nominal
Stored biprop	Nominal
MELS aerobrake scaling	Nominal
MDV adapter scaling	Nominal
MAV inert module mass	Nominal

Cargo Mission
Mass Allocation Report

Mission: BCC-HQ
L06.Hum8c.SpDsm.ChHO.MAb.1D4cP.15s2R.ChHO.ECCVo
L05.Car.Cn.ChHO.MAb

Reference mission: BCC-HQ *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-2R Ref., Miss 2. (Sprint '06, 485/470/320, MAb, 2t rad shield)
PhEV to Phobos, 1 MRSR. ECCV into LEO.

	Mass (t)			
MCV (IMLEO)				741.79
TMIS			487.14	
Stage(s) to be dropped post-TMI		64.50		
TMI Propellant		422.65		
MTV			254.65	
MTV				254.65
Mars transfer expendables (MTX)			8.22	
MCC propellant		4.08		
RCS ETM propellant		4.15		
ETM MCC prop. sys. (dry)			0.00	
MOCS			25.91	
Mars capture Ab		22.05		
Prop. sys. (dry)		1.20		
Propellant		0.66		
other MOCS		2.00		
MOV (F1)			220.52	
MOV (F1)				220.52
Structure			0.50	
Mars orbit expendables (MOX)			39.26	
MOO propellant		32.31		
RCS MOO propellant		6.94		
MOO prop. sys. (dry)			5.95	
RCS ETM/MOO prop. sys. (dry)			1.14	
Support Services			0.21	
Data management system (DMS)		0.03		
Electrical power system (EPS)		0.05		
Thermal control system (TCS)		0.10		
Communications system		0.03		
Payload			173.47	
Satellites		2.00		
RelayComSat(s)	1.00			
MarsSciSat(s)	1.00			
Ph/D teleoperator(s)	0.00			
Teleoperated MRSR		3.50		
MOSE		0.15		
ISE		0.30		
Solar/SPE monitoring	0.20			
Astro/Planetary	0.10			
MDV(s)		51.12		
Phobos EV		9.79		

TEIS		106.61
Stage	14.86	
Propellant	91.75	

Technology Status

Propulsion system masses	
LH2/LOX	Nominal
Stored biprop	Nominal
Mars aerobrake scaling	Nominal

Cargo Mission
Dry* Masses Deployed or Jettisoned Report

Mission: BCC-HQ
L06.Hum8c.SpDsm.ChHO.MAb.1D4cP.15s2R.ChHO.ECCVo
L05.Car.Cn.ChHO.MAb

Reference mission: BCC-HQ *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-2R Ref., Miss 2. (Sprint '06, 485/470/320, MAb, 2t rad shield)
PhEV to Phobos, 1 MRSR. ECCV into LEO.

What	When	Mission phase	Mass (t)
TMIS stage	during TMI	C	64.50
ETM MCC prop. sys.	pre-MOC	E	0.00
Mars Ab	post-MOC	E	22.05
MOC prop. sys.	post-MOC	E	1.20
Other MOCS	post-MOC	E	2.00
RelayComSat(s)	Mars orbit	F1	1.00
MarsSciSat(s)	Mars orbit	F1	1.00
Ph/D teleoperator(s)	Mars orbit	F1	0.00
Teleoperated MRSR(s)	Mars orbit	F1	3.50
MDV 1 (without MAV 1)	Mars orbit	F2	22.92
MAV 1	Mars orbit	F5	3.08
Phobos EV	Mars orbit	F1	9.79
MOO prop. sys.	Mars orbit	F5	5.95
MOV, pre-human TEI	Mars orbit	F5	2.30
		Total	139.29

*Not included:
propellants, consumables, crew mass, spacesuits, other expendables

Cargo Mission
Total Propellant* Report

Mission: BCC-HQ
L06.Hum8c.SpDsm.ChHO.MAb.1D4cP.15s2R.ChHO.ECCVo
L05.Car.Cn.ChHO.MAb

Reference mission: BCC-HQ *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-2R Ref., Miss 2. (Sprint '06, 485/470/320, MAb, 2t rad shield)
PhEV to Phobos, 1 MRSR. ECCV into LEO.

		Propellant Mass Expended (t)			
		Main		ACS/RCS	
Period	Mission phase	Cryo	Biprop	Cryo	Biprop
TMI	C	422.645	0.000	0.000	0.000
Mars transfer	D	0.000	4.078	0.000	4.146
MOC	E	0.656	0.000	0.000	0.000
Mars orbit operations	F	32.311	0.000	0.000	6.945
RelayComSat(s)	F	0.000	0.000	0.000	0.000
MarsSciSat(s)	F	0.000	0.000	0.000	0.000
Ph/D teleoperator(s)	F	0.000	0.000	0.000	0.000
Teleoperated MRSR(s)	F	0.000	0.000	0.000	0.000
MDV 1 deorbit	F2a	0.000	0.247	0.000	0.000
MDV 1 terminal descent	F2b	0.000	3.265	0.000	0.000
MAV 1	F4	0.000	19.764	0.000	0.028
	Totals	455.612	27.354	0.000	11.118

*Includes boiloff and reserves for Isp, delta V, and bulk propellant margins.

Cargo Mission
Propellant Reserves* Report

Mission: BCC-HQ
L06.Hum8c.SpDsm.ChHO.MAb.1D4cP.15s2R.ChHO.ECCVo
L05.Car.Cn.ChHO.MAb

Reference mission: BCC-HQ *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-2R Ref., Miss 2. (Sprint '06, 485/470/320, MAb, 2t rad shield)
PhEV to Phobos, 1 MRSR. ECCV into LEO.

		Propellant Reserve (t)			
		Main		ACS/RCS	
Period	Mission phase	Cryo	Biprop	Cryo	Biprop
TMI	C	25.290	0.000	0.000	0.000
Mars transfer	D	0.000	0.120	0.000	0.122
MOC	E	0.019	0.000	0.000	0.000
Mars orbit operations	F	1.016	0.000	0.000	0.207
RelayComSat(s)	F	0.000	0.000	0.000	0.000
MarsSciSat(s)	F	0.000	0.000	0.000	0.000
Ph/D teleoperator(s)	F	0.000	0.000	0.000	0.000
Teleoperated MRSR(s)	F	0.000	0.000	0.000	0.000
MDV 1 deorbit	F2a	0.000	0.007	0.000	0.000
MDV 1 terminal descent	F2b	0.000	0.062	0.000	0.000
MAV 1	F4	0.000	3.314	0.000	0.001
	Totals	26.325	3.504	0.000	0.330

*Reserves for Isp, delta V, and bulk propellant margins.

Cargo Mission
Cryo-propellant Boiloff Report

Mission: BCC-HQ
L06.Hum8c.SpDsm.ChHO.MAb.1D4cP.15s2R.ChHO.ECCVo
L05.Car.Cn.ChHO.MAb

Reference mission: BCC-HQ *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-2R Ref., Miss 2. (Sprint '06, 485/470/320, MAb, 2t rad shield)
PhEV to Phobos, 1 MRSR. ECCV into LEO.

		Cryo-propellant Boiloff (t)	
Period	Mission phase	Main	ACS/RCS
TMI	C	5.097	0.000
Mars transfer	D	0.000	0.000
MOC	E	0.024	0.000
Mars orbit operations	F	1.690	0.000
RelayComSat(s)	F	0.000	0.000
MarsSciSat(s)	F	0.000	0.000
Ph/D teleoperator(s)	F	0.000	0.000
Teleoperated MRSR(s)	F	0.000	0.000
MDV 1 deorbit	F2a	0.000	0.000
MDV 1 terminal descent	F2b	0.000	0.000
MAV 1	F4	0.000	0.000
	Totals	6.811	0.000

Cargo Mission
Astrodynamics Report

Mission: BCC-HQ
L06.Hum8c.SpDsm.ChHO.MAb.1D4cP.15s2R.ChHO.ECCVo
L05.Car.Cn.ChHO.MAb

Reference mission: BCC-HQ *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-2R Ref., Miss 2. (Sprint '06, 485/470/320, MAb, 2t rad shield)
PhEV to Phobos, 1 MRSR. ECCV into LEO.

Trajectory Type: Conjunction

	Date	Apoapsis (km)	Periapsis (km)	Inclination (deg)	Vinf (km/s)	C3 (km^2/s^2)
Earth departure (TMI)	1-Sep-05	500.0	500.0	28.50	3.930	15.44
Mars arrival (MOC)	8-Oct-06	33840.0	250.0	45.00	3.507	12.30

Duration

	Days	Sols	Months
Marsbound (ETM)	402.0		13.21
Mars orbit	344.5	334.9	11.32

Delta V Summary

Item	Delta V (km/s)
TMI	3.850
ETM MCC	0.050
RCS ETM	0.050
MOC	1.355
MOO	0.719
RCS MOO	0.100

Human Mission
Mass Allocation Report

Mission: JBM-HH
L09.Hum8c.SpVsDsm.ChHO.MAb.1D4cP.15s2R.ChHO.ECCVo
L07.Car.Cn.ChHO.MAb

Reference mission: JBM-HH *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-3R Ref., Miss 3. (Sprint '09, 485/470/320, MAb, 2t rad shield)
PhEVs to Phobos and Deimos, 1 MRSR. ECCV into LEO.

	Mass (t)		
MSS (IMLEO)			1511.95
TMIS		1183.45	
Stage(s) to be dropped post-TMI	156.28		
TMI Propellant	1027.18		
MTV		328.50	
MTV			328.50
Mars transfer expendables (MTX)		92.38	
Crew consumables	9.36		
DSM propellant	75.94		
MCC propellant	3.86		
RCS ETM propellant	3.22		
ETM DSM prop. sys. (dry)		12.49	
Venus swingby probes(s)		0.00	
MOCS		32.43	
Mars capture Ab	28.68		
Prop. sys. (dry)	1.18		
Propellant	0.56		
other MOCS	2.00		
MOV (F1)		191.20	
MOV (F1)			191.20
Mars orbit expendables (MOX)		89.87	
Crew consumables	0.88		
MOO 1 propellant	28.40		
MOO 2 propellant	56.25		
RCS MOO propellant	4.34		
MOO prop. sys. (dry)		13.79	
Satellites		0.00	
RelayComSat(s)	0.00		
MarsSciSat(s)	0.00		
Ph/D teleoperator(s)	0.00		
Teleoperated MRSR		0.00	
MOSE		0.00	
MOV (Ff)		285.56	
TEIS (received from MCV)		-198.01	
MOV (Ff)			285.56
TEIS		198.01	
Stage to be dropped post-TEI	26.78		
TEI propellant	171.23		
ETV		87.54	

ETV			87.54
IMM		64.85	
External Services (power, com, thermal)		1.05	
Earth transfer expendables (ETX)		9.05	
Crew consumables	6.78		
Flyaround propellant	0.00		
MCC propellant	1.41		
RCS MTE propellant	0.86		
ETM/MTE MCC prop. sys. (dry)		1.23	
RCS ETM/MOO/MTE prop. sys. (dry)		0.99	
Spacesuits		0.70	
ISE		0.50	
Solar/SPE monitoring	0.40		
Astro/Planetary	0.10		
ECCV		9.18	
ECCV			9.18
Payload		6.82	
Crew+returnables+consum+suits	1.11		
Inert module	5.71		
EOCS		2.36	
Ab	0.68		
Propulsion	1.33		
Other EOCS	0.34		
IMM			64.85
Cylindrical Module(s)		51.00	
Disk Module(s)		2.50	
Tunnel(s)		2.70	
Resource Nodes (docking, prox ops)		0.00	
Airlock(s)		0.95	
Radiation shelter shielding		2.00	
Life support system (LSS)		2.80	
Data management system (DMS)		0.30	
Internal Com/EPS/TCS		2.60	
External Services			1.05
Electrical power system (EPS), external		0.40	
Thermal control system (TCS), external		0.50	
Communications system, external		0.15	

Technology Status

Propulsion system masses	
LH2/LOX	Nominal
Stored biprop	Nominal
Mars aerobrake scaling	V. conservative

Human Mission
Crew Consumables* Report

Mission: JBM-HH
L09.Hum8c.SpVsDsm.ChHO.MAb.1D4cP.15s2R.ChHO.ECCVo
L07.Car.Cn.ChHO.MAb

Reference mission: JBM-HH *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-3R Ref., Miss 3. (Sprint '09, 485/470/320, MAb, 2t rad shield)
PhEVs to Phobos and Deimos, 1 MRSR. ECCV into LEO.

Crew composition: Nominal U. S. male crew

Period	Mission phase	# of crew	Time	Person-days	Margin	Total mass (t)
LEO Checkout	A	8	21 day	168	20 %	0.83
MTV	D	8	238 day	1903	20 %	9.36
MOV	F		30 day	178	20 %	0.88
MDV 1	F3	4	15 sol	62	5 %	0.30
MAV 1	F4	4	1 day	4	200 %	0.07
ETV	H	8	172 day	1377	20 %	6.78
ECCV	I	8	1 day	8	200 a %	0.13
					Total (incl. margin)	18.34
					Total (w/o margin)	15.22

Total supply = 10.13 person-years = 3700 person-days
Average supply = 4.96 kg/person-day

Consumables Baseline (nominal U. S. gender-mixed crew, kg/person-day):

	Food	Water	Other	Total
Spaceborne	1.5	0.5	2.1	4.1
Surface	1.5	2.5	0.7	4.7
MAV, ECCV	1.5	2.0	2.0	5.5

* Consumables includes LSS + Food
(a) To provide interplanetary safe-haven capability.

Human Mission
Dry* Masses Deployed or Jettisoned Report

Mission: JBM-HH
L09.Hum8c.SpVsDsm.ChHO.MAb.1D4cP.15s2R.ChHO.ECCVo
L07.Car.Cn.ChHO.MAb

Reference mission: JBM-HH *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-3R Ref., Miss 3. (Sprint '09, 485/470/320, MAb, 2t rad shield)
PhEVs to Phobos and Deimos, 1 MRSR. ECCV into LEO.

What	When	Mission phase	Mass (t)
TMIS stage 1	during TMI	C	106.16
TMIS stage 2	during TMI	C	50.11
Venus probes	Venus swingby	D	0.00
ETM DSM prop. sys.	Mars transfer	D	12.49
Mars Ab	post-MOC	E	28.68
MOC prop. sys.	post-MOC	E	1.18
Other MOCS	post-MOC	E	2.00
RelayComSat(s)	Mars orbit	F1	0.00
MarsSciSat(s)	Mars orbit	F1	0.00
Ph/D teleoperator(s)	Mars orbit	F1	0.00
Teleoperated MRSR(s)	Mars orbit	F1	0.00
MOO prop. sys.	Mars orbit	F5	13.79
TEI stage	Post TEI	G	26.78
ECCV	pre-EOC	I	7.66
ETV	pre-EOC	I	68.62
		Total	317.49

*Not included:
propellants, consumables, crew mass, spacesuits, other expendables

Human Mission
Total Propellant* Report

Mission: JBM-HH
L09.Hum8c.SpVsDsm.ChHO.MAb.1D4cP.15s2R.ChHO.ECCVo
L07.Car.Cn.ChHO.MAb

Reference mission: JBM-HH *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-3R Ref., Miss 3. (Sprint '09, 485/470/320, MAb, 2t rad shield)
PhEVs to Phobos and Deimos, 1 MRSR. ECCV into LEO.

		Propellant Mass Expended (t)			
		Main		ACS/RCS	
Period	Mission phase	Cryo	Biprop	Cryo	Biprop
TMI stage 1	C	700.410	0.000	0.000	0.000
TMI stage 2	C	326.766	0.000	0.000	0.000
Mars transfer (DSM)	D	75.940	0.000	0.000	0.000
Mars transfer (MCC, RCS)	D	0.000	3.856	0.000	3.219
MOC	E	0.561	0.000	0.000	0.000
Mars orbit (RCS)	F	0.000	0.000	0.000	4.344
Mars orbit (MOO 1)	F	28.404	0.000	0.000	0.000
Mars orbit (MOO 2)	F	56.250	0.000	0.000	0.000
RelayComSat(s)	F	0.000	0.000	0.000	0.000
MarsSciSat(s)	F	0.000	0.000	0.000	0.000
Ph/D teleoperator(s)	F	0.000	0.000	0.000	0.000
Teleoperated MRSR(s)	F	0.000	0.000	0.000	0.000
TEI	G	171.229	0.000	0.000	0.000
Earth transfer (MCC, RCS)	H	0.000	1.411	0.000	0.858
ECCV (EOCS)	I	0.000	0.401	0.000	0.000
	Totals	1359.559	5.668	0.000	8.421

*Includes boiloff and reserves for Isp, delta V, and bulk propellant margins.

Human Mission
Propellant Reserves* Report

Mission: JBM-HH
L09.Hum8c.SpVsDsm.ChHO.MAb.1D4cP.15s2R.ChHO.ECCVo
L07.Car.Cn.ChHO.MAb

Reference mission: JBM-HH *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-3R Ref., Miss 3. (Sprint '09, 485/470/320, MAb, 2t rad shield)
PhEVs to Phobos and Deimos, 1 MRSR. ECCV into LEO.

		Propellant Reserve (t)			
		Main		ACS/RCS	
Period	Mission phase	Cryo	Biprop	Cryo	Biprop
TMI stage 1	C	34.042	0.000	0.000	0.000
TMI stage 2	C	15.882	0.000	0.000	0.000
Mars transfer (DSM)	D	2.695	0.000	0.000	0.000
Mars transfer (MCC, RCS)	D	0.000	0.114	0.000	0.095
MOC	E	0.016	0.000	0.000	0.000
Mars orbit (RCS)	F	0.000	0.000	0.000	0.129
Mars orbit (MOO 1)	F	0.894	0.000	0.000	0.000
Mars orbit (MOO 2)	F	1.747	0.000	0.000	0.000
RelayComSat(s)	F	0.000	0.000	0.000	0.000
MarsSciSat(s)	F	0.000	0.000	0.000	0.000
Ph/D teleoperator(s)	F	0.000	0.000	0.000	0.000
Teleoperated MRSR(s)	F	0.000	0.000	0.000	0.000
TEI	G	13.017	0.000	0.000	0.000
Earth transfer (MCC, RCS)	H	0.000	0.042	0.000	0.025
ECCV (EOCS)	I	0.000	0.012	0.000	0.000
	Totals	68.293	0.168	0.000	0.249

*Reserves for Isp, delta V, and bulk propellant margins.

Human Mission
Cryo-propellant Boiloff Report

Mission: JBM-HH
L09.Hum8c.SpVsDsm.ChHO.MAb.1D4cP.15s2R.ChHO.ECCVo
L07.Car.Cn.ChHO.MAb

Reference mission: JBM-HH *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-3R Ref., Miss 3. (Sprint '09, 485/470/320, MAb, 2t rad shield)
PhEVs to Phobos and Deimos, 1 MRSR. ECCV into LEO.

Period	Mission phase	Cryo-propellant Boiloff (t) Main	ACS/RCS
TMI stage 1	C	6.327	0.000
TMI stage 2	C	2.952	0.000
Mars transfer (DSM)	D	1.152	0.000
Mars transfer (MCC, RCS)	D	0.000	0.000
MOC	E	0.013	0.000
Mars orbit (RCS)	F	0.000	0.000
Mars orbit (MOO 1)	F	0.759	0.000
Mars orbit (MOO 2)	F	1.470	0.000
RelayComSat(s)	F	0.000	0.000
MarsSciSat(s)	F	0.000	0.000
Ph/D teleoperator(s)	F	0.000	0.000
Teleoperated MRSR(s)	F	0.000	0.000
TEI	G	20.306	0.000
Earth transfer (MCC, RCS)	H	0.000	0.000
ECCV (EOCS)	I	0.000	0.000
	Totals	32.978	0.000

Boiloff rates

Low Earth orbit	0.150 %/month
Interplanetary ~~(at 1 AU)~~	0.300 %/month
Mars orbit	0.065 %/month

Human Mission Astrodynamics Report

Mission: JBM-HH
L09.Hum8c.SpVsDsm.ChHO.MAb.1D4cP.15s2R.ChHO.ECCVo
L07.Car.Cn.ChHO.MAb

Reference mission: JBM-HH *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-3R Ref., Miss 3. (Sprint '09, 485/470/320, MAb, 2t rad shield)
PhEVs to Phobos and Deimos, 1 MRSR. ECCV into LEO.

Trajectory Type: Sprint with Venus swingby(s) and deep space maneuver(s)

	Date	Apoapsis (km)	Periapsis (km)	Inclination (deg)	Vinf (km/s)	C3 (km^2/s^2)
Earth departure (TMI)	6-Feb-09	500.0	500.0	28.50	7.818	61.12
Venus swingby	30-Mar-09					
Mars arrival (MOC)	2-Oct-09	33840.0	250.0	45.00	6.970	48.58
Mars departure (TEI)	1-Nov-09	33840.0	250.0	45.00	6.850	46.92
Earth arrival (EOC)	22-Apr-10	500.0	500.0	28.50	4.325	18.71

Duration	Days	Sols	Months
Marsbound (ETM)	237.8		7.81
Mars orbit	30.0	29.2	0.99
Mars surface			
MDV1		15.0	
Earthbound (MTE)	172.2		5.66
Total trip	440.0		14.46

Delta V Summary

Item	Delta V (km/s)
TMI	5.693
ETM DSM	1.173
ETM MCC	0.050
RCS ETM	0.030
MOC	3.859
MOO 1	0.719
MOO 2	0.619
RCS MOO	0.070
MDV 1 deorbit	0.000
MDV 1 terminal descent	0.150
MAV 1 ascent	5.408
MAV 1 RCS	0.050
TEI	3.761
MTE DSM	0.000
MTE MCC	0.050
RCS MTE	0.030
EOC	3.992

Mars Descent and Ascent Vehicles
Mass Allocation Report

Mission: JBM-HH
L09.Hum8c.SpVsDsm.ChHO.MAb.1D4cP.15s2R.ChHO.ECCVo
L07.Car.Cn.ChHO.MAb

Reference mission: JBM-HH *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-3R Ref., Miss 3. (Sprint '09, 485/470/320, MAb, 2t rad shield)
PhEVs to Phobos and Deimos, 1 MRSR. ECCV into LEO.

	Mass (t)			
MDV 1 (w/o crew)				51.12
MELS			10.87	
Mars entry Ab		2.38		
Propulsion, deorbit+descent		4.63		
Parachute		3.86		
Adapter structure (incl. landing legs)			3.69	
Landed P/L (w/o crew)			13.50	
MLMM		5.63		
MLOE		6.57		
MLSE	2.30			
MLTE (RVRs, suits)	1.02			
Teleoperated equip.	3.00			
MLCE	0.15			
MLME (ISRU demo's)	0.10			
Crew*		(0.32)		
Mars landed expendables (MLX)		1.30		
Crew consumables	0.30			
other	1.00			
MAV (w/o crew)			23.06	
MAV (w/o crew)				23.06
Payload			1.24	
(Crew+returnables)*		(0.47)		
MAV Crew consumables + suits		0.19		
Inert module		0.97		
RCS propulsion		0.08		
Ascent propulsion			21.82	
MLMM				5.63
Structure			2.80	
Pressure shell, support structure		2.80		
Partitions, equipment acceptors		0.00		
Windows		0.00		
Airlock (AL)			0.00	
Hyperbolic airlock (HAL)			0.00	
Man-systems			0.50	
Living quarters		0.30		
Galley		0.10		
Personal hygiene		0.10		
ECLSS			0.79	
DMS			0.05	
EPS			0.54	

TCS	0.70
Communications	0.25

* Used only to size propulsion. Crew+returnables allocated in ECCV.

Technology Status

Propulsion system masses	
LH2/LOX	Nominal
Stored biprop	Nominal
MELS aerobrake scaling	Nominal
MDV adapter scaling	Nominal
MAV inert module mass	Nominal

Cargo Mission Mass Allocation Report

Mission: JBM-HH
L09.Hum8c.SpVsDsm.ChHO.MAb.1D4cP.15s2R.ChHO.ECCVo
L07.Car.Cn.ChHO.MAb

Reference mission: JBM-HH *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-3R Ref., Miss 3. (Sprint '09, 485/470/320, MAb, 2t rad shield)
PhEVs to Phobos and Deimos, 1 MRSR. ECCV into LEO.

	Mass (t)			
MCV (IMLEO)				1113.43
TMIS			716.06	
Stage(s) to be dropped post-TMI		94.36		
TMI Propellant		621.70		
MTV			397.37	
MTV				397.37
Mars transfer expendables (MTX)			12.83	
MCC propellant		6.36		
RCS ETM propellant		6.47		
ETM MCC prop. sys. (dry)			0.00	
MOCS			38.85	
Mars capture Ab		34.57		
Prop. sys. (dry)		1.25		
Propellant		1.02		
other MOCS		2.00		
MOV (F1)			345.69	
MOV (F1)				345.69
Structure			0.50	
Mars orbit expendables (MOX)			62.01	
MOO propellant		51.03		
RCS MOO propellant		10.99		
MOO prop. sys. (dry)			8.75	
RCS ETM/MOO prop. sys. (dry)			1.51	
Support Services			0.21	
Data management system (DMS)		0.03		
Electrical power system (EPS)		0.05		
Thermal control system (TCS)		0.10		
Communications system		0.03		
Payload			272.70	
Satellites		2.00		
RelayComSat(s)	1.00			
MarsSciSat(s)	1.00			
Ph/D teleoperator(s)	0.00			
Teleoperated MRSR		3.50		
MOSE		0.15		
ISE		0.30		
Solar/SPE monitoring	0.20			
Astro/Planetary	0.10			
MDV(s)		51.12		
Phobos EV		9.79		

Deimos EV		7.83
TEIS		198.01
Stage	26.78	
Propellant	171.23	

Technology Status

Propulsion system masses	
LH2/LOX	Nominal
Stored biprop	Nominal
Mars aerobrake scaling	Nominal

Cargo Mission
Dry* Masses Deployed or Jettisoned Report

Mission: JBM-HH
L09.Hum8c.SpVsDsm.ChHO.MAb.1D4cP.15s2R.ChHO.ECCVo
L07.Car.Cn.ChHO.MAb

Reference mission: JBM-HH *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-3R Ref., Miss 3. (Sprint '09, 485/470/320, MAb, 2t rad shield)
PhEVs to Phobos and Deimos, 1 MRSR. ECCV into LEO.

What	When	Mission phase	Mass (t)
TMIS stage	during TMI	C	94.36
ETM MCC prop. sys.	pre-MOC	E	0.00
Mars Ab	post-MOC	E	34.57
MOC prop. sys.	post-MOC	E	1.25
Other MOCS	post-MOC	E	2.00
RelayComSat(s)	Mars orbit	F1	1.00
MarsSciSat(s)	Mars orbit	F1	1.00
Ph/D teleoperator(s)	Mars orbit	F1	0.00
Teleoperated MRSR(s)	Mars orbit	F1	3.50
MDV 1 (without MAV 1)	Mars orbit	F2	22.92
MAV 1	Mars orbit	F5	3.08
Phobos EV	Mars orbit	F1	9.79
Deimos EV	Mars orbit	F1	7.83
MOO prop. sys.	Mars orbit	F5	8.75
MOV, pre-human TEI	Mars orbit	F5	2.67
		Total	192.73

*Not included:
propellants, consumables, crew mass, spacesuits, other expendables

Cargo Mission
Total Propellant* Report

Mission: JBM-HH
L09.Hum8c.SpVsDsm.ChHO.MAb.1D4cP.15s2R.ChHO.ECCVo
L07.Car.Cn.ChHO.MAb

Reference mission: JBM-HH *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-3R Ref., Miss 3. (Sprint '09, 485/470/320, MAb, 2t rad shield)
PhEVs to Phobos and Deimos, 1 MRSR. ECCV into LEO.

		Propellant Mass Expended (t)			
		Main		ACS/RCS	
Period	Mission phase	Cryo	Biprop	Cryo	Biprop
TMI	C	621.702	0.000	0.000	0.000
Mars transfer	D	0.000	6.364	0.000	6.469
MOC	E	1.024	0.000	0.000	0.000
Mars orbit operations	F	51.025	0.000	0.000	10.987
RelayComSat(s)	F	0.000	0.000	0.000	0.000
MarsSciSat(s)	F	0.000	0.000	0.000	0.000
Ph/D teleoperator(s)	F	0.000	0.000	0.000	0.000
Teleoperated MRSR(s)	F	0.000	0.000	0.000	0.000
MDV 1 deorbit	F2a	0.000	0.247	0.000	0.000
MDV 1 terminal descent	F2b	0.000	3.265	0.000	0.000
MAV 1	F4	0.000	19.764	0.000	0.028
	Totals	673.751	29.640	0.000	17.484

*Includes boiloff and reserves for Isp, delta V, and bulk propellant margins.

Cargo Mission
Propellant Reserves* Report

Mission: JBM-HH
L09.Hum8c.SpVsDsm.ChHO.MAb.1D4cP.15s2R.ChHO.ECCVo
L07.Car.Cn.ChHO.MAb

Reference mission: JBM-HH *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-3R Ref., Miss 3. (Sprint '09, 485/470/320, MAb, 2t rad shield)
PhEVs to Phobos and Deimos, 1 MRSR. ECCV into LEO.

		Propellant Reserve (t)			
		Main		ACS/RCS	
Period	Mission phase	Cryo	Biprop	Cryo	Biprop
TMI	C	36.262	0.000	0.000	0.000
Mars transfer	D	0.000	0.188	0.000	0.191
MOC	E	0.029	0.000	0.000	0.000
Mars orbit operations	F	1.608	0.000	0.000	0.328
RelayComSat(s)	F	0.000	0.000	0.000	0.000
MarsSciSat(s)	F	0.000	0.000	0.000	0.000
Ph/D teleoperator(s)	F	0.000	0.000	0.000	0.000
Teleoperated MRSR(s)	F	0.000	0.000	0.000	0.000
MDV 1 deorbit	F2a	0.000	0.007	0.000	0.000
MDV 1 terminal descent	F2b	0.000	0.062	0.000	0.000
MAV 1	F4	0.000	3.314	0.000	0.001
	Totals	37.899	3.571	0.000	0.520

*Reserves for Isp, delta V, and bulk propellant margins.

Cargo Mission
Cryo-propellant Boiloff Report

Mission: JBM-HH
L09.Hum8c.SpVsDsm.ChHO.MAb.1D4cP.15s2R.ChHO.ECCVo
L07.Car.Cn.ChHO.MAb

Reference mission: JBM-HH *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-3R Ref., Miss 3. (Sprint '09, 485/470/320, MAb, 2t rad shield)
PhEVs to Phobos and Deimos, 1 MRSR. ECCV into LEO.

Period	Mission phase	Cryo-propellant Boiloff (t) Main	ACS/RCS
TMI	C	7.237	0.000
Mars transfer	D	0.000	0.000
MOC	E	0.035	0.000
Mars orbit operations	F	2.561	0.000
RelayComSat(s)	F	0.000	0.000
MarsSciSat(s)	F	0.000	0.000
Ph/D teleoperator(s)	F	0.000	0.000
Teleoperated MRSR(s)	F	0.000	0.000
MDV 1 deorbit	F2a	0.000	0.000
MDV 1 terminal descent	F2b	0.000	0.000
MAV 1	F4	0.000	0.000
	Totals	9.833	0.000

Boiloff rates

Low Earth orbit	0.150 %/month
Interplanetary ~~(at 1 AU)~~	0.300 %/month
Mars orbit	0.065 %/month

Cargo Mission Astrodynamics Report

Mission: JBM-HH
L09.Hum8c.SpVsDsm.ChHO.MAb.1D4cP.15s2R.ChHO.ECCVo
L07.Car.Cn.ChHO.MAb

Reference mission: JBM-HH *Trajectory file:* TRJ.6/30.02.Sp.Ab

Mission purpose:
CS-2, TIC-3R Ref., Miss 3. (Sprint '09, 485/470/320, MAb, 2t rad shield)
PhEVs to Phobos and Deimos, 1 MRSR. ECCV into LEO.

Trajectory Type: Conjunction

	Date	Apoapsis (km)	Periapsis (km)	Inclination (deg)	Vinf (km/s)	C3 (km^2/s^2)
Earth departure (TMI)	22-Sep-07	500.0	500.0	28.50	3.570	12.74
Mars arrival (MOC)	25-Sep-08	33840.0	250.0	45.00	2.829	8.00

Duration

	Days	Sols	Months
Marsbound (ETM)	369.0		12.12
Mars orbit	403.5	392.2	13.26

Delta V Summary

Item	Delta V (km/s)
TMI	3.732
ETM MCC	0.050
RCS ETM	0.050
MOC	0.985
MOO	0.719
RCS MOO	0.100

Conclusions and Recommendations

6.1 Conclusions and Recommendations

The conclusions and recommendations for all of the case studies are given in Tables 6.1-1 through 6.1-4. Each table lists the issues, suggests solutions, and relates the lessons learned from the studies. Several problems are common to all of the case studies, such as the need for light-weight, low-boiloff tanks and for designs which are compatible with on-orbit assembly technology. Table 6.1-5 recommends a course of study for 1989. Not only does it suggest further exploration of the current case studies, but it also proposes several new ones.

Table 6.1-1 CS-1: Transportation Conclusions/Recommendations

Programmatics

Parallel development of
- Phobos Excursion Vehicle (PhEV) and EVA support equipment
- Earth Crew Capture Vehicle (ECCV)
- man-rated propulsion systems (TMIS/TEIS)
- spaceship (hab modules, external services)
- SS, HLLV, microgravity countermeasures

Enabling/Enhancing Technologies

Enabling:
- Designs which are compatible with on-orbit assembly technology
- From IMLEO standpoint, aerocapture may be considered to be enabling
- Station-keeping, MMU ops, and other aspects of human exploration of Phobos

Significantly enhancing:
- Advanced, light-weight and low-boiloff tanks; dual wet-dry launch (Siamese-

Interfacing

In-flight communications (Earth-to-Mars)
Phobos surface operations and requirements
Connection/requirements with overall transportation infrastructure

Table 6.1-1 (cont.) CS-1: Transportation Conclusions/Recommendations

Issues

Reliability
- HLLV
- Duplicate hardware builds (flight units #1, #2, etc.?)

Aerobrake technology availability

Mission contingencies
- Human safety
- Mission success

Precursors
- Transportation man-rating
- Technology demonstrations in-space (Ab, ECLSS, etc.)

Hazards of Split Missions

Lessons Learned

IMLEO sensitivity of various technology advancements
(example: relative leverage of advanced tankage vs advanced engines)

ΔV and IMLEO penalties for Phobos accessibility

Table 6.1-2 CS-2: Transportation Conclusions/Recommendations

Programmatics

Parallel development of
- spaceship (hab modules, external services)
- SS, HLLV, microgravity countermeasures
- Earth-orbit retrieval of ECCV

Flight demonstrations of
- planetary aerocapture
- man-rated TMIS/TEIS
- Mars descent and landing (esp. parachute assist)

Enabling/Enhancing Technologies

Enabling:
- Designs which are compatible with on-orbit assembly technology
- zero-gravity countermeasures effectiveness
- Mars aerobrake for aerocapture

Significantly enhancing:
- Advanced, light-weight and low-boiloff tanks; dual wet-dry launch (Siamese-
- Lightweight aerobraking (reusable brakes; ablators)

Interfacing

SS Servicing (OMV; construction crew; assembly)
In-flight communications (Earth-to-Mars)
Surface operations and requirements
Connection/requirements with overall transportation infrastructure

Table 6.1-2 (cont.) CS-2: Transportation Conclusions/Recommendations

Issues

- Reliability
  - HLLV
  - Duplicate hardware builds (flight units #1, #2, etc.?)
  - Abort modes
- Aerobrake technology timeliness
- Advanced Structures
- Mission contingencies
  - Human safety
  - Mission success
- Precursors
  - Transportation man-rating
  - Technology demonstrations in-space (Ab, ECLSS, etc.)
  - Mars missions, including sample return and landed nav-aides
- All-up vs additional risk of split missions
- Rendezvous in elliptical orbit

Lessons Learned

- IMLEO sensitivity to split missions vs all-up conjunction class missions
- Operations penalties and dangers of Phobos and/or Deimos missions
  (combined with landed missions)

Table 6.1-3 CS-3: Transportation Conclusions/Recommendations

Programmatics

Parallel development of
- Lunar Transfer Vehicle (LTV)
- Lunar Descent Vehicle - Piloted (LDV-P) and -Cargo (LDV-C)
- Observatory scientific instruments

Flight demonstrations of
- Earth aerocapture
- man-rated LTV

Enabling/Enhancing Technologies

Enabling:
- Designs which are compatible with on-orbit assembly technology
- On-orbit propellant transfer (for LTV refilling)

Significantly enhancing:
- Advanced, light-weight and low-boiloff tanks

Interfacing

SS Servicing
Surface operations and requirements
Connection/requirements with overall transportation infrastructure

Table 6.1-3 (cont.) CS-3: Transportation Conclusions/Recommendations

Issues

- Reliability
  - HLLV
  - Duplicate hardware builds (flight units #1, #2, etc.?)
  - Abort modes
- Aerobrake technology timeliness
- Mission contingencies
  - Human safety
  - Mission success
- Precursors
  - Transportation man-rating

Lessons Learned

This mission readily enabled by HLLV

Table 6.1-4 CS-4: Transportation Conclusions/Recommendations

Programmatics

Extensive development of
- Lunar LOX production plant
- space-rated nuclear thermal reactor
- large ion engines (or equivalent electric thruster)
- spaceship (hab modules, external services)
- SS, HLLV, microgravity countermeasures
- Earth-orbit retrieval of ECCV
- Transportation vehicles (P-STV, ECV, LPL, LCL, MLL, MPEM)

Flight demonstrations of
- planetary aerocapture
- Lunar LOX production and transport
- man-rated TMIS/TEIS
- Mars descent and landing (esp. parachute assist)

Table 6.1-4 (cont.) CS-4: Transportation Conclusions/Recommendations

Enabling/Enhancing Technologies

Enabling:

- High power, high-efficiency electric propulsion engines
- Multimegawatt nuclear reactor and electrical conversion system
- Multimegawatt thermal radiator systems
- LLOX production from lunar regolith
- Designs which are compatible with on-orbit assembly technology
- zero-gravity countermeasures effectiveness
- Mars aerobrake for aerocapture

Significantly enhancing:

- Advanced, light-weight and low-boiloff tanks; dual wet-dry launch (Siamese-twins)
- Lightweight aerobraking (reusable brakes; ablators)

Interfacing

SS Servicing (OMV; construction crew; assembly)
In-flight communications (Earth-to-Mars-to-Moon-to-interplanetary-to-cis-lunar)
Surface operations and requirements (Mars, moon, Earth)
Connection/requirements with overall transportation infrastructure

Table 6.1-4 (cont.) CS-4: Transportation Conclusions/Recommendations

Issues

- Reliability
  - HLLV
  - Duplicate hardware builds (flight units #1, #2, etc.?)
  - Abort modes
- Mission contingencies
  - Human safety
  - Mission success
- Precursors
  - Transportation man-rating
  - Technology demonstrations in-space (NEP, Ab, ECLSS, etc.)
  - Lunar missions, including LLOX pilot production and regolith prospecting
  - Mars missions, including sample return and landed nav-aides
- Additional risks of split missions
- Multiple dockings and material/personnel transfers in Lunar and Martian orbits

Lessons Learned

- Leveraging LLOX is extremely complex and far exceeds previous U.S. operations in space.
  - Visibility to savings is clouded.
  - Mars missions are delayed compared to other Case Studies.
- Many links occur in the chain of overall mission success.
- There appears to be little commonality between transportation systems.
- Electric cargo vehicle and Phobos LOX utilization lead to heavy emphasis on Phobos orbital rendezvous.

Table 6.1-5 Recommended Cases for 1989

Additional work on Case Studies 1-4:

CS-1. Pursue aerobraking and other mass-reduction approaches.

CS-2. All-up missions, including Opposition class and Conjunction class trajectories. (see N2, N4 below)

CS-3. Reusable lunar vehicles, including LTV and LDV frame.

CS-4. Complete trades of chemical vs electric vs nuclear thermal rocket.

Proposed New Case Studies:

CS-N1. Evolution from Space Station directly to Mars Outpost

CS-N2. Fast-track artificial gravity pathway to early Mars exploration (1st arrival, 2003)

CS-N3. Telerobotic lunar exploration combined with Mars human exploration

CS-N4. Artificial gravity mass minimization to enable low-IMLEO Mars missions

CS-N5. Minimum crew sizes to resuce mass requirements

CS-N6. Flotilla approach (two or more crew vehicles on the same trajectory)

ORIGINAL PAGE IS OF POOR QUALITY

A MANNED MARS ARTIFICIAL GRAVITY VEHICLE

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Abstract

A number of ways of accomplishing manned Mars missions are under assessment at the present time. These include "all-up" missions (in which the crew and all equipment and supplies are transported together in one vehicle) and "split" missions (in which an unmanned cargo vehicle is sent earlier and stored in Mars orbit for rendezvous with a manned vehicle about 1.5 years later). Potential vehicle options for either type of mission include zero-gravity concepts (with "countermeasures" to reduce physiological degradation) and artificial-gravity concepts. This paper presents data on an artificial-gravity vehicle sized for the manned mission of a split-mission concept. A brief comparison with a zero-g vehicle is provided. Comparative data for a few alternative systems options within the concept and limited treatment of subsystems, programmatics, and other considerations are provided with the principal thrust to define one feasible overall configuration.

Introduction

Recently, NASA has performed relatively low effort studies of manned missions to Mars. These studies have been very preliminary in nature, with their primary purpose to provide early data for long-range planning of future missions and systems.

Many options are considered in on-going studies. One of the major areas of consideration is whether to fly an "all-up" versus a "split" type of mission. Most of the work to date has been done on the "all-up" type of mission, in which the crew and all equipment and supplies are transported together in one vehicle. In an attempt to find types of missions having shorter mission durations for the crew, studies have recently been done on "split" missions[1], in which an unmanned cargo vehicle is launched about 1.5 years early and stored in Mars orbit for later rendezvous with a manned vehicle. Using such an approach, mission time is decreased for some opportunities, although some unattractive attributes are incurred. Each of these approaches has pros and cons associated with it, and much more study must be done.

Another key area of consideration is whether to use a zero-g versus an artificial-g vehicle. There are pros and cons associated with either of these approaches, just as there are with the all-up versus split-mission approach, and considerably more study must be done on all options.

To help provide data for the gravity-field question, a preliminary concept was generated for an artificial-g vehicle that could be compared to an existing concept of a zero-g piloted vehicle of a split-mission pair. This was a NASA multi-center in-house activity, with MSFC serving in a lead role and with ARC, JSC, KSC, LaRC, LeRC, and HQ participating. The study took place over about a six-week period in mid-1987.

Groundrules

The study groundrules are listed in Table 1. Since many of these such as g-levels and spin rates are soft at this time, a fair amount of conservatism is used. The split mission concept is utilized with the cargo vehicle preceding the piloted vehicle by about a year and a half. Maximum use is made of the existing definition of the Mars mission concept and cargo vehicle elements from Reference 1. Low Earth Orbit (LEO) departure for the cargo vehicle is mid-2003 and late 2004 for the piloted vehicle with a crew of six. Aerodynamic braking both into Mars orbit and into Earth return orbit is utilized. For most of the Earth-Mars and Mars-Earth transit, a gravity level of 1 g is specified with the possibility of a reduction to Mars gravity (approximately 3/8 g) for up to a month prior to Mars entry. In preparation for aerobraking, all elements are retracted to a zero-g configuration. Gravity levels during aerobraking for either Mars or Earth are specified to not exceed 3.5 g for more than 1 min with a 5 g maximum. A spin rate of 2 rpm initially was selected although this was parameterized from 1 to 4 rpm. The preferred crew module orientation is perpendicular to the spin plane so that crew motion with respect to the floor is maximized in the direction of the spin axis, thereby minimizing coriolis effects. Liquid hydrogen and liquid oxygen cryogenic propellants are used both to and from Mars for the main propulsion system. Provision of safe haven and storm protection for solar flares with a 15 min warning is specified for crew safety. Planned extra vehicular activitiy (EVA) is required in Mars orbit to prepare for surface operations, refuel the piloted vehicle, etc., and contingency EVA is assumed to be required for the Earth-Mars and Mars-Earth transit phases. Use is made of available space and launch elements for operations and derivatives of these elements for Mars vehicle components with a simple, safe, and low-cost concept desired.

Mission Sequence

The split option trajectory profile is illustrated by Figure 1. The cargo vehicle is launched in mid-2003 and arrives in Mars orbit near the end of that year with the return fuel for the piloted vehicle, landers and necessary equipment for the Mars surface operations, probes, and other experiments and equipment as required. Almost a year after the cargo vehicle is safely in Mars orbit, the piloted vehicle is launched, arrives at Mars in mid-2005 for a 30 day stay to permit 10 to 20 days of surface operations, and arrives back at Earth in early 2006. Figure 2 shows a Mars piloted mission scenario which illustrates on-orbit assembly of the Mars space vehicle, trans-Mars injection, deployment of artificial gravity, Mars aerocapture and operations, trans-Earth injection, redeployment of artificial gravity, and Earth aerocapture for Earth return.

Options and Trades

The key system options considered during the study are as depicted in Table 2. The options enclosed by the boxes are those selected. Since life science data currently is insufficient to permit selection of a fractional gravity level for missions up to a year or longer, 1 g is baselined for most of the transit to and from Mars. Although 2 rpm is selected, rotation rate was parameterized from 1 to 4 rpm with coriolis and head-to-foot gravity gradient effects two of the key considerations. The arrangement of the crew modules (i.e. separated by long distances or aggregated/clustered together) is a strong configuration driver. Because of the high cost of transportation to and from Mars, the use of required elements only for the counterweight is selected versus use of Space Station (SS) trash or the Shuttle external tank. Other options are two-body versus three-body masses, booms versus trusses versus tethers, multiple versus single tethers, manned module orientation (long axis in the spin plane versus perpendicular to the spin plane) and spin vector orientation (inertially fixed or Sun tracking).

Figure 3[2] depicts some of the key parameters and the comfort zone used for the study. Rotational radius and percent change in gravity from head to foot are shown as a function of angular velocity (rotation rate). The overall shaded area shows the generally accepted so-called comfort zone for man in a rotating coordinate frame experiencing artificial gravity. For this study, however, the area was restricted to the darker shaded area which ranges from 0.35 g (just under Mars gravity) to 1 g (Earth gravity). The heavy dot shows the design reference point which has a rotational radius of 734 ft and a head-to-foot gravity gradient less than 1%. Also shown in parentheses are values of coriolis force on the crew, calculated in percent of apparent weight, assuming a 3 ft/sec movement of either a portion (such as the head) or all of the body. For the design reference, the coriolis force is almost 4%. As may be observed, for a given radial g level, as angular velocity or rotation rate is increased, coriolis also increases. At the maximum rate of 4 rpm for 1 g, the rotational radius is reduced to 184 ft but the coriolis force is increased to almost 8%. Although currently available life science data in this area is insufficient to permit a definite conclusion, a consensus opinion of the study participants has judged that 1 g and 2 rpm are reasonably acceptable choices to accommodate artificial gravity for a manned Mars mission.

Some of the module arrangement trades are depicted in Table 3. Almost all of the considerations favored the aggregated or clustered crew modules rather than a separated configuration, particularly from a safe haven and storm shelter point of view. Since the counterweight consists of all available mass not required for the crew quarters, a two body overall configuration is then implied to obtain a reasonable mass balance. For example, starting from the original definition of the piloted vehicle[1], this would give approximately 2/3 of the mass for the crew quarters and about 1/3 of the mass for the counterweight. The final configuration was fairly close to these values.

Most of the artificial gravity configuration options that were considered are illustrated by Figure 4. The shaded area shows just the two body configurations with aggregated habitability modules. Since Configuration I has all floors for the crew modules generally at the same gravity level and also has acceptable end body dynamic stability, it was selected for further definition as shown by the asterisk. Although Configuration II also has good end body dynamic stability, its floors have different gravity levels for the crew modules and the crew must climb up and down a "gravity well." There are many other features of this type of floor arrangement that also make this configuration undesirable (this is consistent with trades that have been done in SS studies). Configuration III has all floors at the same gravity level with the modules perpendicular to the spin plane. This tends to maximize crew relative velocity (motion with respect to the floors) in the direction of the spin axis thereby minimizing coriolis effects (i.e. relative velocity perpendicular to the spin axis maximizes coriolis force and relative velocity along the spin axis reduces coriolis force to zero). All attempts to achieve crew module end body dynamic stabilization, however, were unsuccessful. Although this configuration is preferred, the coriolis effects were judged to be second order with respect to dynamic stability and Configuration I was selected. Future work in this area would be beneficial to find a configuration arrangement that would satisfy these criteria simultaneously.

End body masses for Configuration I were then estimated to be 4263 slugs for the manned module assembly and 2367 slugs for the counterweight assembly. Using these values, Figure 5 shows the length between bodies (solid lines) and the rotational radius (dashed lines) as a function of rotation rate from 1 rpm to 4 rpm for both 1 g (Earth gravity) and 3/8 g (Mars gravity). The heavy dots show that the length between bodies is 2058 ft and the rotational radius is 734 ft for the 1 g and 2 rpm reference values. Using the same mass and rotation rate values, Figure 6 gives the fuel weight required for spinup and spindown assuming a complement of cryogenic thrusters with an Isp of 460 sec on each of the end bodies. Note that slightly more than 12,000 lb of spinup-spindown propellant is required for 1 g and 2 rpm as shown by the heavy dot.

Additional subsystems trades were performed to identify orientation options, structural interfaces, and thermal control considerations. As shown in Figure 7, the spin axis can either remain normal (or perpendicular) to the ecliptic plane or be pointed at the sun by continuously precessing the spinning configuration using a bipropellant reaction control system (RCS) on each end body. For the 1 g and 2 rpm reference values, about 32,400 lb of bipropellant fuel is required to precess the spin-axis for the entire Earth-Mars and Mars-Earth trip (approximately 396 deg of solar vector precession angle). For the spin axis continuously pointed at the sun, the solar arrays always face the sun as shown in Figure 7, and the power system is simpler and lighter weight. However, with the spin axis normal to the ecliptic plane, the power system, although more complicated (arrays continuously rotate at spin speed with respect to the sun and must be active on both sides), is only about 4,000 lb heavier so this orientation was selected. Figure 8 shows typical weights for key interfaces and structural support. Figure 9 shows the current thermal control concept using SS modules modified by adding single phase body mounted radiators since two-phase radiators are not useable in the 1-g environment.

Tethers and Dynamic Stability

End Body Stability. Euler's equations describe the motion of a body subject to initial rotation rates and external torques. Inspection of these equations shows that rotations about the principal axes continue about that axis in the absence of external torques. For this condition, the rotation is defined to be stable.

In the spinning configuration of the manned Mars vehicle, the bodies must be aligned so that the spin axis is parallel to a principal axis of each of the end bodies. This orientation prevents the end bodies from tumbling. With the long distance between the end bodies, maintaining a stable orientation is important for both tethers and rigid structures to minimize reaction torques which can cause tether system instability or destroy rigid booms and trusses.

Figure 10 shows the three stable end body orientations connected together by a tether. Any of the crew or manned module end-body orientations can be connected to any of the counterweight end-body orientations. The different orientations of the solar arrays cause the crew end-body moments-of-inertia to be different for the three connections. For each connection, there is a stable and unstable orientation about the tether axis. The stable orientation is shown. The unstable orientation has the end-body rotated 90 deg about the tether-axis. The tendency either to be stable or tumble is assessed by calculating the angular accelerations of the body with a slight offset in the body orientation. This is done for 1 deg errors about the three body axes. The resulting accelerations are divided by the moments-of-inertia to give equivalent Euler moments or torques as shown. This Euler moment is the torque required to force the spin to be about a non-principal axis and represents the torsion in the tether or rigid structure. For the cases shown, the torques are negative for an offset about the tether-axis, meaning the acceleration is in a direction to return the body to the zero orientation for which the spin is about a principal axis. Small offsets about the principal axis parallel to the spin axis do not result in instability, and the torque caused by tether attach point relative to the body center-of-gravity restores the orientation to zero. Likewise, small positive torques about the axis normal to the figure can be effectively countered by a bridle.

Tether Design Requirements. Design requirements for the connection between the end bodies are identical for the rigid boom, truss, and tether approaches. The structure must survive the space environment with high probability, must be capable of deploying and retracting without EVA and withstanding the static and dynamic loads, and must be stowed behind the aerobrake shield. The structural approach which meets these requirements with the minimum weight is the desired approach.

A rigid structure was found to not meet these mandatory requirements. Relaxing the packaging requirement results in otherwise suitable rigid designs. However, because of the large separation distance between the two end bodies, the rigid structure adds little to the stability of the system, while the weight of the rigid structure far exceeds the weight of the tether approach as shown in Figure 11.

Tether design concepts are shown in Figure 12. The approach to maximize micrometeoroid impact survivability is subject to debate, and further technology studies and testing are required to determine the best approach. Cross-members are required to equalize loads in the event of failure of a member; however, overdesign can cause failures to propagate. Tether strength requirements are extrapolated from a study by Martin Marietta Aerospace.[3] The designs shown have a safety factor of 7 compared to the breaking strength.

The tether deployment and retrieval concept for the Mars mission uses a thruster system to separate the end bodies from each other while playing out the cable under low tension. A simple winch-type mechanism can be used since little energy dissipation is involved. The deployer concept used in Reference 3 can contain a large volume of tether but it uses an electrodynamic brake which is not required. Rather than perform a new design, the deployer mass of the Martin design was used for the Mars mission weight statement, recognizing that the deployer motor, power control, and thermal system were excessive for this application but the tether reel should be somewhat larger. Table 4 shows the capability of the Martin design versus the Mars mission cable deployer requirements.

Future studies are required in the areas of tether design technology, deployment and retraction operations, attachment design for no EVA, dynamics, and spin-up/spin-down operations. The spin fuel is sized assuming thruster torques are applied at both ends resulting in equal angular accelerations on the two end bodies. Significant savings can be obtained if a method can be employed which applies most of the spin thrust at the counterweight end which has the longest lever arm from the configuration center-of-mass.

Abort and Degraded Modes

For mission safety, several abort and degraded modes were considered as illustrated in Figure 13. Abort return to LEO is possible until approximately 10 min into the third stage burn, after which Mars transit is mandatory. If a critical problem develops on the way to Mars, a free return to Earth is available using a small aeropass maneuver at Mars for velocity adjustment. In the event of a severed tether, a recovery sequence is available as illustrated in the lower right corner of the figure. Cryogenic thrusters and a bipropellant RCS on each of the manned module and counterweight assemblies will null the separation velocities, after which a rendezvous sequence will bring the two bodies back to the final docking configuration as shown. From the time the tether breaks, this uses slightly more than 3,000 lb of propellant and approximately 1 hr. This propellant is about equal to that for a normal spindown so most of it already would be available in the design. If the tether can be repaired (or a spare tether is available), the configuration can again be spun up for artificial gravity. However, if the tether is unable to be repaired, abort to zero-g is the final degraded mode for the remainder of the mission. For subsystems safety and reliability, the tether is sized with an overall factor of safety of 7, SS subsystems are utilized with various levels of redundancy, 5,000 lb of spare parts are assumed for maintenance and repair, and contingency propellant is included for one extra spinup and spindown as well as severed-tether recovery for each trajectory leg to and from Mars. Degraded power modes also range down to about half the all-up operational level of 25 kW as shown.

Configuration

The vehicle is composed of the crew module assembly and the counterweight assembly. During the orbit change maneuvers, the vehicle is stacked together as shown in Figure 14. For the transit phase of the mission, the two portions of the vehicle are separated but held in position by the tether system as illustrated by Figure 15. Structural members are not shown in the figures, but weight estimates have been included for them. The structural members would support the elements from their trunion fittings in the Earth-to-orbit launch vehicles.

The crew module assembly contains modules assembled into a modified SS "race-track" configuration shown in the lower right hand corner of Figure 16. The assembly uses two shortened SS habitability modules, four SS nodes, and an airlock. One of the nodes serves as the solar flare storm shelter. Mounted to the module assembly are the cryogenic and bipropellant RCS thrusters and cryogenic spinup/spindown propellant tanks (bipropellant tanks not shown). Figure 16 also shows the tether joined to the crew module and counterweight assemblies with a four point bridle. The tether deployment system is mounted between the habitability modules. The solar array system is mounted to the module assembly and deployable/retractable beams are used to deploy the arrays during the zero-g parts of the mission. The arrays gimbal about one axis (tether axis) only, because of the severe weight penalties which would be imposed to gimbal them out of the spin plane. The module assembly includes a despun platform (not shown) which contains the communications package and additional science and engineering instruments.

The counterweight assembly contains the aerobrake, a propulsion stage, cryogenic and bipropellant RCS thrusters, cryogenic spinup/spindown propellant tanks (bipropellant tanks not shown), and the Earth return capsule (ERC). The aerobrake is a 100 ft x 70 ft ellipsoid. The ellipsoid gives a larger lift to drag (L/D) range during the aerobraking maneuver than does a circular aerobrake. All other components of the counterweight nest inside the aerobrake. The propulsion stage is used for Earth-Mars and Mars-Earth trajectory insertion (after being refueled in Mars orbit from the cargo vehicle). The empty stage serves as part of the counterweight mass during transit to and from Mars. The ERC is used during the Earth aerobraking maneuver.

Weights and Launch Vehicle Requirements

A summary of the weights for the piloted vehicle is shown in Table 5. To compare with the 1 g artificial gravity case, an updated zero gravity configuration is also shown. The piloted vehicle with artificial-g weighs a little over 45,000 lb more than the updated zero-g configuration (approximately 20% more). Various elements and corresponding weights, as well as delta weight increases, are also shown in the table along with the total system weight for each case. Since the piloted vehicle is refueled from the cargo vehicle and a portion of the aerobrake system for the zero-g piloted vehicle is jettisoned in Mars orbit (this is needed for the artificial gravity counterweight), the total system weights have been adjusted for performance as shown by the asterisks. Figure 17 then shows the required number of 200 Klb class Heavy Lift Vehicle (HLV) launches as a function of these adjusted payload weights. The updated zero-g case requires 7 HLVs for the cargo vehicle and 9 HLVs for the piloted vehicle to transport all required mass from the surface of Earth to the SS altitude for assembly (assumed to be 220 n.mi. rendezvous altitude). For the artificial gravity case, this grows to 9 HLVs for the cargo vehicle and 12 HLVs for the piloted vehicle, or a total of 5 additional HLV launches.

To include the weights of the stages for both piloted and cargo vehicles, an overall weight summary is given in Table 6. Addition of the total delta weights gives an overall increase of 26% to both piloted and cargo vehicles for the addition of artificial gravity.

Other Considerations

Commonality. The artificial-g vehicle concept can have a high degree of commonality with the SS, which should give confidence in safety and reliability, and should help keep costs low and schedules short. There can be a high degree of commonality between an artificial-g and a zero-g vehicle. It is not much of an overstatement to say that the artificial-g vehicle is basically a zero-g vehicle with an add-on system for artificial gravity.

Schedule. A very preliminary assessment indicates that the overall time span required for development, production, and mission preparation for the artificial-g vehicle is not significantly greater than that required for the zero-g vehicle. Due to the potential high commonality between the zero-g and artificial-g vehicles, development and production of the common systems can progress fairly far towards completion before a decision to go one way or the other is necessary. A more sizable programmatic implication is the potential need for a precursor LEO Variable-g Facility to verify the human physiological implications of artificial-g. Such a facility can also provide verification of the artificial-g vehicle systems, to some degree. The facility can be of a minimal nature having much smaller habitable volume than the artificial-g vehicle, and possibly can use discarded space hardware (SS "trash", the STS External Tank, etc.) for a counterweight.

Cost. A preliminary assessment indicates that the cost of a piloted/cargo vehicle pair for a split mission is increased approximately 10% by adding artificial-g. This includes costs of development and unit purchases, but does not include a LEO Variable-g Facility.

Technology. Table 7 lists some key areas of technology or advanced development applicable to an artificial-g vehicle. None of these require any major advances of existing technology. Research in the life sciences areas shown is necessary because of the limited understanding we currently have of the physiological implications of gravity fields induced by rotation. Artificial-g provides some definite benefits, but also introduces some problems (Coriolis effects, for example) and these must be better understood. Research can produce results which permit g-levels and rotation rates to be used with much less severe systems design implications than those used in this study. The items listed under the "module systems" category probably are the least demanding of any on the list from the technology/development standpoint, but are shown here for completeness.

Summary

Table 8 provides a summary of the key findings from this study. Most of these are discussed earlier in the paper, so will not be addressed again here. At the present time, there appears to be a fairly high degree of optimism among many life science people that "countermeasures" can be found to offset or prevent the deleterious physiological effects of zero-g. Life sciences research, currently being planned for the early period of Space Station operation to certify a routine crew stay-time of 180 days at the station, should go far towards proving the effectiveness of zero-g countermeasures.

At this time, insufficient data exists to determine whether or not an artificial-g vehicle will be required for manned Mars missions. Both concepts should be studied further, and more life sciences research must be done. The high commonality potential of systems between zero-g and artificial-g vehicles would allow development and production to begin early on the common systems and a decision to be made later on the gravity question. The consideration of whether or not a LEO Variable-g Facility is required becomes of key importance early if it is determined that (1) an artificial-g vehicle is needed and (2) it is desired to not slip the schedule for the initial mission beyond that currently envisioned for the zero-g vehicle in NASA's manned Mars "new initiative."

References

1. John Niehoff, "Humans to Mars - A Space Leadership Initiative." Science Applications International Corporation presented at MSFC/NASA Mars Status and Orientation Meeting, May 28, 1987.

2. Hill, Paul R. and Schnitzer, Emanuel, LaRC, "Rotating Manned Space Stations" Astronautics, September 1962.

3. "Selected Tether Applications in Space" by Martin Marietta Aerospace, NASA Contract NAS8-36616, Phase III Final Report, September 1986.

TABLE 1. Manned Mars Mission – Artificial Gravity Accommodation

GROUNDRULES

- USE SPLIT OPTION MISSION CONCEPT
  - CARGO VEHICLE PRECEEDS PILOTED VEHICLE
- USE EXISTING DEFINITION OF MARS CARGO VEHICLE ELEMENTS AND MARS SURFACE ACTIVITIES
  - NO NEW ASSESSMENT
- LEO DEPARTURE
  - CARGO VEHICLE: MID 2003 - CONJUNCTION TRAJECTORY
  - PILOTED VEHICLE: LATE 2004 - SPRINT TRAJECTORY
- CREW SIZE OF 6
- AERODYNAMIC BRAKING INTO MARS ORBIT AND INTO EARTH RETURN ORBIT
- GRAVITY LEVELS
  - 1G FOR MOST OF EARTH-MARS AND MARS-EARTH TRANSIT
  - 3/8G NEAR MARS INTERFACE
  - OG PRIOR TO MARS OR EARTH AEROBRAKING
  - NOT TO EXCEED 3.5G FOR MORE THAN 1 MINUTE WITH 5G MAXIMUM
- INITIALLY SIZE FOR SPIN RATE OF 2 RPM
  - UP TO 4 RPM ULTIMATELY MAY BE POSSIBLE
- PREFERRED MODULE ORIENTATION (FLOOR) PERPENDICULAR TO SPIN PLANE
- CRYO PROPELLANTS FOR EARTH-MARS & MARS-EARTH TRANSFER
- PROVIDE SAFE HAVEN FOR CREW SAFETY
- PROVIDE STORM PROTECTION FOR SOLAR FLARES WITH 15 MIN WARNING
- UTILIZE PLANNED EVA IN MARS ORBIT
- CONTINGENCY EVA DURING EARTH-MARS & MARS-EARTH TRANSIT
- CONSIDER AVAILABLE SPACE AND LAUNCH ELEMENTS FOR OPERATIONS AND DERIVATIVES OF THESE ELEMENTS FOR MARS VEHICLE COMPONENTS
  - SPACE STATION
  - OTV
  - OMV
  - SDV OR HLLV
  - STS
- SIMPLE, SAFE, LOW COST CONCEPT DESIRED

TABLE 2. Manned Mars Mission – Artificial Gravity Accommodation

KEY SYSTEM OPTIONS

- GRAVITY LEVEL: PARTIAL G VS. 1G
- ROTATION RATE: 2RPM VS. OTHER (UP TO 4RPM)
  - CORIOLIS CONSIDERATIONS
  - HEAD-TO-FOOT GRAVITY-GRADIENT
- SEPARATED MODULES VS. AGGREGATED MODULES
- USE OF SPACE STATION TRASH OR SHUTTLE EXTERNAL TANK FOR COUNTERWEIGHT VS. USE OF ONLY REQUIRED ELEMENTS (SEPARATED AS NECESSARY)
- TWO-BODY VS. THREE-BODY MASSES
- BOOM VS. TRUSS VS. TETHER
- MULTIPLE VS. SINGLE TETHERS
- MODULE ORIENTATION: IN SPIN PLANE VS. PERPENDICULAR TO SPIN PLANE
- ORIENTATION OF SPIN VECTOR: INERTIALLY-FIXED VS. SUN-TRACKING

TABLE 3. Manned Mars Missions – Artificial Gravity Accommodation

MODULE ARRANGEMENT TRADES	SEPARATED	AGGREGATED
• SAFE HAVEN CONSIDERATIONS		
– NUMBER OF EXITS PER MODULE		√
– AVAILABILITY OF MULTIPLE VOLUMES WITH MINIMUM MODULES		√
• STORM SHELTER ASPECTS		
– RAPID ACCESS		√
– NEED FOR JUST ONE SHELTER		√
• PSYCHOLOGICAL CONSIDERATIONS		
– LARGE GROUP INTERACTION		√
– MINIMIZATION OF FEELINGS OF ISOLATION		√
• CREW SKILL MIX AVAILABLE		√
• SYSTEMS CONSIDERATIONS		
– MINIMUM NUMBER OF AIRLOCKS		√
– MASSES MORE EQUAL	√	

TABLE 4. Tether Deployer

STS TETHER DEORBIT EXAMPLE

DEPLOYER MASS 5,255 LB (2,380 KG)

VOLUME OF TETHER (48 FT^3 (1.31 M^3)

MANNED MARS REQUIREMENT

STRENGTH 179,990 LB (8,181KG)

LENGTH 2,058 FT (627M)

VOLUME OF TETHER 65 FT^3 (1.73 M^3)

CONCLUSION

DEPLOYER MASS 5,255 LB (2,389 KG) (MOTORS, REEL, STRUCTURE, CONTROLLER, NO TETHER)

APPROXIMATE REEL SIZE: 2.2 M DIA X 0.7 M LONG)

TETHER MASS 3,770 LB (1713 KG)

TABLE 5. Manned Mars Split Mission Piloted Vehicle Weight Summary (lb)
(Excluding Propulsive Stages)

DESCRIPTION	UPDATED OG	1G CASE	DELTA
COUNTER WEIGHT			
AEROBRAKE	29121	31605	2484
EARTH RETURN CAPSULE	14883	14883	
3rd STAGE	32174	32174	
AVIONICS		393	393
RCS SUBSYSTEM (15% CONT ON TANKS)		11142	11142
SUPPORT STRUCTURE	2838	4652	1814
SUBTOTAL	79016	94849	15833
MANNED MODULE ASSEMBLY			
MODULE STRUCTURE (2)	25864	25864	
NODE STRUCTURE (4)	24157	24157	
AIRLOCK	6000	6000	
THERMAL	9675	9675	
AVIONICS	7496	7496	
CREW SYSTEMS	9448	9448	
ECLSS	14768	14768	
MODULE/NODE CONTINGENCY (15%)	14611	14611	
SUPPLEMENTAL SHIELDING (STORM SHELTER)	3000	3000	
SPARES	5000	5000	
FLUIDS, THERMAL	432	432	
FLUIDS, ELECTRICAL	64	64	
ECLSS CONSUMABLE	6294	6294	
CREW SYSTEM CONSUMABLES	11267	11267	
TETHER (INC 15% CONT)		4335	4335
TETHER DEPLOYER (INC 15% CONT)		6009	6009
RCS SUBSYSTEM (15% CONT ON TANKS)		12802	12802
ELECTRICAL POWER SYSTEM	4470	8466	3996
SUPPORT STRUCTURE	1589	3994	2405
MISSION SCIENCE	5168	5168	
CREW (6)	1140	1140	
SUBTOTAL	150443	179990	29547
TOTAL SYSTEM WEIGHT(*ADJ FOR PERFORMANCE)	229459 (196383)*	274839 (269276)*	45380

TABLE 6. Artificial-G Vehicles Overall Weight Summary

	UPDATED ZERO-G WET WEIGHT (LBS)	ARTIFICIAL G WET WEIGHT (LBS)	DELTA WET WEIGHT (LBS)
PILOTED VEHICLE			
• COUNTERWEIGHT	79,016	94,849	15,833
• MANNED MODULE ASSY	150,443	179,990	29,547
• STAGE 1	727,974	865,579	137,605
• STAGE 2	359,952	572,679	212,727
• STAGE 3	202,903	282,740	79,837
TOTAL	1,520,288	1,995,737	475,549
CARGO VEHICLE			
• PAYLOAD	427,403	519,240	91,837
• STAGE 1	727,974	865,579	137,605
TOTAL	1,155,377	1,384,819	229,442

TABLE 7. Key Enabling Technology/Advanced Development Areas for Artificial-G Mission

	FOR PRIMARY MODE	FOR BACKUP MODE
• LIFE SCIENCES		
– PHYSIOLOGICAL ATTRIBUTES OF ROTATIONAL g-FIELDS	√	
– ALLOWABLE ROTATION RATE	√	
– ALLOWABLE g-LEVEL	√	
– ALLOWABLE g-FIELD GRADIENT (HEAD-TO-FOOT)	√	
– ZERO-g COUNTER MEASURES		√ *
– ZERO-g MEDICAL TOOLS & TECHNIQUES		√ *
• ARTIFICIAL-g SYSTEMS	√	
– SYSTEMS IMPLICATIONS OF VARIABLES LISTED UNDER LIFE SCIENCES		
– TETHERS		
– SPINUP/SPINDOWN TECHNIQUES		
– SEVERED-TETHERED RECOVERY		
• MODULE SYSTEMS FOR OPERATION IN 0-g AND 1-g	√	√
– PRELIMINARY ASSESSMENT SHOWS AREAS NEEDING MODIFICATION ARE:		
• THERMAL CONTROL SYSTEM		
• SLEEP STATION		
• SHOWER & WASTE MGMT.		
• HUMAN FACTORS		
• HIGH-CYCLE-LIFE/"HIGH-FREQUENCY"(2 RPM) SYSTEMS	√	
– GIMBAL SYSTEMS FOR DESPUN PLATFORM/INSTRUMENTS		
– ENERGY STORAGE SYSTEM		

* TECHNOLOGY WORK PLANNED FOR CERTIFICATION OF >90-DAY STAY AT SPACE STATION MAY PROVIDE MOST OF THIS.

TABLE 8. Summary Observations

GENERAL

- AN ARTIFICIAL-G VEHICLE FOR MANNED MARS MISSIONS APPEARS TO BE FEASIBLE TECHNICALLY AND PROGRAMMATICALLY; A VARIETY OF SAFETY/RELIABILITY FEATURES CAN BE PROVIDED.
- THERE CAN BE A HIGH DEGREE OF COMMONALITY BETWEEN A ZERO-G AND AN ARTIFICIAL-G VEHICLE; THE DIFFERENCES ARE MOSTLY ADDITIONS OF EQUIPMENT.
- USING AN ARTIFICIAL-G VEHICLE INSTEAD OF A ZERO-G VEHICLE FOR THE PILOTED PORTION OF A SPLIT MISSION TO MARS:
  - ADDS ABOUT 478K LB. TO THE PILOTED VEHICLE WEIGHT AND ABOUT 230K LB. TO THE CARGO VEHICLE WEIGHT (26% TOTAL INCREASE).
  - ADDS ABOUT 10% INCREASE TO THE COST OF A PILOTED/CARGO PAIR, INCLUDING COSTS OF THE 5 ADDITIONAL EARTH-TO-ORBIT HLV LAUNCHES REQUIRED.
  - PROVIDES BENEFITS IN PHYSIOLOGICAL AND HUMAN FACTORS AREAS.
  - DOES NOT ELIMINATE REQUIREMENTS FOR ZERO-G COUNTERMEASURES RESEARCH (SINCE ZERO-G IS AN ABORT MODE), BUT SHIFTS THE PRIMARY EMPHASIS TO VERIFICATION OF ARTIFICIAL-G UNKNOWNS (POTENTIAL PHYSIOLOGICAL & ADAPTATION EFFECTS).
    - COULD POSSIBLY REDUCE SOME LIFE SCIENCE ACTIVITIES AT SPACE STATION
    - REQUIRES SOME TESTING & SIMULATION OF ARTIFICIAL-G MISSION, PROBABLY NECESSITATING A LEO VARIABLE-G FACILITY.
  - DOES NOT IMPOSE SIGNIFICANT SCHEDULE IMPACTS

DESIGN CONSIDERATIONS

- GROUPING/LOCATION OF ELEMENTS FOR CREW QUARTERS AND COUNTERWEIGHT IS A STRONG DRIVER ON CONFIGURATION
- LARGE SEPARATION DISTANCE (2058 FT) BETWEEN CREW QUARTERS AND COUNTERWEIGHT REQUIRED FOR 1 G AND 2 RPM; DISTANCE IS VERY SENSITIVE TO g-LEVEL AND RPM.
- TETHER SYSTEM IS THE ONLY REASONABLE STRUCTURAL CHOICE FOR LONG SEPARATION DISTANCES
- G-LEVELS LOWER THAN 1G AND SPIN RATES UP TO 4 RPM WOULD SIGNIFICANTLY REDUCE THE SIZE OF THE ARTIFICIAL G SYSTEM (AT PRESENT LIFE SCIENCE DATA NOT SUFFICIENT TO PERMIT THESE CHOICES)
- ALL CREW SYSTEMS AND HUMAN FACTORS REQUIREMENTS CAN BE MET EXCEPT DESIRE FOR MODULE LONG AXIS ORIENTATION PERPENDICULAR TO SPIN PLANE (JUDGED TO BE SECOND-ORDER CONSIDERATION).

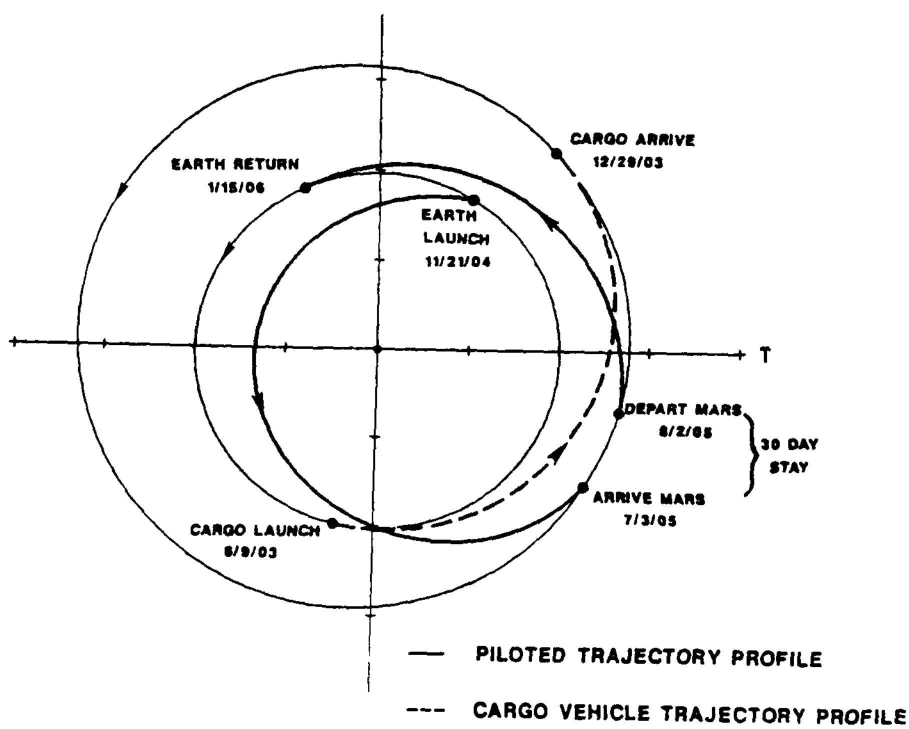

Figure 1. Split Option Trajectory Profile.

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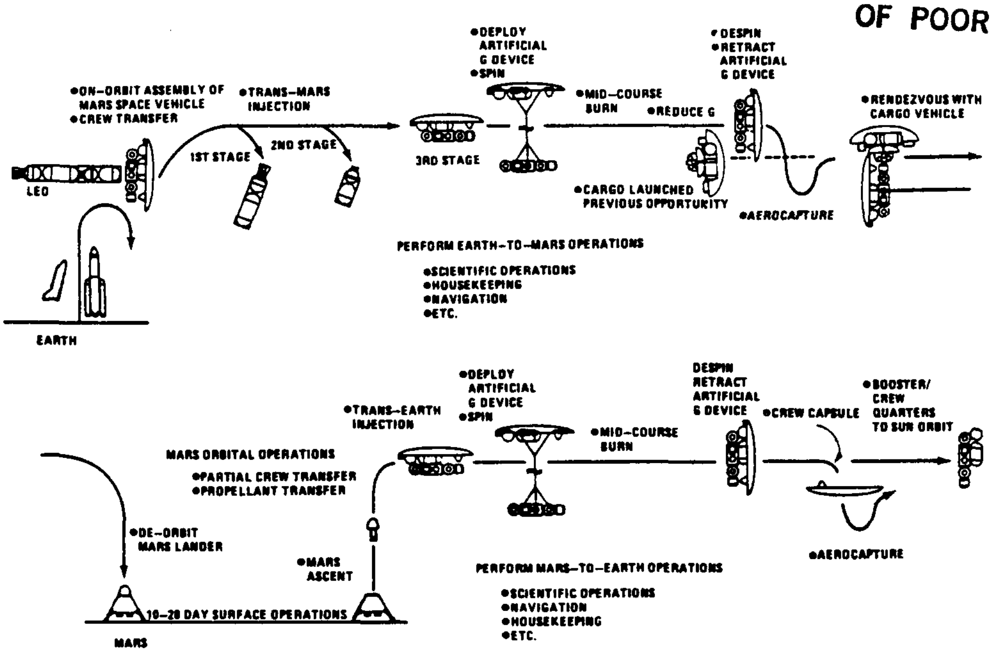


Figure 2. Mars Piloted Mission Scenario.

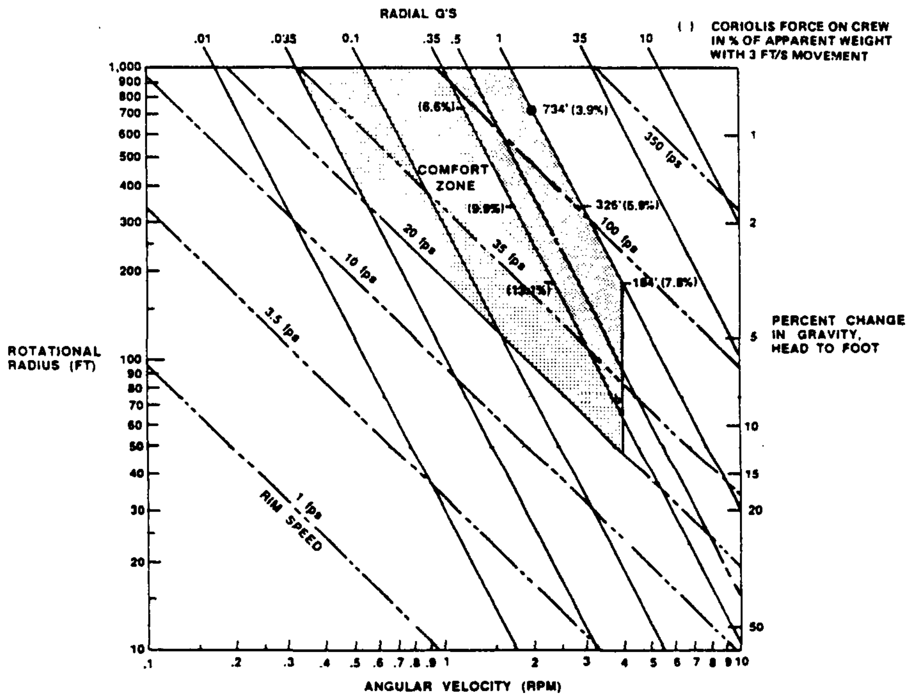


Figure 3. Rotational Parameters and Comfort Zone.

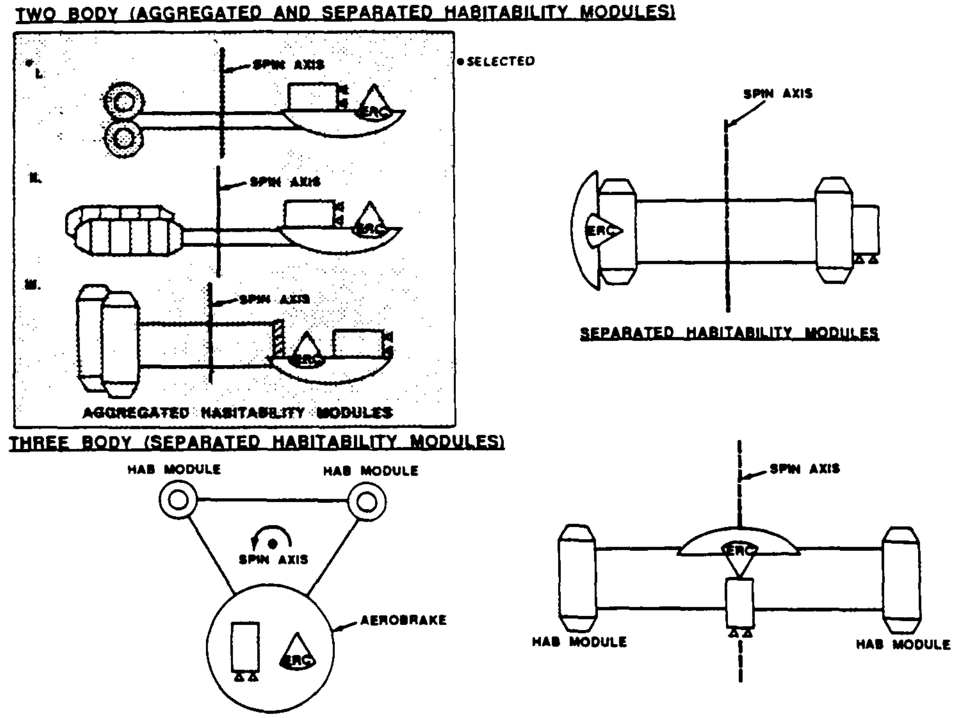


Figure 4. Artificial Gravity Configuration Options.

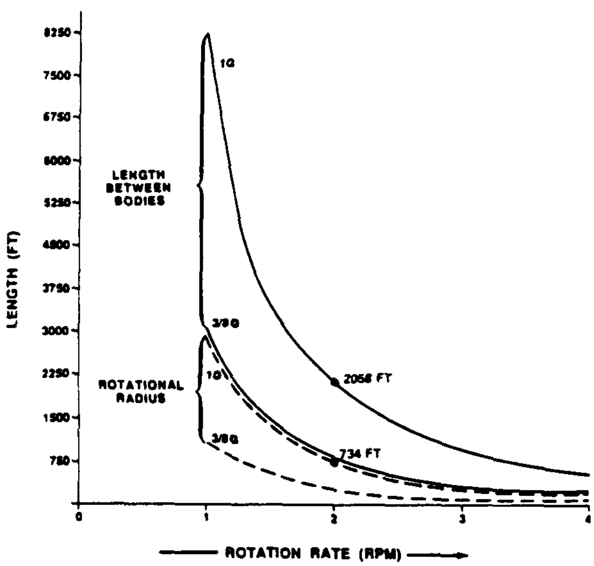


Figure 5. Length Between Bodies Versus Rotation Rate.

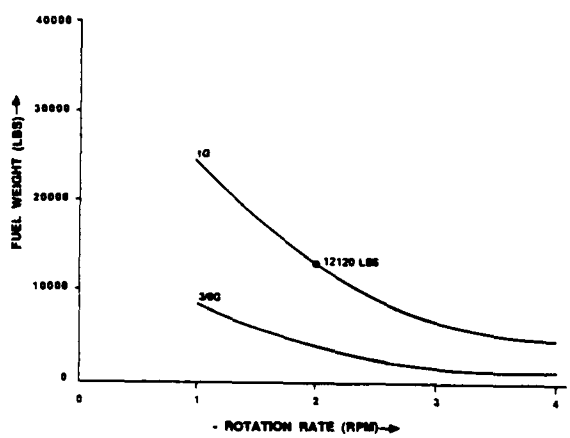


Figure 6. Cryo Fuel Weight Versus Rotation Rate.

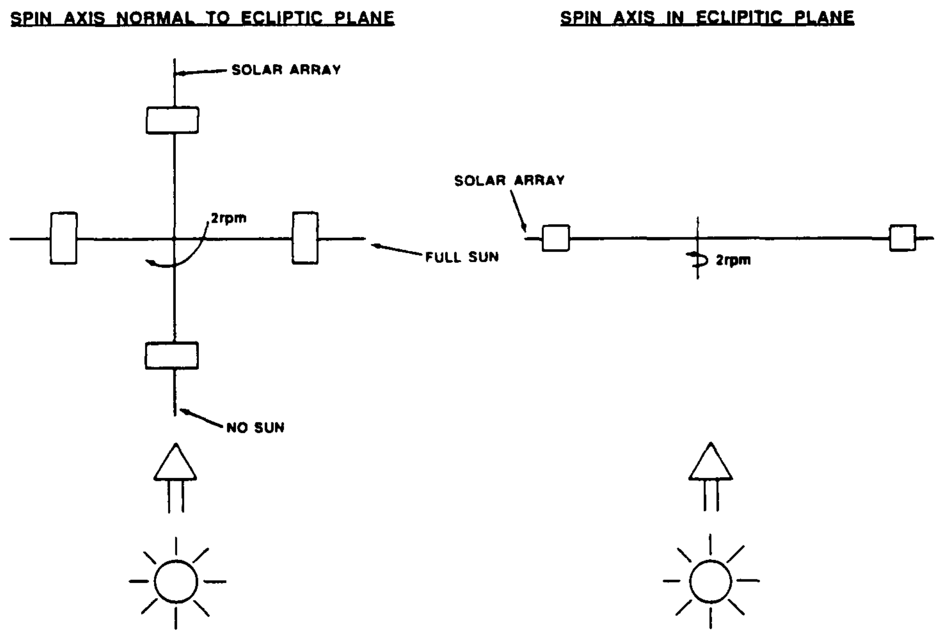


Figure 7. Orientation Options.

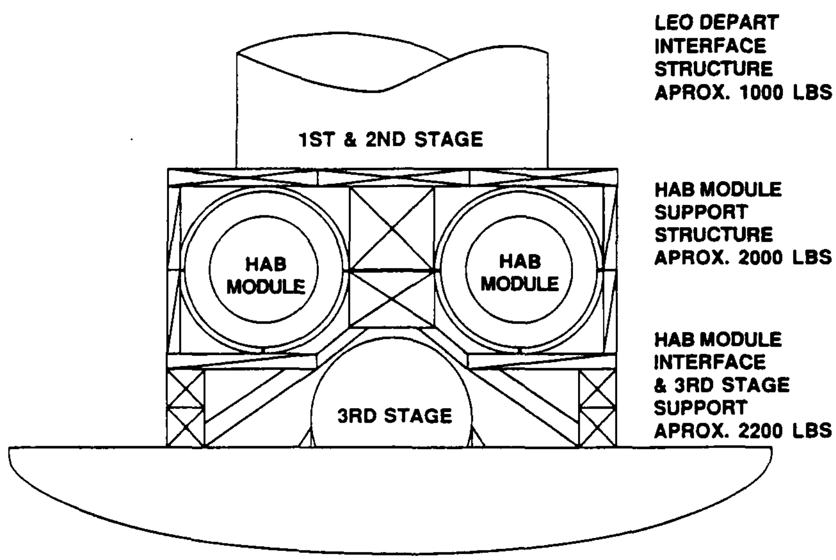


Figure 8. Manned Mars Mission.

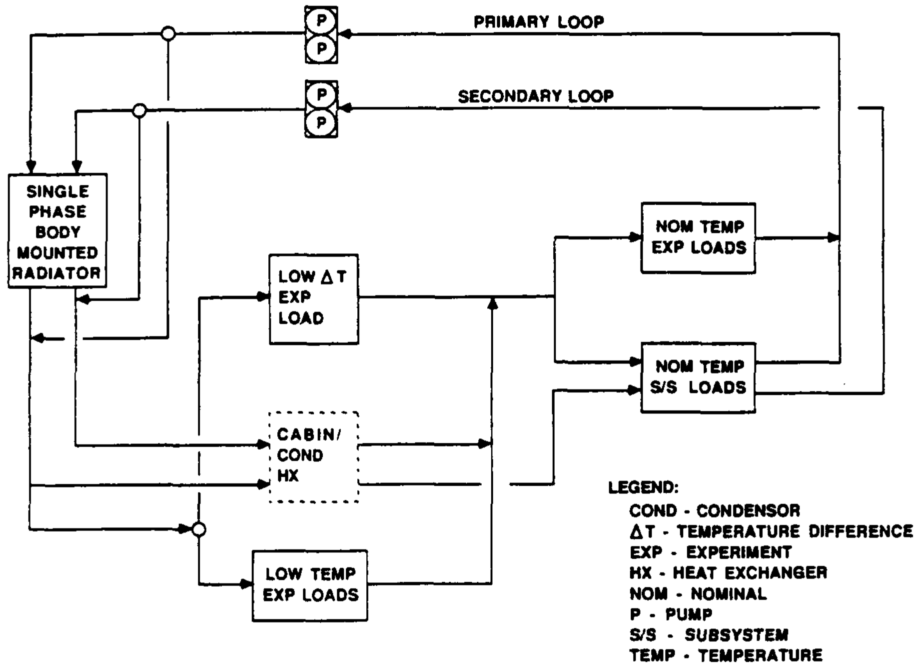
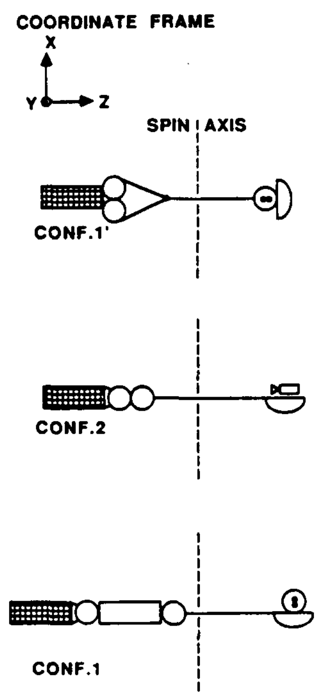


Figure 9. Manned Mars Mission – Artificial Gravity Accommodation Thermal Control.



	MOMENTS OF INERTIA, SLUG-FT²		EULER MOMENTS, FT-LB/DEG	
	CREW END	SHIELD END	CREW END	SHIELD END
X	3,000,601	1,956,467	0	0
Y	2,643,672	881,674	-679	385
Z	2,112,561	2,460,166	-273	-822
X	4,223,698	2,460,166	0	0
Y	3,006,507	1,956,467	-2115	-1208
Z	1,460,318	881,674	-931	-385
X	4,762,152	2,460,166	0	0
Y	4,037,350	881,674	-2904	-385
Z	967,929	1,956,467	-554	-1208

Figure 10. Three Stable Orientations.

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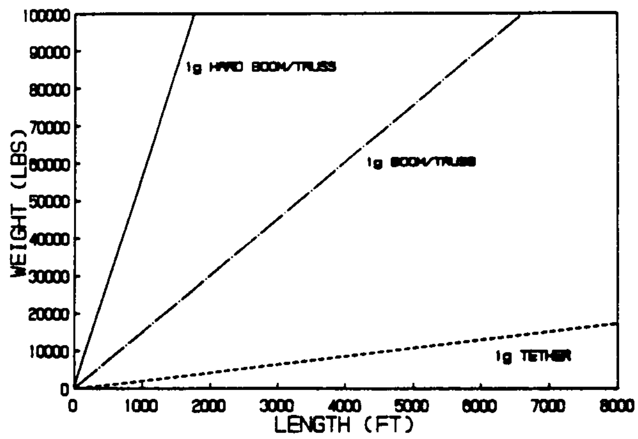


Figure 11. Manned Mars Mission Boom and Tether Length Versus Weight.

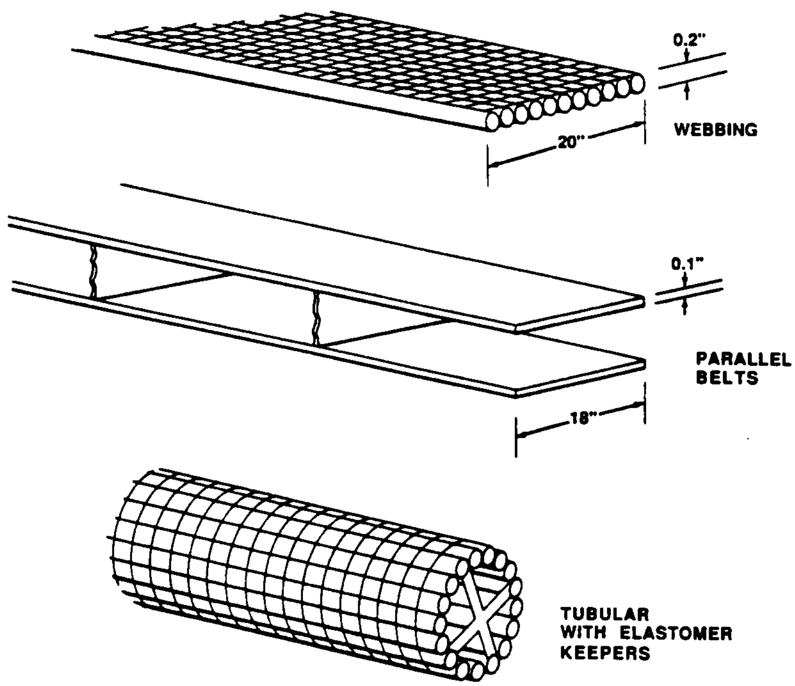


Figure 12. Some Redundant Tether Ideas.

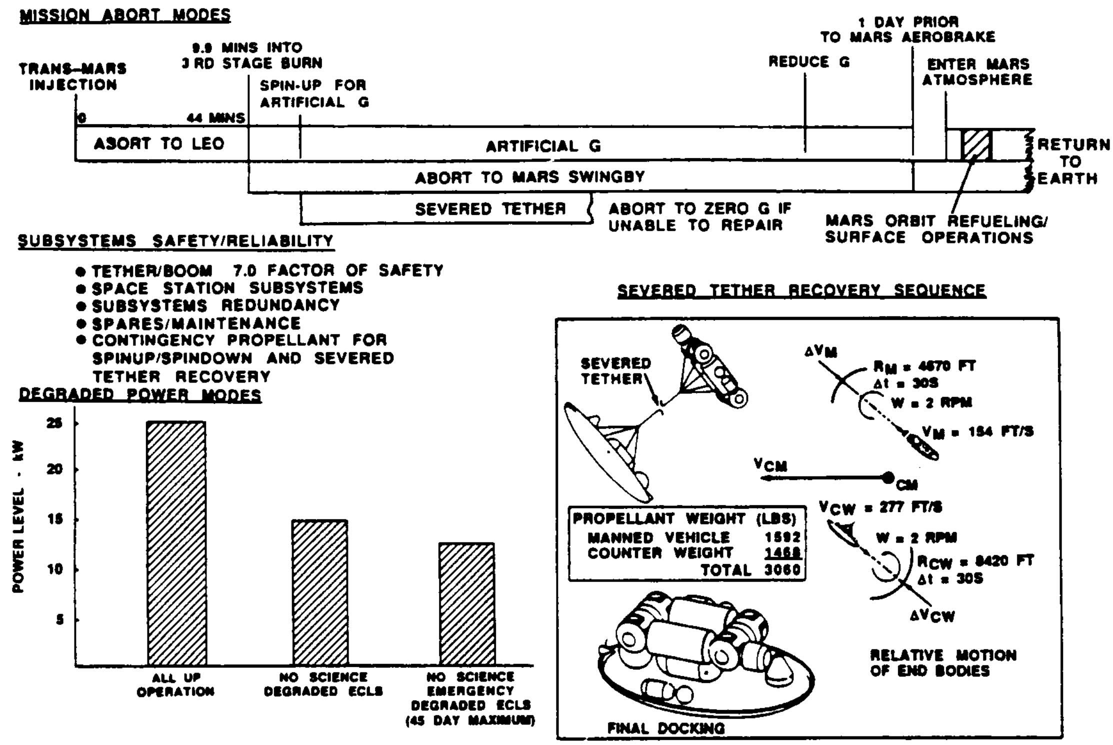


Figure 13. Manned Mars Missions – Artificial Gravity Accommodation Abort/Degraded Modes/Safety Factors

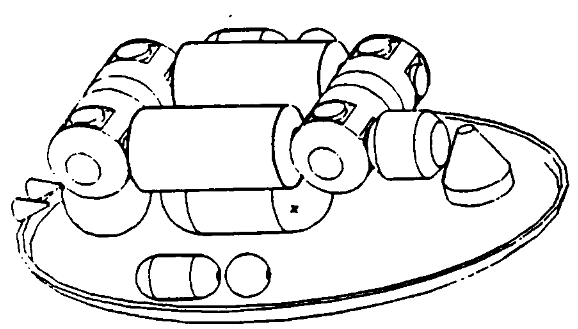


Figure 14. Manned Mars Artificial-G Vehicle Stowed Configuration.

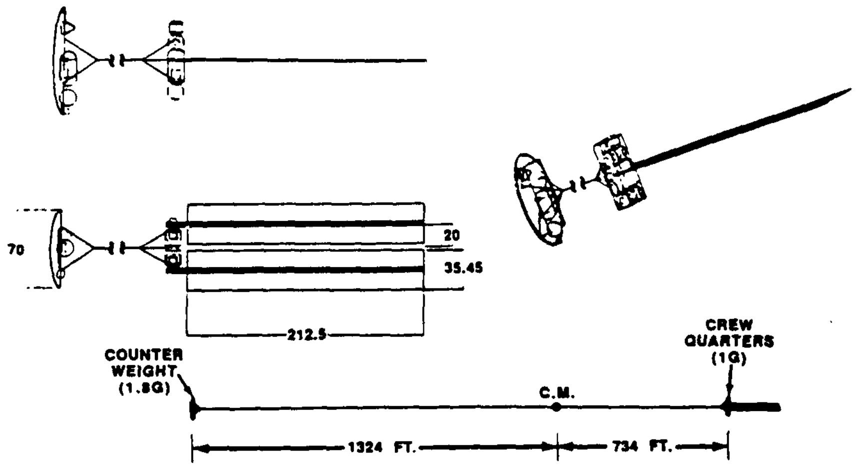


Figure 15. Manned Mars Artificial-G Vehicle Deployed Configuration.

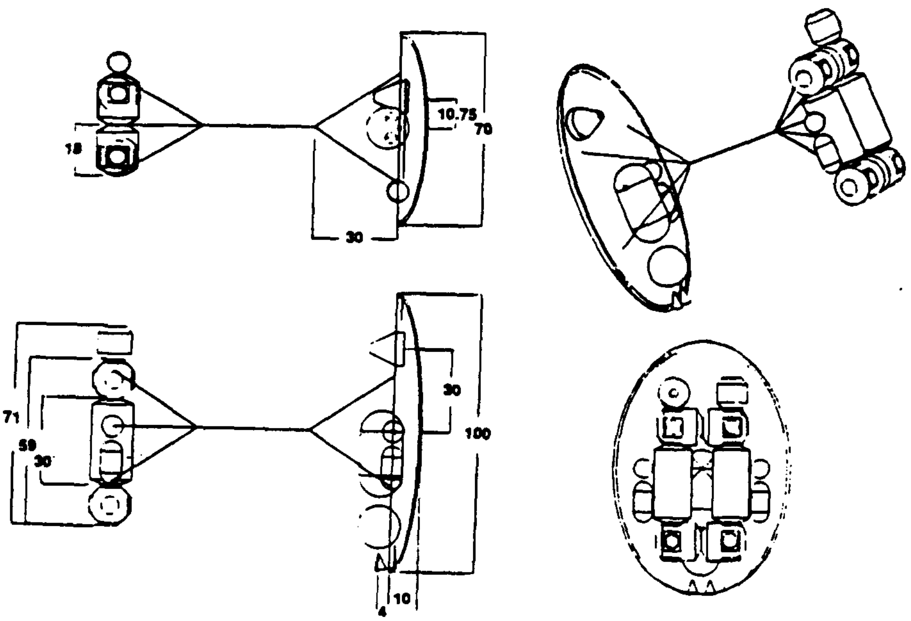


Figure 16. Manned Mars Artificial-G Vehicle Partially Deployed.

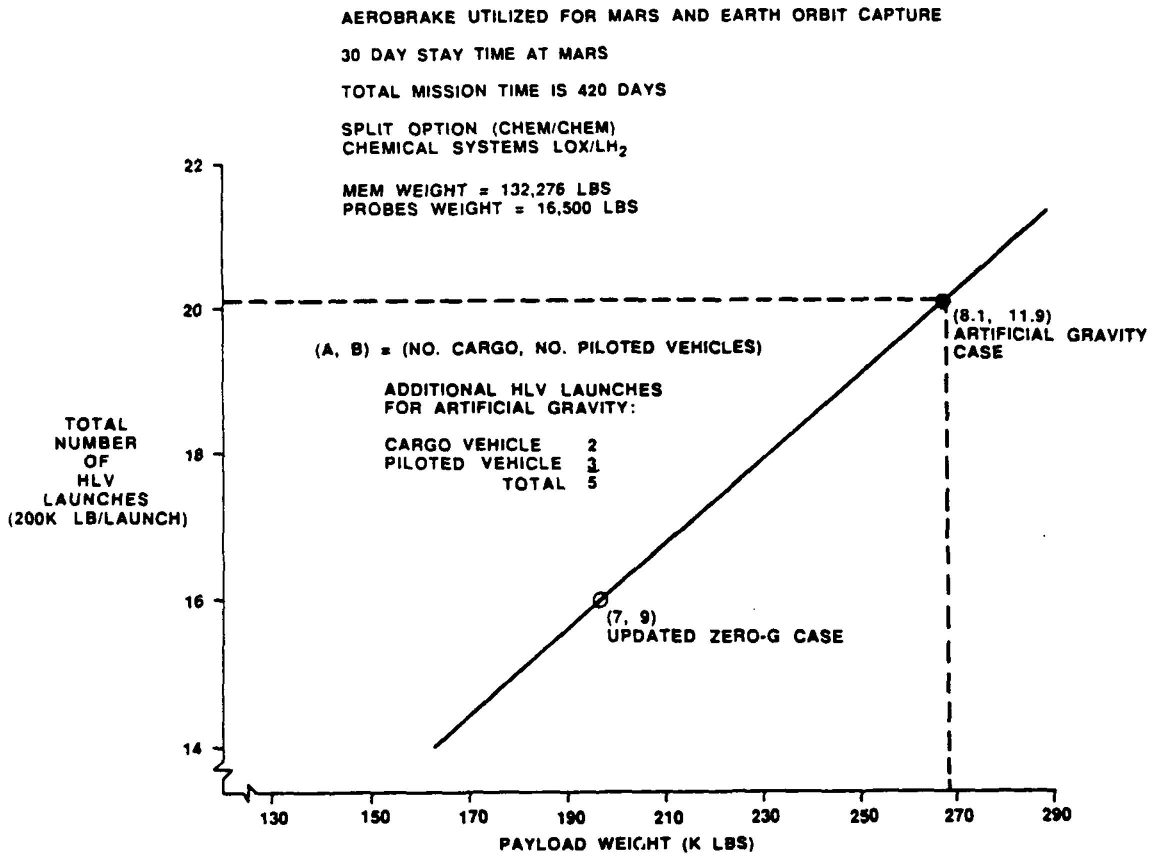


Figure 17. Earth-to-Orbit Launch Vehicle Requirements.

MANNED MARS

ARTIFICAL-GRAVITY VEHICLE STUDY

FINAL PRESENTATION

JULY 13, 1987

1-3868-7-9T

ARTIFICIAL-G FINAL REVIEW
AGENDA
JULY 13, 1987

TIME	SUBJECT	SPEAKER	MINUTES
2:30 PM	INTRODUCTION	BUTLER (MSFC)	5
2:35	EXECUTIVE SUMMARY	SCHULTZ (MFSC)	30
3:05	LIFE SCIENCES	SULZMAN (HQ)	20
3:25	DYNAMICS/TETHERS/OTHER STRUCTURES	RUPP/LEMKE (MSFC/ARC)	25
3:50	POWER	VALGORA (LeRC)	20
4:10	THERMAL/COMM./DATA MGMT.	LIVINGSTON (JSC)	15
4:25	SPACE STATION IMPLICATIONS	PRITCHARD (LaRC)	20
4:45	COST	RUTHERFORD (MSFC)	10
4:55	SCHEDULE/SUMMARY	BUTLER (MSFC)	5
5:00	ADJOURN		

INTRODUCTION

BUTLER

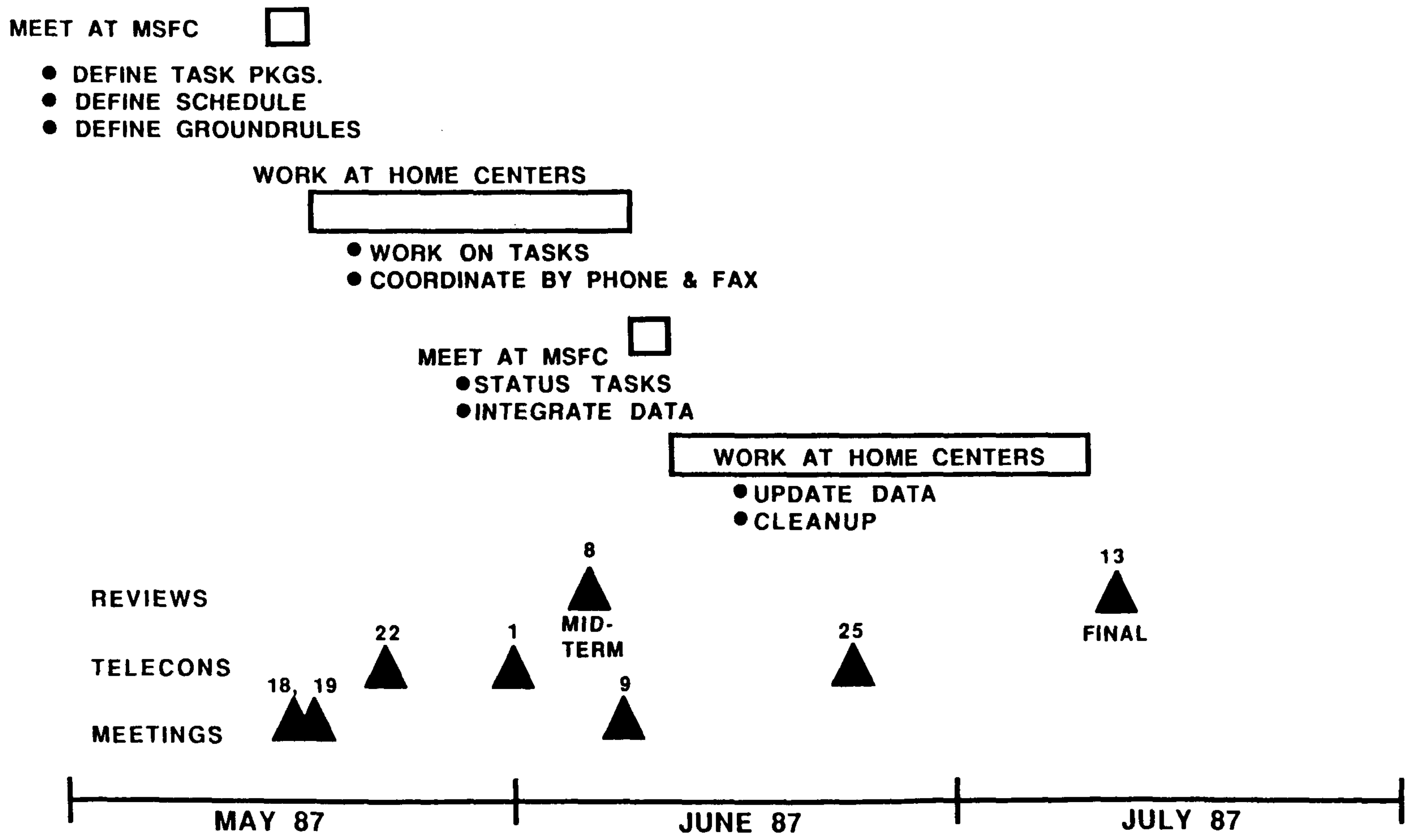
OBJECTIVES

- TO DEVELOP A FEASIBILE CONCEPT OF AN ARTIFICIAL-GRAVITY MANNED MARS VEHICLE SUITABLE FOR USE ON THE MARS SPLIT-MISSION INITIATIVE.

- TO PROVIDE CONCEPTUAL DESIGN, OPERATIONS, COST, TECHNOLOGY, & PROGRAMMATIC DATA FOR THE ARTIFICIAL-GRAVITY VEHICLE

1-3803-7-6D

STUDY SCHEDULE



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TASKS & RESPONSIBILITIES

5-28-87

TASKS	MSFC	JSC	LaRC	LeRC	ARC	KSC	HQ
	RESPONSIBILITIES						
1. OVERALL CONFIGURATION/INTEGRATION	L						
2. MODULES DEFINITION	L	*X					
3. EXTERNAL SYSTEMS DEFINITION	L	L		L			
4. TETHER/OTHER STRUCT. DEF./ANAL.	L				X		
5. LAUNCH SEQUENCE/ON-ORBIT ASSY./PREP.	X	(X)	L			X	(X)
6. MISSION OPERATIONS	L						
7. COREOLIS & CREW ACCOM.	X	L			X		X
8. MISSION PROFILE/TRAJECTORY	L						
9. PROGRAMMATICS	L						
10. COST	L						

L= LEAD RESPONSIBILITY, *NON-PRIME MANNED SYSTEMS, ( ) ON-CALL PARTICIPANT.

1-2372-7-4D

TEAM MEMBERS

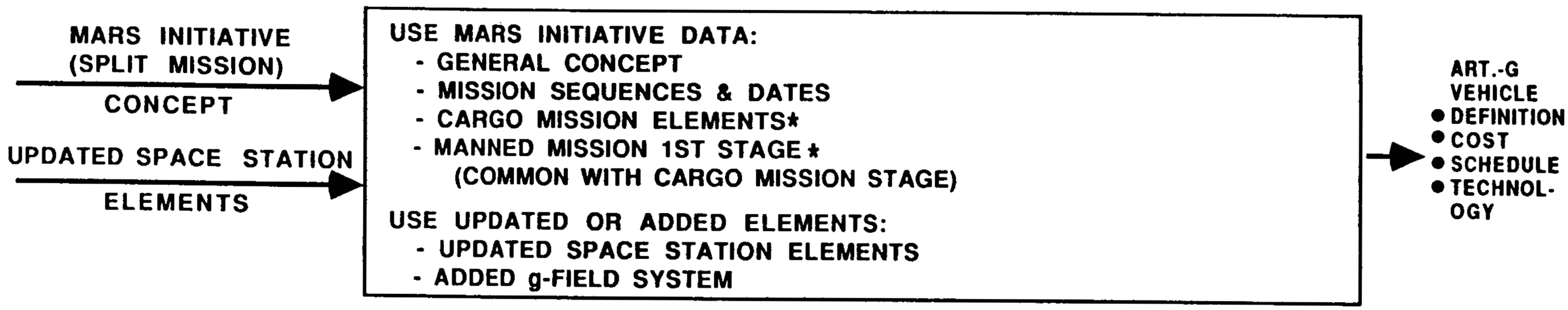
JULY 7, 1987

	MSFC	JSC	LaRC	LeRC	ARC	KSC	HQ
1.OVERALL CONFIG./ INTEGR.	BUTLER/SCHULTZ * HAJOS/HERMANN						
2.MODULES DEF.	SAME AS 1 AND OTHERS (MULTI-DISCIPLINE)	** LIVINGSTON/ BAILLIE					
3.EXTERNAL SYS. DEF.	SAME AS 1 AND OTHERS (MULTI-DISCIPLINE)	LIVINGSTON/ BAILLIE		VALGORA			
4.TETHER/OTHER STRUCT. SYSTEM	RUPP				LEMKE		
5.LAUNCH SEQ/ ASSY./ PREP.	BARISA/ VARNER	(ARMSTRONG)	PRITCHARD			PETERSON POWERS	(ASKINS)
6 MISSION OPERATIONS	BARISA/ VARNER						
7.COREOLIS & CREW ACCOM.	UPSHAW/HARWELL HALL	SEDDON HOMICK			BILLINGHAM		KEEFE
8.MISSION PROFILE/ TRAJECTORY	YOUNG						
9.PROGRAM-MATICS	TURNER						
10.COST	RUTHERFORD POWELL						

* OVERALL STUDY LEADER, ** NON-PRIME MANNED SYSTEMS, ( ) ON-CALL PARTICIPANT

GENERAL TASK FLOW

1-3802-7-6D



* ADJUSTED AS REQUIRED IF MANNED MISSION ELEMENTS CHANGE

EXECUTIVE SUMMARY

SCHULTZ

1-3502-7-8T

MANNED MARS MISSIONS - ARTIFICIAL GRAVITY ACCOMMODATION

- INTRODUCTION
- SPLIT OPTION SPRINT MISSION
- KEY SYSTEM OPTIONS
- SYSTEMS/ENGINEERING TRADES
- ABORT/DEGRADED MODES/SAFETY FACTORS
- CONFIGURATIONS
- WEIGHTS
- EARTH-TO-ORBIT LAUNCH VEHICLE REQUIREMENTS
- SUMMARY

MANNED MARS MISSION - ARTIFICIAL GRAVITY ACCOMMODATION

OBJECTIVES

- PERFORM FIRST ORDER DESIGN STUDY OF AN ARTIFICIAL-GRAVITY MARS SPACESHIP FOR MANNED MARS MISSIONS
- COMPARE MASS, COMPLEXITY, OPERATIONS, PROGRAMMATICS, AND COST WITH EQUIVALENT ZERO-GRAVITY DESIGN

APPROACH

- USE MARS INITIATIVE DATA FOR SPLIT OPTION MISSION TO THE EXTENT POSSIBLE
- UPDATE WITH CURRENT SPACE STATION ELEMENTS
- ADD ARTIFICIAL GRAVITY SYSTEM FOR PILOTED VEHICLE USING ROTATING ELEMENTS

MANNED MARS MISSION - ARTIFICIAL GRAVITY ACCOMMODATION

<u>GROUNDRULES</u>

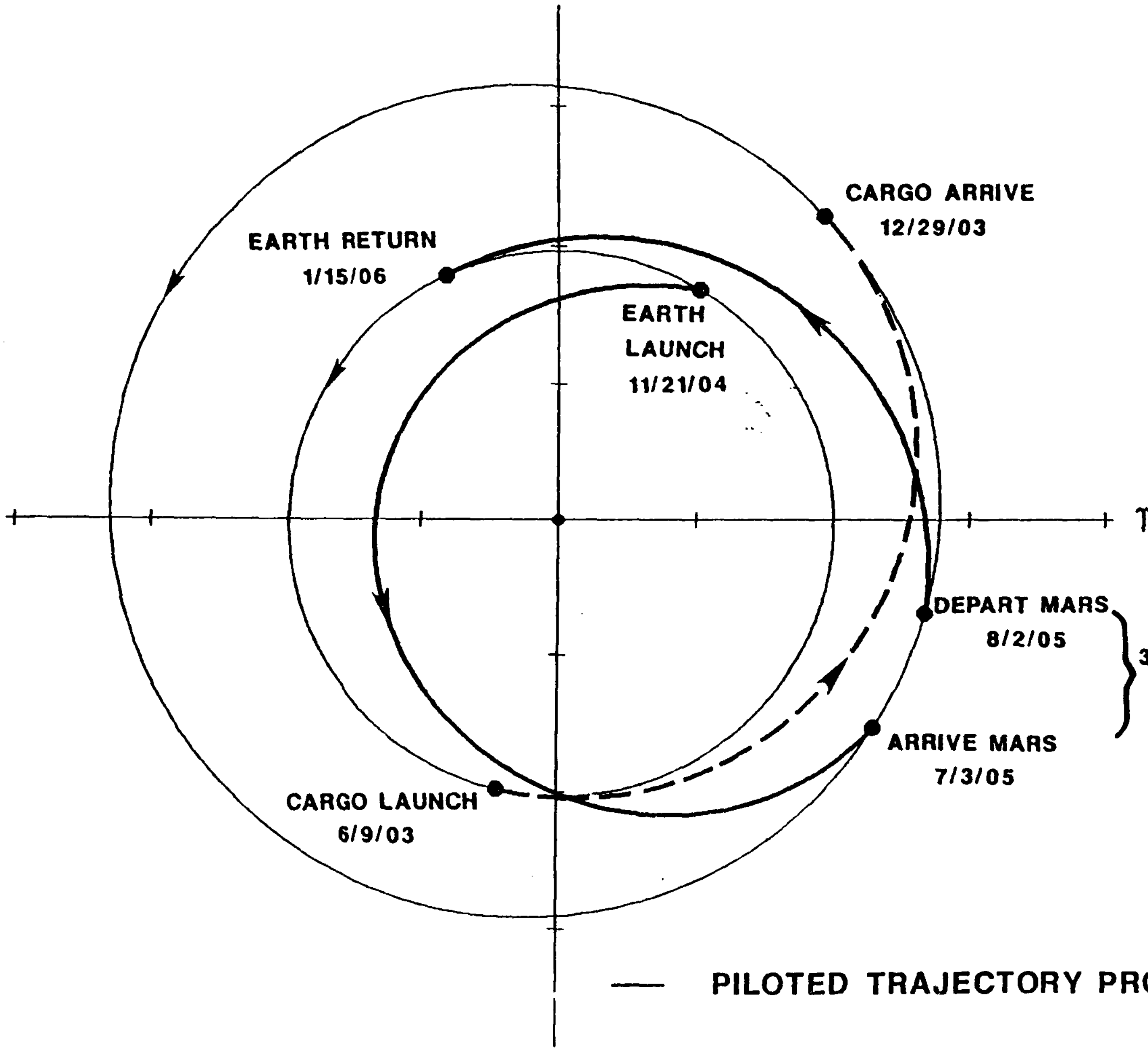
- USE SPLIT OPTION MISSION CONCEPT
  - CARGO VEHICLE PRECEEDS PILOTED VEHICLE
- USE EXISTING DEFINITION OF MARS CARGO VEHICLE ELEMENTS AND MARS SURFACE ACTIVITIES
  - NO NEW ASSESSMENT
- LEO DEPARTURE
  - CARGO VEHICLE: MID 2003 – CONJUNCTION TRAJECTORY
  - PILOTED VEHICLE: LATE 2004 – SPRINT TRAJECTORY
- CREW SIZE OF 6
- AERODYNAMIC BRAKING INTO MARS ORBIT AND INTO EARTH RETURN ORBIT
- GRAVITY LEVELS
  - 1G FOR MOST OF EARTH-MARS AND MARS-EARTH TRANSIT
  - 3/8G NEAR MARS INTERFACE
  - OG PRIOR TO MARS OR EARTH AEROBRAKING
  - NOT TO EXCEED 3.5G FOR MORE THAN 1 MINUTE WITH 5G MAXIMUM
- INITIALLY SIZE FOR SPIN RATE OF 2 RPM
  - UP TO 4 RPM ULTIMATELY MAY BE POSSIBLE
- PREFERRED MODULE ORIENTATION (FLOOR) PERPENDICULAR TO SPIN PLANE
- CRYO PROPELLANTS FOR EARTH-MARS & MARS-EARTH TRANSFER
- PROVIDE SAFE HAVEN FOR CREW SAFETY

MANNED MARS MISSION - ARTIFICIAL GRAVITY ACCOMMODATION (CONTINUED)

GROUNDRULES

- PROVIDE STORM PROTECTION FOR SOLAR FLARES WITH 15 MIN WARNING
- UTILIZE PLANNED EVA IN MARS ORBIT
- CONTINGENCY EVA DURING EARTH-MARS & MARS-EARTH TRANSIT
- CONSIDER AVAILABLE SPACE AND LAUNCH ELEMENTS FOR OPERATIONS AND DERIVATIVES OF THESE ELEMENTS FOR MARS VEHICLE COMPONENTS
  - SPACE STATION
  - OTV
  - OMV
  - SDV OR HLLV
  - STS
- SIMPLE, SAFE, LOW COST CONCEPT DESIRED

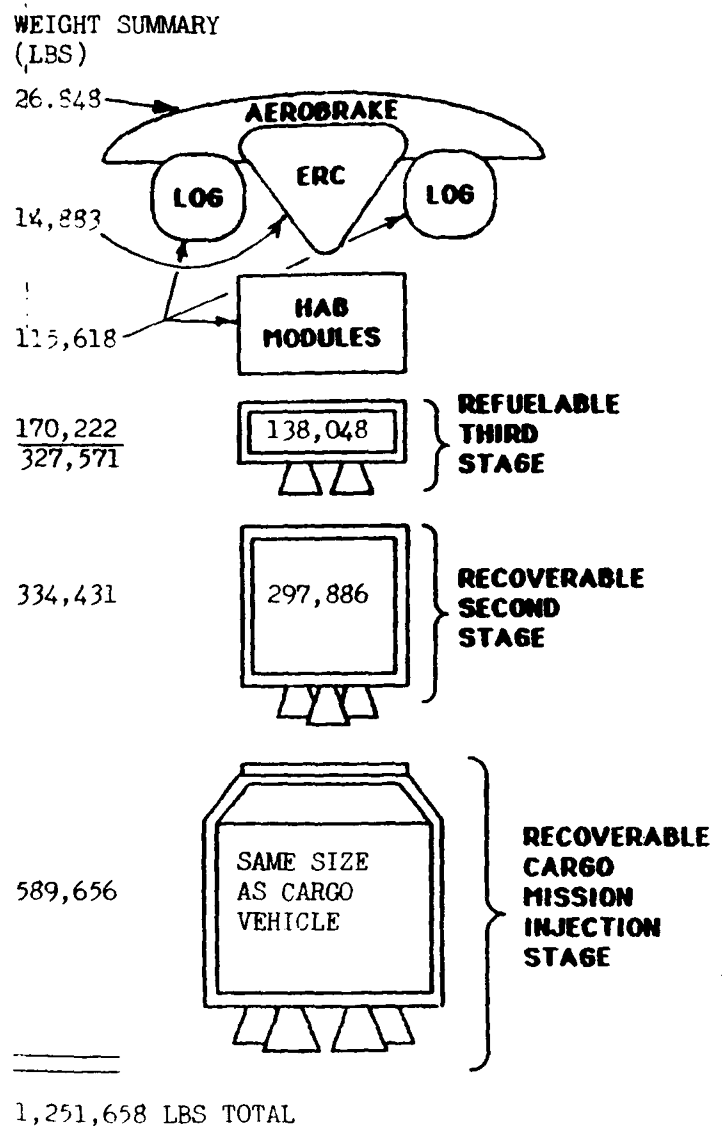
PILOTED MARS MISSION - SPLIT OPTION TRAJECTORY PROFILE

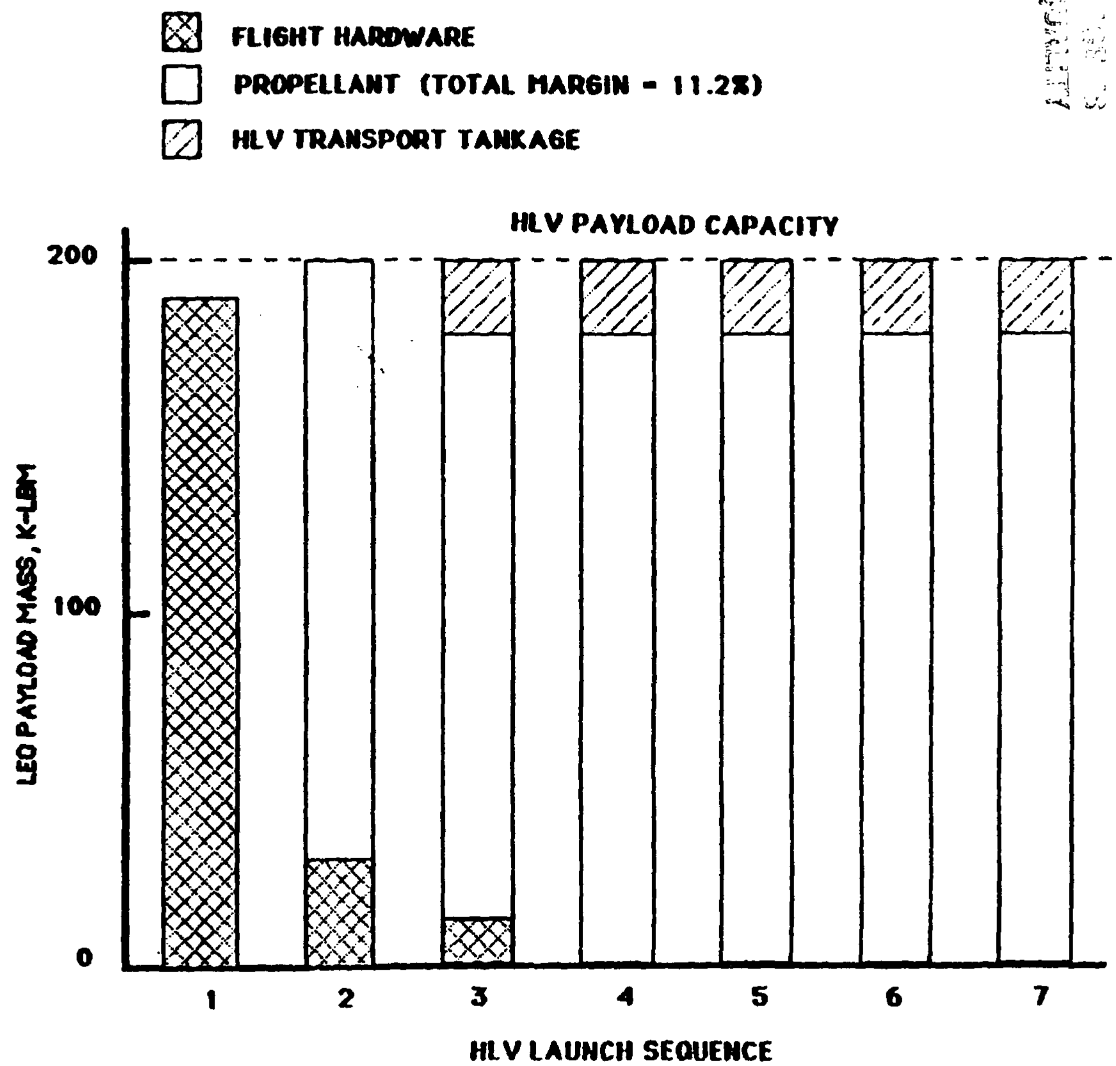


— PILOTED TRAJECTORY PROFILE

--- CARGO VEHICLE TRAJECTORY PROFILE

PILOTED MISSION BUILD-UP SCENARIO





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MANNED MARS MISSIONS - ARTIFICIAL GRAVITY ACCOMMODATION

KEY SYSTEM OPTIONS

- GRAVITY LEVEL: PARTIAL G VS. 1G
- ROTATION RATE: 2RPM VS. OTHER (UP TO 4RPM)
  - —CORIOLIS CONSIDERATIONS
  - —HEAD-TO-FOOT G-GRADIENT
- SEPARATED MODULES VS. AGGREGATED MODULES
- USE OF SPACE STATION TRASH OR SHUTTLE EXTERNAL TANK FOR COUNTERWEIGHT VS. USE OF ONLY REQUIRED ELEMENTS (SEPARATED AS NECESSARY)
- TWO-BODY VS. THREE-BODY MASSES
- BOOM VS. TRUSS VS. TETHER
- MULTIPLE VS. SINGLE TETHERS
- MODULE ORIENTATION IN G-FIELD IN SPIN PLANE VS.PERPENDICULAR TO SPIN PLANE
- ORIENTATION OF SPIN VECTOR: INERTIALLY-FIXED VS. SUN-TRACKING

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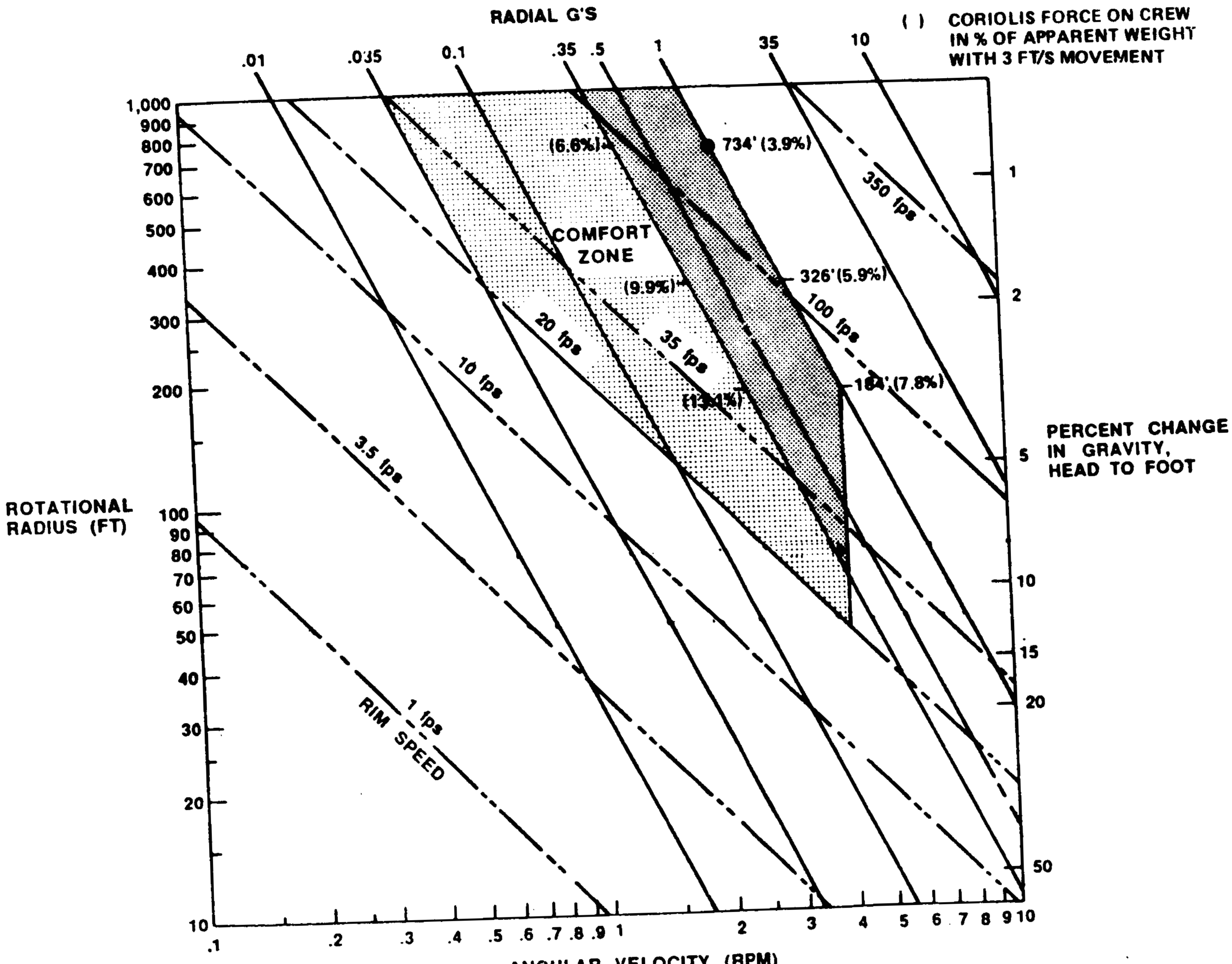
ROTATIONAL PARAMETERS AND COMFORT ZONE



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MANNED MARS MISSIONS - ARTIFICIAL GRAVITY ACCOMMODATION

MODULE ARRANGEMENT TRADES	SEPARATED	AGGREGATED
● SAFE HAVEN CONSIDERATIONS		
– NUMBER OF EXITS PER MODULE		√
– AVAILABILITY OF MULTIPLE VOLUMES WITH MINIMUM MODULES		√
● STORM SHELTER ASPECTS		
– RAPID ACCESS		√
– NEED FOR JUST ONE SHELTER		√
● PSYCHOLOGICAL CONSIDERATIONS		
– LARGE GROUP INTERACTION		√
– MINIMIZATION OF FEELINGS OF ISOLATION		√
● CREW SKILL MIX AVAILABLE		√
● SYSTEMS CONSIDERATIONS		
– MINIMUM NUMBER OF AIRLOCKS		√
– MASSES MORE EQUAL	√	

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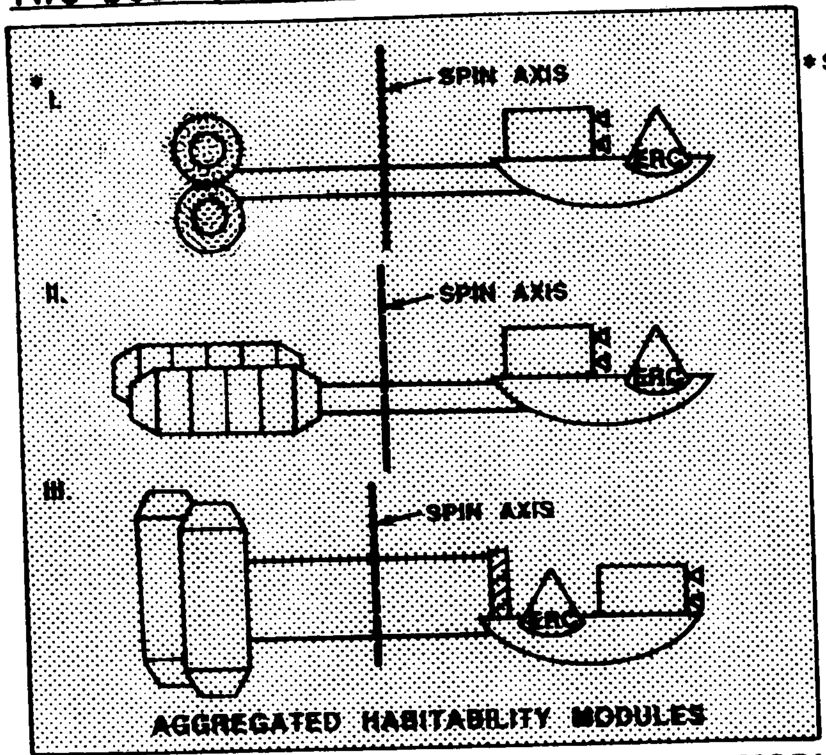
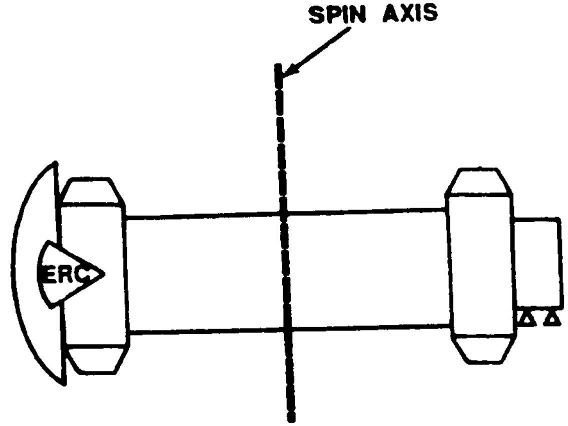
ARTIFICIAL GRAVITY CONFIGURATION OPTIONS

TWO BODY (AGGREGATED AND SEPARATED HABITABILITY MODULES)



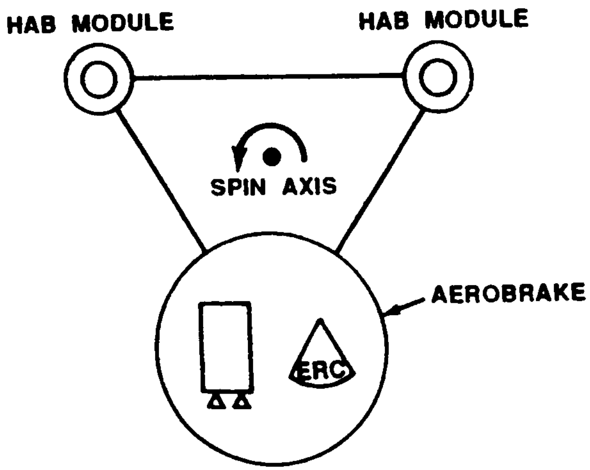
AGGREGATED HABITABILITY MODULES

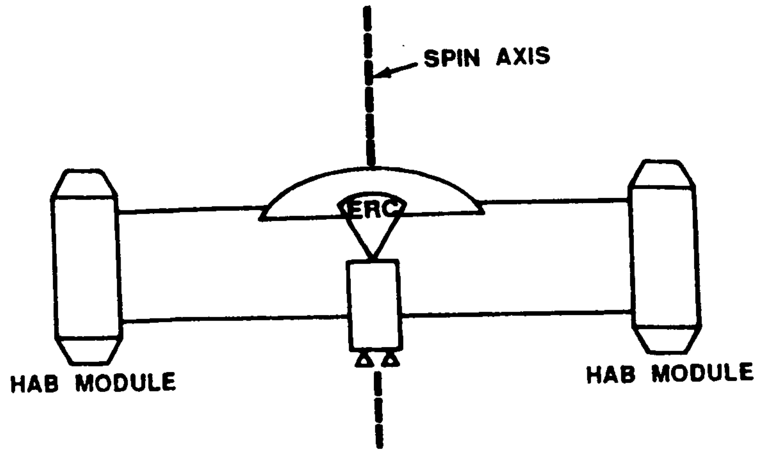
* SELECTED



SEPARATED HABITABILITY MODULES

THREE BODY (SEPARATED HABITABILITY MODULES)





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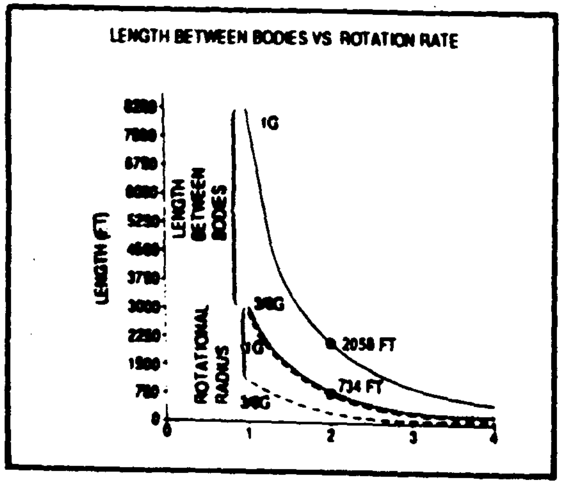
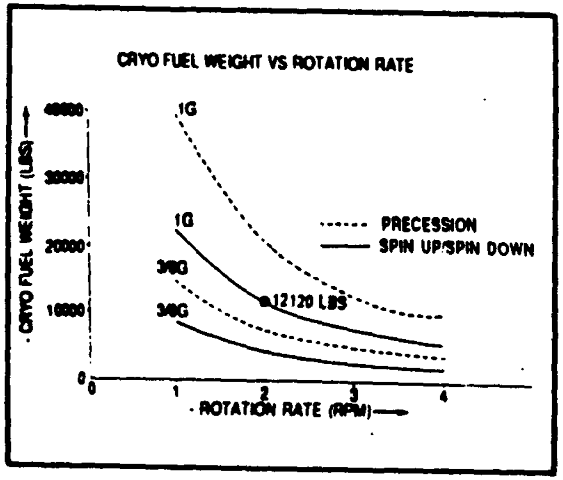
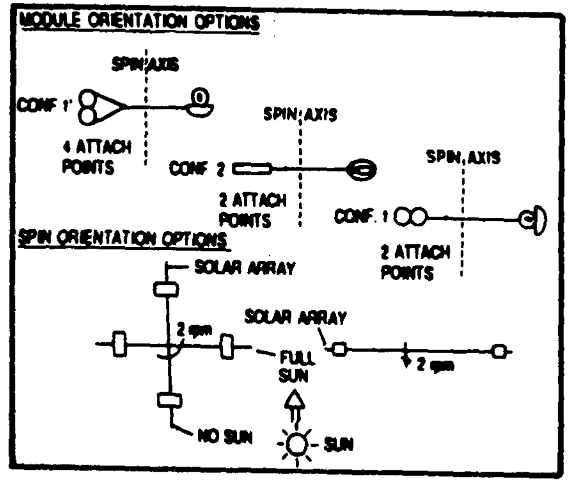
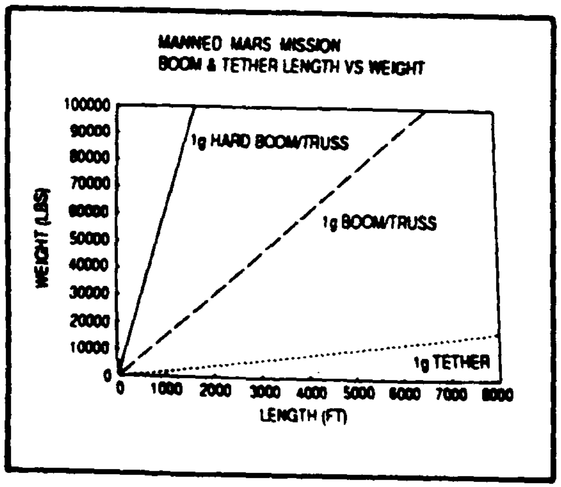
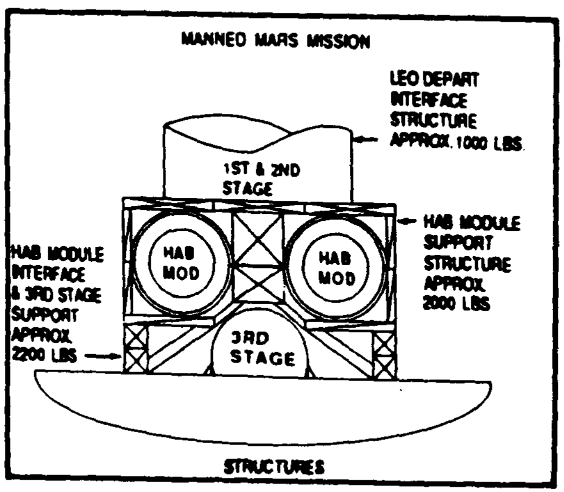
MANNED MARS MISSIONS - ARTIFICIAL GRAVITY ACCOMMODATION ENGINEERING TRADES

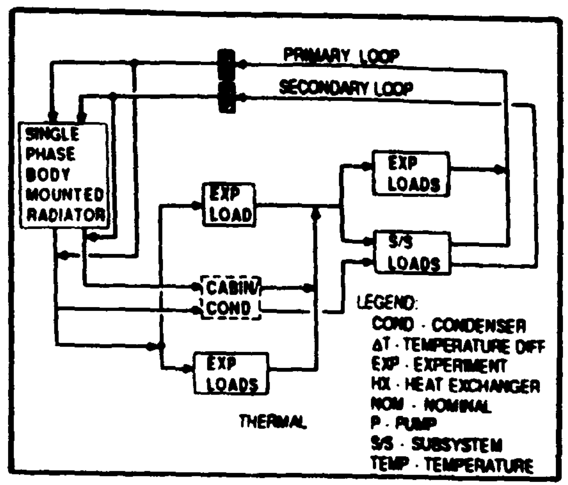












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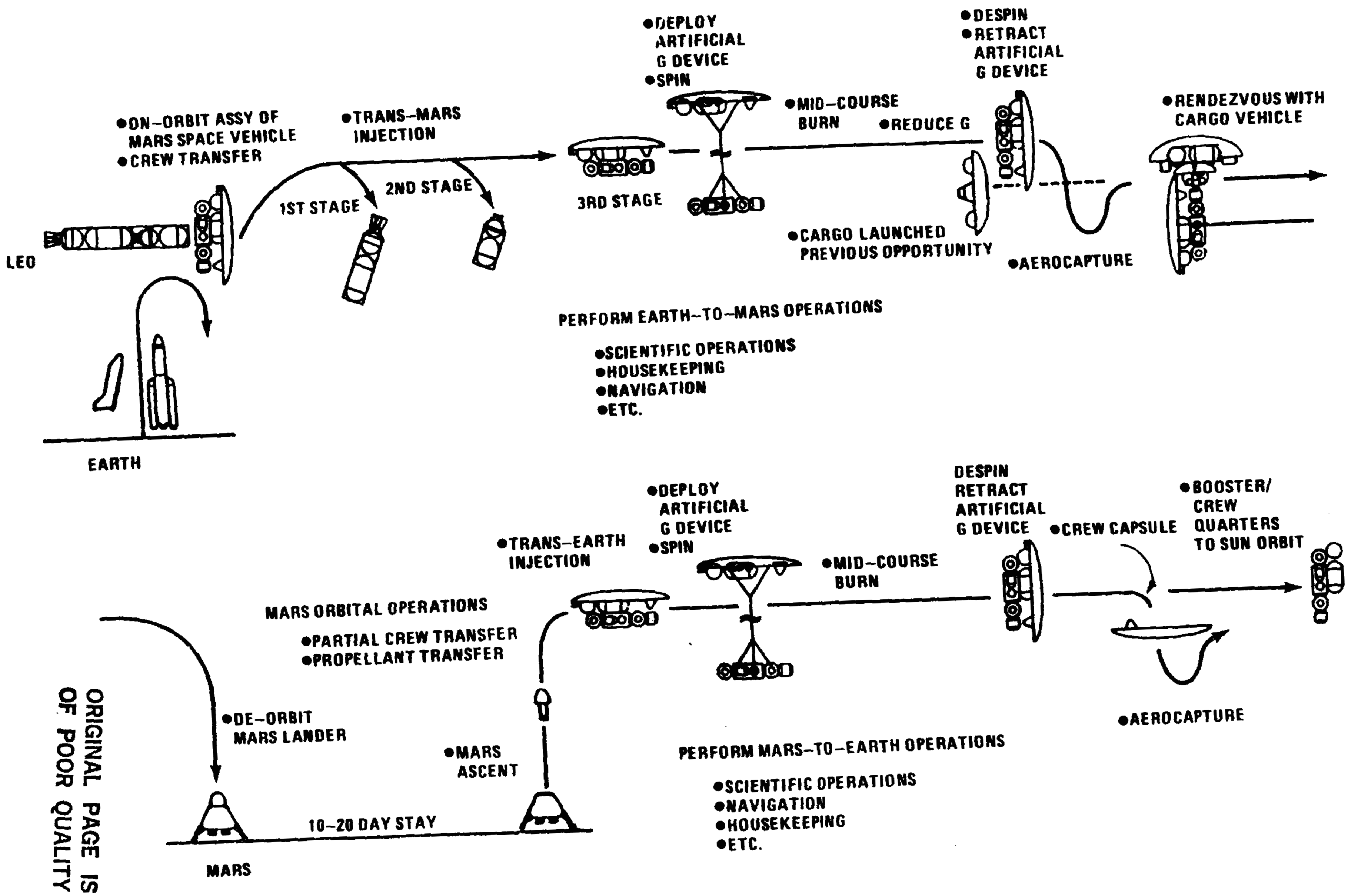
MARS PILOTED MISSION SCENARIO



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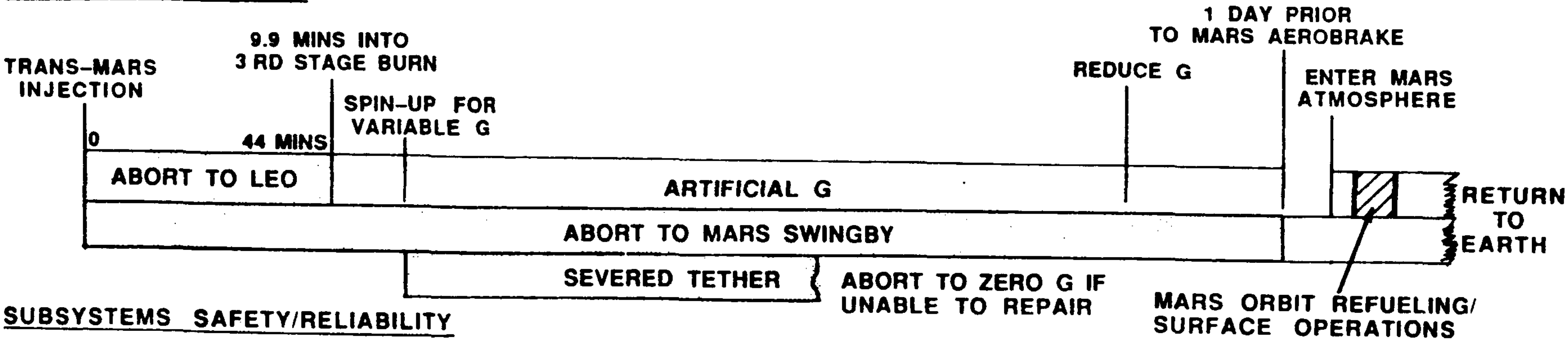
MANNED MARS MISSIONS – ARTIFICIAL GRAVITY ACCOMMODATION
ABORT/DEGRADED MODES/SAFETY FACTORS

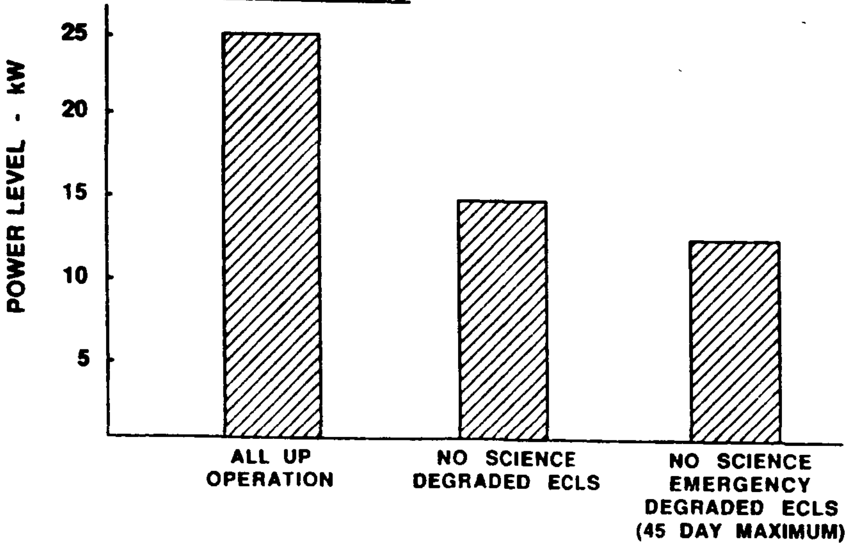
MISSION ABORT MODES



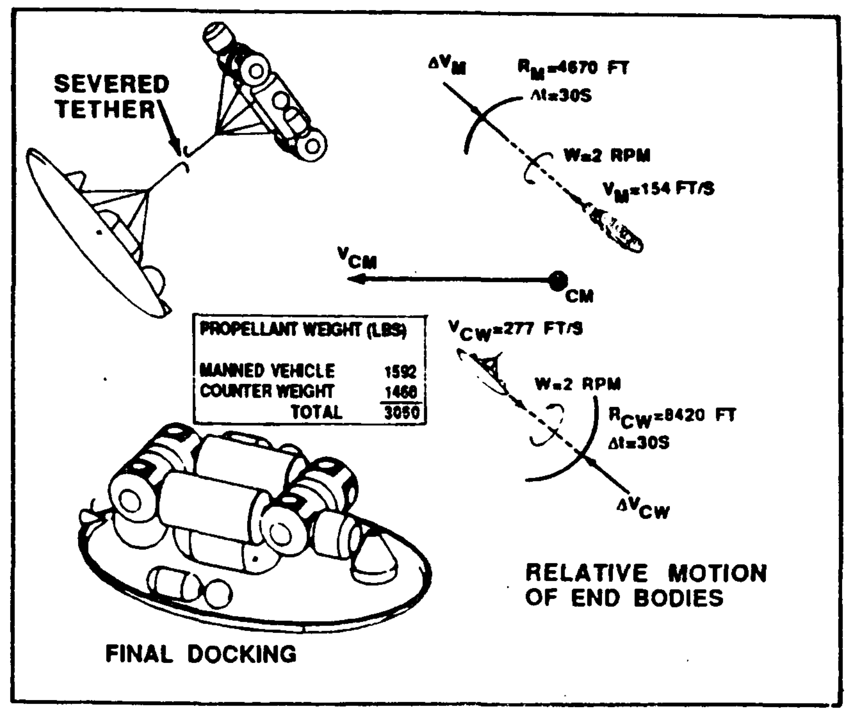
SUBSYSTEMS SAFETY/RELIABILITY

- TETHER/BOOM 2.0 FACTOR OF SAFETY
- SPACE STATION SUBSYSTEMS
- SUBSYSTEMS REDUNDANCY
- SPARES/MAINTENANCE
- CONTINGENCY PROPELLANT FOR SPINUP/SPINDOWN AND SEVERED TETHER RECOVERY

DEGRADED POWER MODES



SEVERED TETHER RECOVERY SEQUENCE



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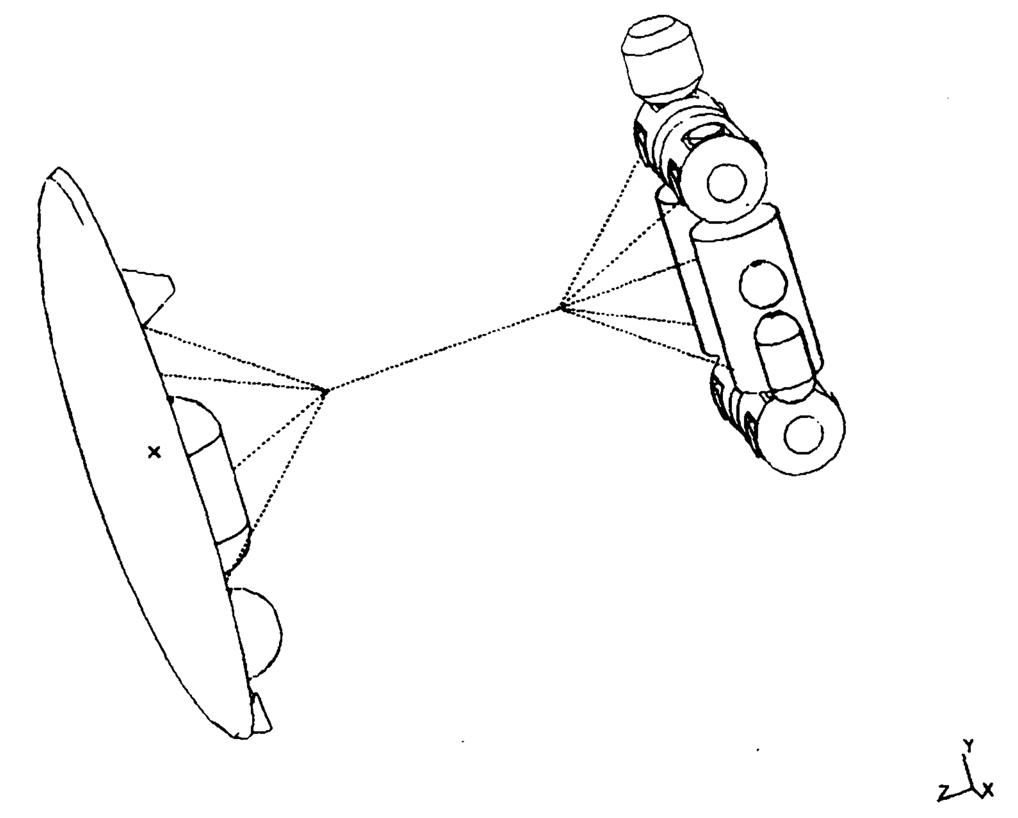
SDRC I-DEAS 3.1: System Assembly 12-JUN-87 13:02:05
DATABASE: MSFC/G.HAJOS MANNED MARS ARTIFICIAL 1G UNITS = BG
VIEW: ISOSORTOF2 DISPLAY: No stored OPTION
Task: HIERARCHY Bin: 1-MAIN
System: 11-MMMMCW (modified) Component: 2-AEROBRAKE

CONFIGURATION AT INITIAL DEPLOYMENT/ FINAL RETRIEVAL



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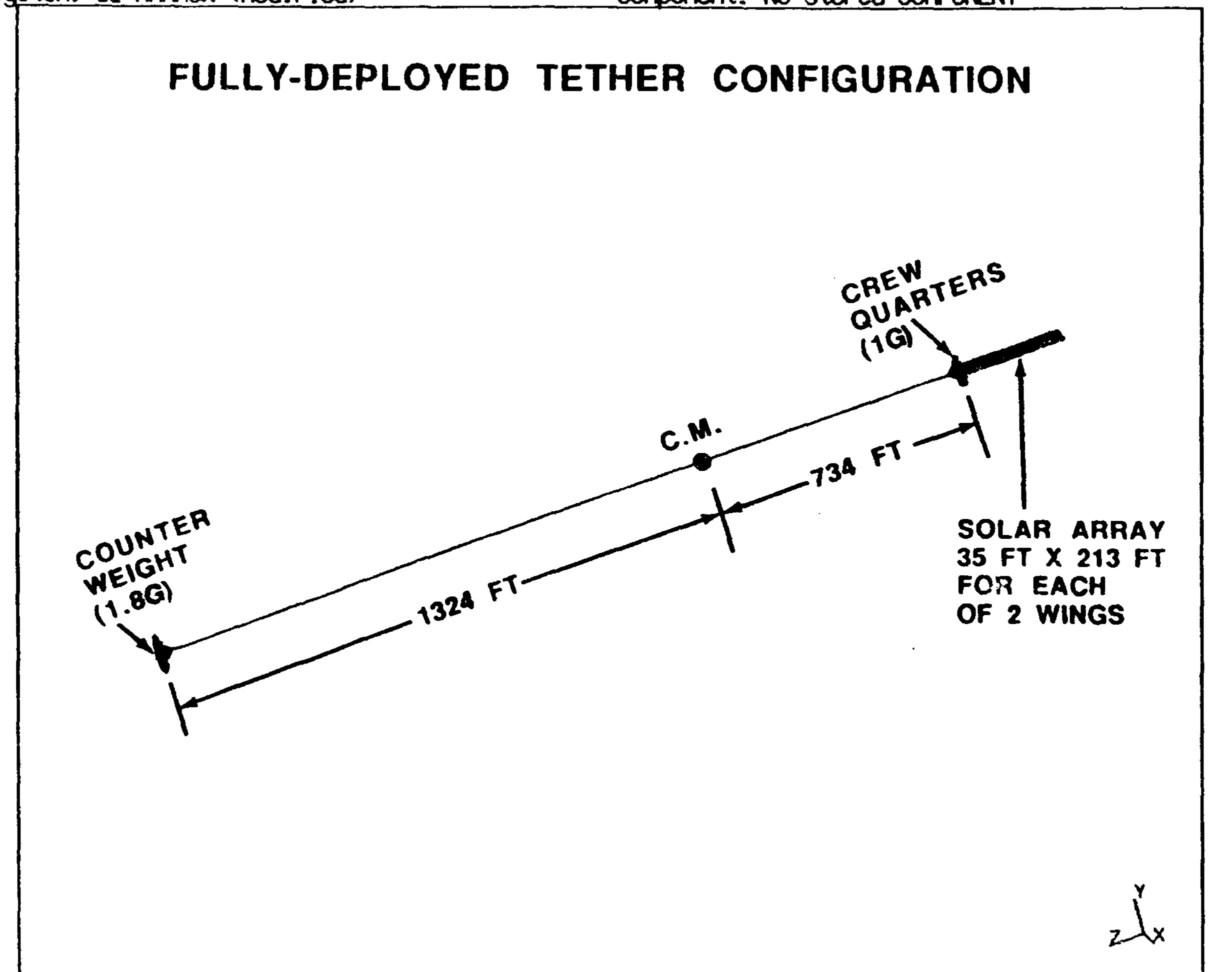
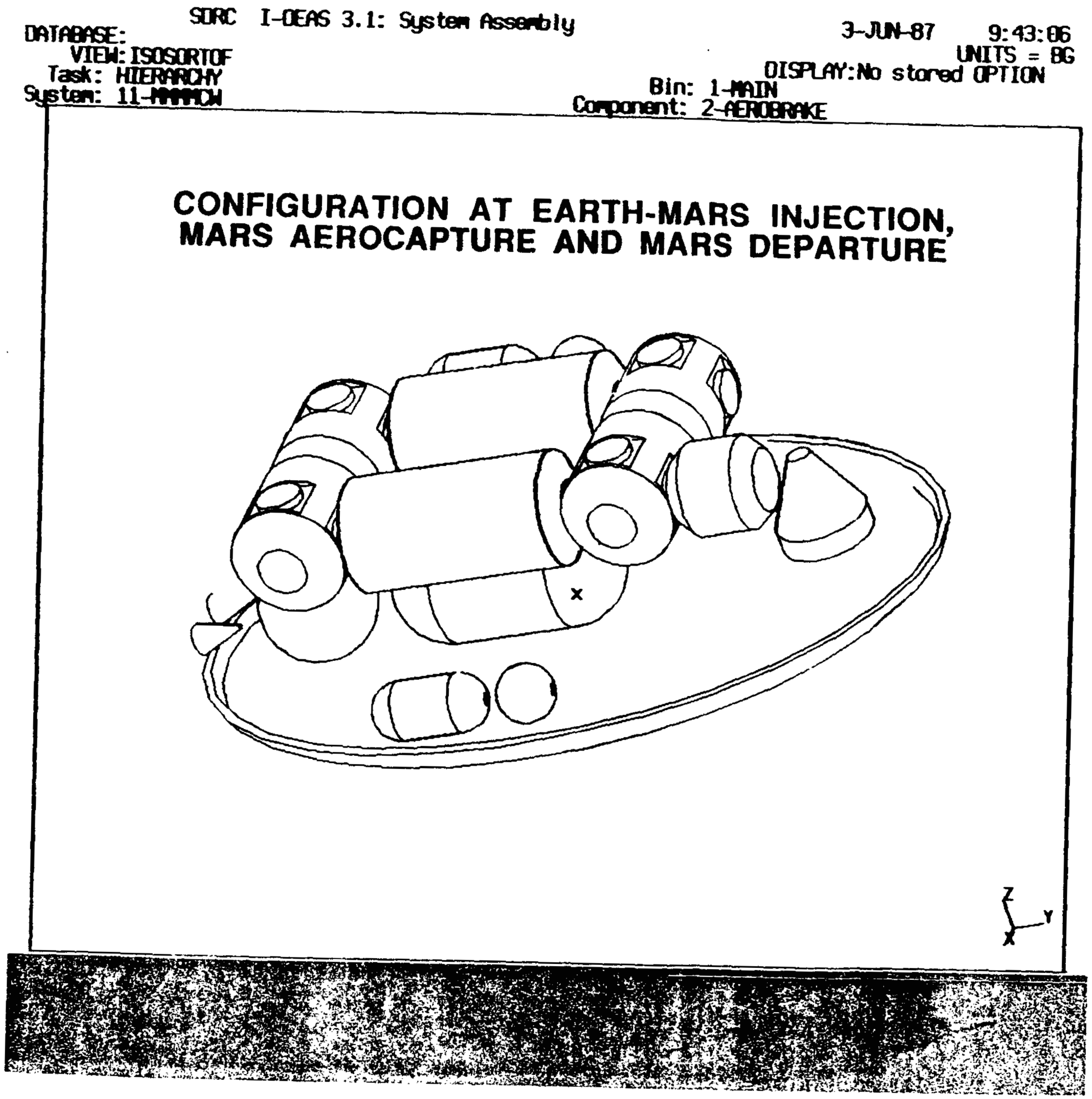
SDRC I-DEAS 3.1: System Assembly 24-JUN-87 7:34:22
DATABASE: UNITS = BG
VIEW:ISOSORTOF2 (modified) DISPLAY:No stored OPTION
Task: SYSTEM Bin: 1-MAIN
System: 11-MMMMCW (modified) Component: No stored COMPONENT

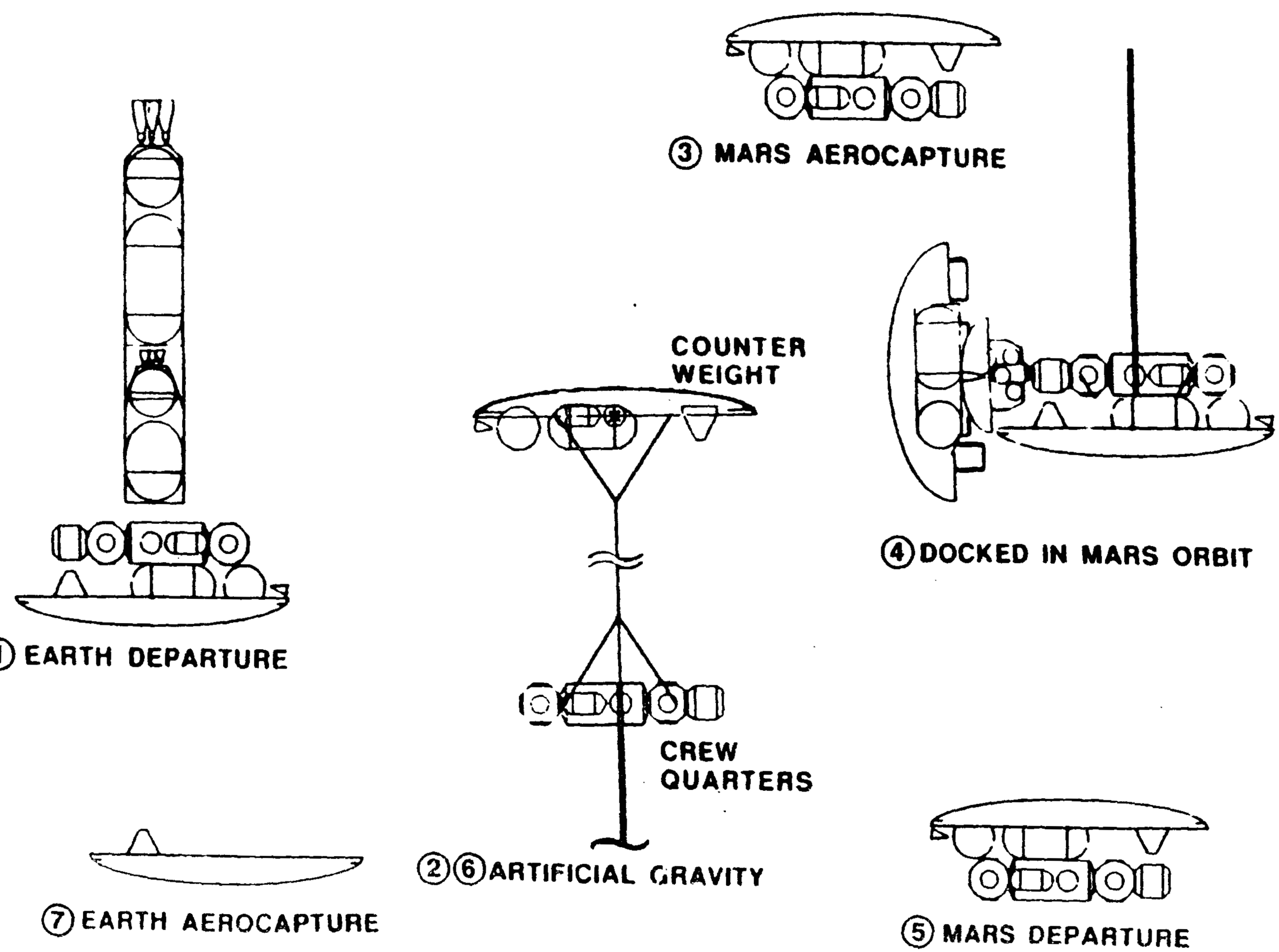


1-3967-7


SDRC I-DEAS 3.1: System Assembly
DATABASE:
VIEW: ISOSORTOF
Task: HIERARCHY
System: 11-MMMMCW
3-JUN-87 9:43:06
UNITS = BG
DISPLAY: No stored OPTION
Bin: 1-MAIN
Component: 2-AEROBRAKE
CONFIGURATION AT EARTH-MARS INJECTION,
MARS AEROCAPTURE AND MARS DEPARTURE
Z
Y
X

1-3973-7

MANNED MARS MISSIONS - ARTIFICIAL GRAVITY ACCOMMODATION
FLIGHT CONFIGURATIONS



MANNED MARS SPLIT MISSION
PILOTED VEHICLE WEIGHT SUMMARY (LBS)
(EXCLUDING PROPULSIVE STAGES)

DESCRIPTION	UPDATED OG	1G CASE	DELTA
COUNTER WEIGHT			
AEROBRAKE	29121	31605	2484
EARTH RETURN CAPSULE	14883	14883	
3rd STAGE	32174	32174	
AVIONICS		393	393
RCS SUBSYSTEM (15% CONT ON TANKS)		12001	12001
SUPPORT STRUCTURE	2838	4652	1814
SUBTOTAL	79016	95708	16692
MANNED MODULE ASSEMBLY			
MODULE STRUCTURE (2)	25864	25864	
NODE STRUCTURE (4)	24157	24157	
AIRLOCK	6000	6000	
THERMAL	9675	9675	
AVIONICS	7496	7496	
CREW SYSTEMS	9448	9448	
ECLSS	14768	14768	
MODULE/NODE CONTINGENCY (15%)	14611	14611	
SUPPLEMENTAL SHIELDING (STORM SHELTER)	3000	3000	
SPARES	5000	5000	
FLUIDS, THERMAL	432	432	
FLUIDS, ELECTRICAL	64	64	
ECLSS CONSUMABLE	6294	6294	
CREW SYSTEM CONSUMABLES	11267	11267	
TETHER (INC 15% CONT)		4335	4335
TETHER DEPLOYER (INC 15% CONT)		6009	6009
RCS SUBSYSTEM (15% CONT ON TANKS)		13710	13710
ELECTRICAL POWER SYSTEM	4470	8466	3996
SUPPORT STRUCTURE	1589	3994	2405
MISSION SCIENCE	5168	5168	
CREW (6)	1140	1140	
SUBTOTAL	150443	180898	30455
TOTAL SYSTEM WEIGHT(*ADJ FOR PERFORMANCE)	229459 (196383)*	276606 (269276)*	47147

1-3765-7

EARTH-TO-ORBIT LAUNCH VEHICLE REQUIREMENTS

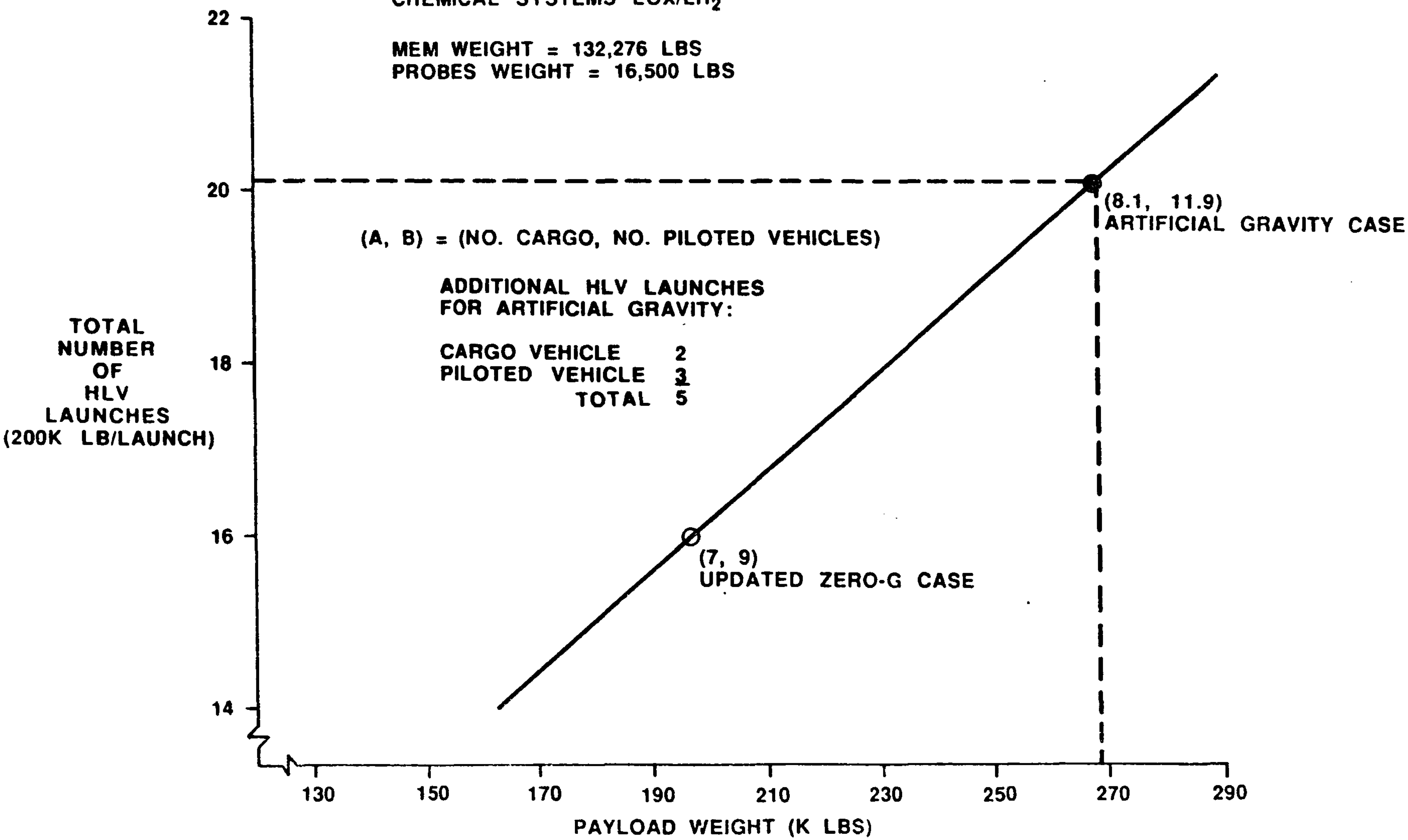
AEROBRAKE UTILIZED FOR MARS AND EARTH ORBIT CAPTURE

30 DAY STAY TIME AT MARS

TOTAL MISSION TIME IS 420 DAYS

SPLIT OPTION (CHEM/CHEM)
CHEMICAL SYSTEMS LOX/LH_2

MEM WEIGHT = 132,276 LBS
PROBES WEIGHT = 16,500 LBS



ARTIFICIAL G VEHICLES OVERALL WEIGHT SUMMARY

	UPDATED ZERO-G WET WEIGHT (LBS)	ARTIFICIAL G WET WEIGHT (LBS)	DELTA WET WEIGHT (LBS)
PILOTED VEHICLE			
• COUNTERWEIGHT	79,016	95,708	16,692
• MANNED MODULE ASSY	150,443	180,898	30,455
• STAGE 1	727,974	865,579	137,605
• STAGE 2	359,952	572,679	212,727
• STAGE 3	202,903	282,740	79,837
TOTAL	1,520,288	1,997,604	477,316
CARGO VEHICLE			
• PAYLOAD	427,403	519,240	91,837
• STAGE 1	727,974	865,579	137,605
TOTAL	1,155,377	1,384,819	229,442

1-3503-7-8T

MANNED MARS MISSIONS - ARTIFICIAL GRAVITY ACCOMMODATION

OBSERVATIONS

- CHOICE OF ELEMENTS FOR CREW QUARTERS AND COUNTERWEIGHT IS A KEY CONSIDERATION
- DYNAMICS IS A KEY CONFIGURATION DRIVER
    - SEPARATION, DEPLOYMENT, RETRACTION, RE-ATTACHMENT
    - SPINUP AND SPINDOWN PROFILE
    - STABLE END BODY ORIENTATION
    - SPIN AXIS ORIENTATION
    - PRECESSION CONTROL
- LARGE SEPARATION DISTANCE BETWEEN (2058 FT) BETWEEN CREW QUARTERS AND COUNTERWEIGHT REQUIRED FOR 1 G AND 2 RPM
- TETHER SYSTEM IS THE ONLY REASONABLE STRUCTURAL CHOICE FOR LONG SEPARATION DISTANCES
- G-LEVELS LOWER THAN 1G AND SPIN RATES UP TO 4 RPM WOULD SIGNIFICANTLY REDUCE THE SIZE OF THE ARTIFICIAL G SYSTEM
    - AT PRESENT LIFE SCIENCE DATA NOT SUFFICIENT TO PERMIT THESE CHOICES
- ALL CREW SYSTEMS AND HUMAN FACTORS REQUIREMENTS CAN BE MET EXCEPT DESIRE FOR MODULE ORIENTATION (FLOOR) PERPENDICULAR TO SPIN PLANE
    - JUDGED TO BE SECOND-ORDER CONSIDERATION
- HIGH DEGREE OF COMMONALITY BETWEEN AN ARTIFICIAL G AND A ZERO G VEHICLE IS POSSIBLE
- TETHERED ARTIFICIAL G MARS VEHICLE FEASIBLE AND CONTROLLABLE WITH REASONABLE WEIGHT IMPACTS
    - VARIETY OF SAFETY/RELIABILITY FEATURES CAN BE PROVIDED

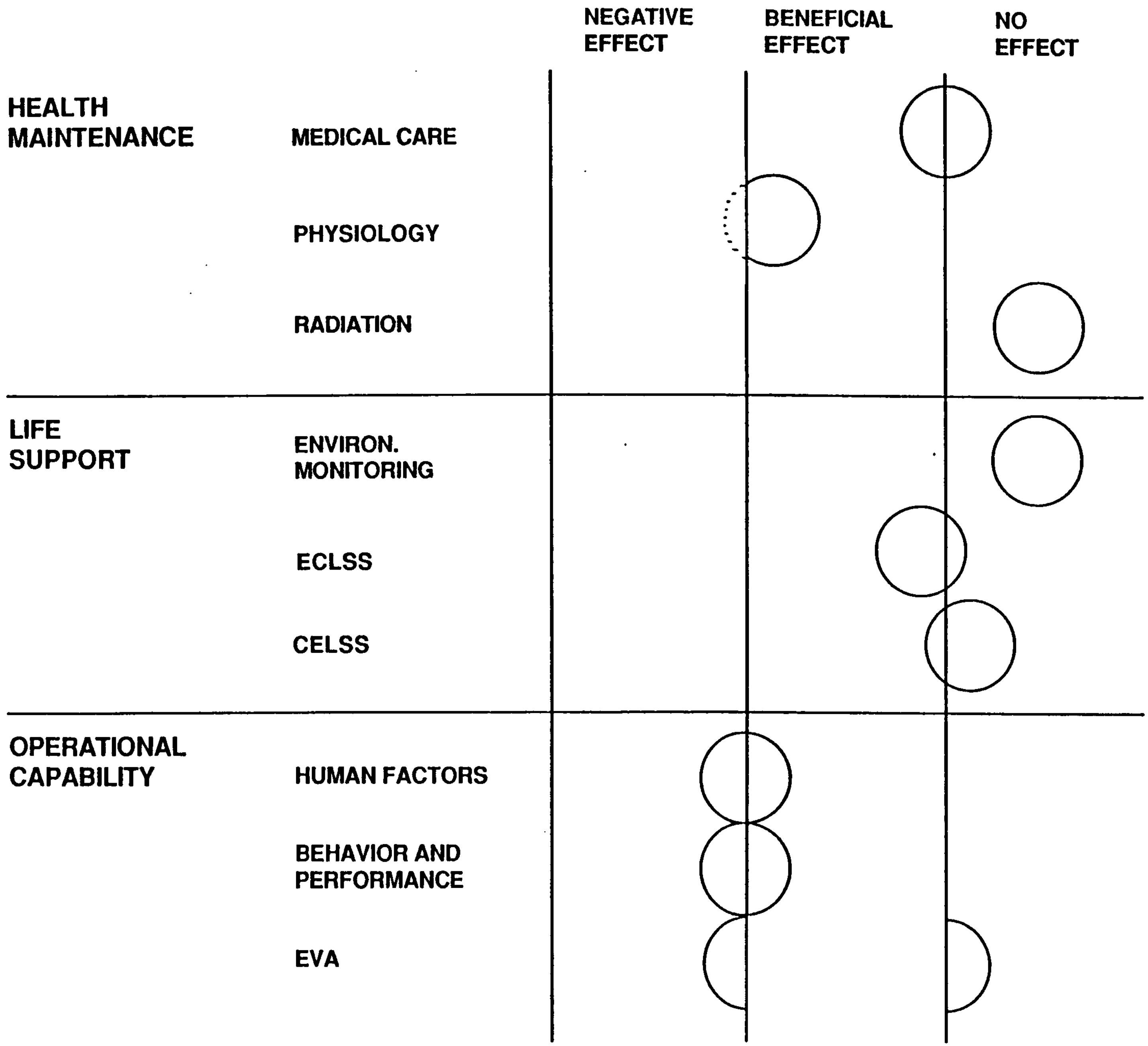
LIFE SCIENCES

SULZMAN

LIFE SCIENCES AREAS OF EMPHASIS

- HEALTH MAINTENANCE
  - MEDICAL CARE
  - MICROGRAVITY PHYSIOLOGY
  - RADIATION
- LIFE SUPPORT
  - ENVIRONMENTAL MONITORING
  - CELSS (BIO)
  - ECLSS (PHYS-CHEM)
- OPERATIONAL CAPABILITY
  - HUMAN FACTORS/TECHNOLOGY
  - BEHAVIOR AND PERFORMANCE
  - EVA

ARTIFICIAL GRAVITY AND IMPACT ON LIFE SCIENCES CONCERNS



MOST CRITICAL ISSUE

ZERO-G OR ARTIFICIAL-G FOR LONG-DURATION FLIGHT AND HABITATION?

- DECISION FOR MARS MISSION REQUIRES PROOF OF EFFICACY OF PHYSIOLOGICAL COUNTERMEASURES AND ARTIFICIAL GRAVITY

- REQUIREMENTS FOR ARTIFICIAL-G AFFECT BUDGET, SCHEDULES, SPACECRAFT DESIGN, AND CREW ACTIVITIES

COUNTERMEASURES: BACKGROUND

- EXERCISE, DIET, DRUGS AND OTHER COUNTERMEASURES CAN PREVENT CARDIOVASCULAR AND MUSCULOSKELETAL DECONDITIONING DURING LONG DURATION SPACE FLIGHT
- PHYSIOLOGICAL COUNTERMEASURES WILL BE REQUIRED FOR SPACE STATION
- COUNTERMEASURES WILL BE REQUIRED AS A BACK-UP FOR ARTIFICIAL GRAVITY
- USE OF COUNTERMEASURES MAY LOWER ARTIFICIAL GRAVITY REQUIREMENT

STATUS OF U.S. COUNTERMEASURES

- **EXERCISE:**
  - SKYLAB RESULTS INDICATE VIGOROUS EXERCISE MAINTAINS INFLIGHT WORK CAPACITY
  - 4 HRS OF DAILY AMBULATION DURING BED REST PREVENTS CALCIUM LOSS
  - 1 HR OF DAILY EXERCISE MAINTAINS WORK CAPACITY DURING BED REST
  - GROUND BASED STUDIES SHOW THAT SUBMAXIMAL SUSTAINED EXERCISE INCREASES RISK OF KIDNEY STONE FORMATION

- **DIET:**
  - HIGH CALORIE DIET IS NECESSARY TO MAINTAIN MUSCLE MASS
  - HIGH PROTEIN DIET INCREASES CALCIUM LOSS IN BED REST
  - MODERATE PROTEIN DIET SLOWS CALCIUM LOSS IN BED REST
  - LOW ANIMAL PROTEIN DIET REDUCES RISK OF KIDNEY STONES
  - FLUID LOADING IMPROVES BLOOD PRESSURE CONTROL AFTER SHORT SPACEFLIGHT

- **DRUGS:**
  - SODIUM FLUORIDE SLOWS CALCIUM LOSS IN BED REST
  - DIPHOSPHONATE* STOPS BONE LOSS IN BED REST

- **LBNP:**
  - LOWER BODY NEGATIVE PRESSURE INFLIGHT MAY IMPROVE POST-FLIGHT BLOOD PRESSURE CONTROL

- **ARTIFICIAL GRAVITY:**
  - NEVER TESTED WITH HUMANS IN SPACE

* - APPROVED FOR HUMAN USE ONLY FOR RESEARCH STUDIES

BENEFITS OF ARTIFICIAL GRAVITY

- PREVENT CARDIOVASCULAR AND MUSCULAR DECONDITIONING
- PREVENT BONE MINERAL LOSS
- ELIMINATE FLOATING DEBRIS FROM CREW HABITAT
- EASE WASTE DISPOSAL PROBLEMS

POTENTIAL PROBLEMS OF ARTIFICIAL GRAVITY

PHYSIOLOGICAL EFFECTS OF CORIOLIS FORCES

- PERFORMANCE
  - LOCOMOTION
  - WORKING ACTIVITIES
- NEUROVESTIBULAR
  - MOTION SICKNESS
  - HABITUATION

ARTIFICIAL GRAVITY
ADAPTATION UNKNOWNS

- ADAPTATION CHARACTERISTICS TO CROSS-COUPLED Z-AXIS ACCELERATION FOR SUBJECTS
- CHANGE IN ADAPTATION CHARACTERISTICS TO ANGULAR ACCELERATION IN ZERO GRAVITY ENVIRONMENT
- PERSISTENCE OF ADAPTATION
- EFFECTS OF TRANSITIONING BETWEEN 1G AND 0G
- ABILITY TO DEVELOP AND MAINTAIN DUALLY ADAPTED STATE
- FACILITATING ADAPTATION BY VISUAL CUES
- HAND-EYE AND HEAD-EYE COORDINATION DURING ROTATION
- APPLICABILITY OF SLOW ROTATING ROOM DATA TO 0G ROTATING HABITAT ADAPTATION

PILOTED MARS MISSION SIMULATION

- SIMULATIONS HAVE PRECEDED ALL PREVIOUS NASA PILOTED MISSIONS

- EXAMPLES:

APOLLO	1/6 G LUNAR TRAINER
SKYLAB	SMATC
SHUTTLE	WETF
SPACELAB	SMD 1, 2 & 3

- PILOTED MARS MISSION WILL ALSO REQUIRE A SIMULATION

  - VERIFY COUNTERMEASURES / ARTIFICIAL G HABITAT
  - TEST OPERATIONS AND PROCEDURES
  - VERIFY HARDWARE

- PILOTED MARS MISSION PRESENTS NEW CHALLENGES

  - ? DURATION
  - ?EARTH vs SPACE
  - ? LEVEL OF INTEGRATION

DYNAMICS/TETHERS/OTHER STRUCTURES

RUPP/LEMKE

ARTIFICIAL GRAVITY DYNAMICS CONSIDERATIONS

- CONFIGURATION
  - THREE BODIES
  - TWO BODIES
- TRANSPORTATION BETWEEN BODIES
  - EVA
  - ELEVATOR
- SPIN UP/SPIN DOWN
  - FUEL VS. MASS RATIO
  - INITIALIZATION
- STRUCTURAL DESIGN
  - MODULE DESIGN
    - SPACE STATION
    - NEW DESIGN
  - SUPPORT STRUCTURE
    - TRUSS
    - TETHER
- SPIN AXIS ORIENTATION
  - ALONG SUN-LINE
  - PERPENDICULAR TO SUN-LINE
- END BODY ORIENTATION
  - GYROSCOPIC STABILITY
  - THERMAL REQUIREMENTS
  - SOLAR ARRAY POINTING

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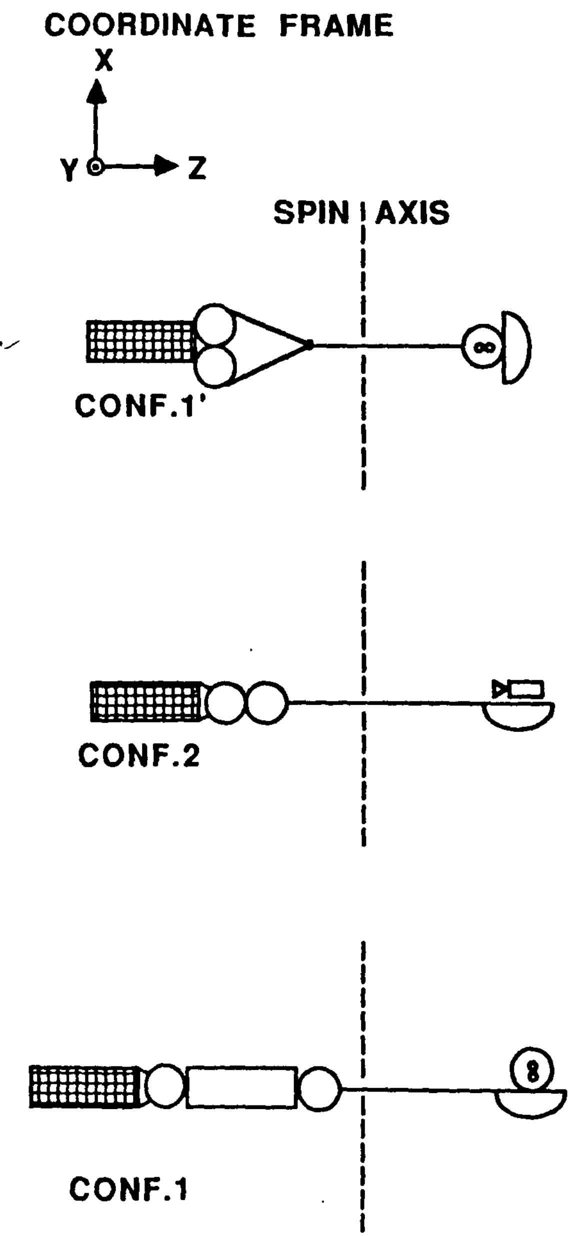

THREE STABLE ORIENTATIONS

	MOMENTS OF INERTIA, SLUG-FT2		EULER MOMENTS, FT-LB/DEG	
	CREW END	SHIELD END	CREW END	SHIELD END
X	3,000,601	1,956,467	0	0
Y	2,643,672	881,674	-679	385
Z	2,112,561	2,460,166	-273	-822
X	4,223,698	2,460,166	0	0
Y	3,006,507	1,956,467	-2115	-1208
Z	1,460,318	881,674	-931	-385
X	4,762,152	2,460,166	0	0
Y	4,037,350	881,674	-2904	-385
Z	967,929	1,956,467	-554	-1208

TETHER DEPLOYER

STS TETHER DEORBIT EXAMPLE

DEPLOYER MASS 5,255 LB (2,380 KG)

VOLUME OF TETHER 48 FT^3 (1.31 M^3)

MANNED MARS REQUIREMENT

VOLUME OF TETHER 65 FT^3 (1.73 M^3)

CONCLUSION

DEPLOYER MASS 5,255 LB (2,389 KG) (MOTORS, REEL, STRUCTURE, CONTROLLER, NO TETHER)

APPROXIMATE SIZE: 2.2 M DIA X 0.7 M LONG

TETHER MASS 3,770 LB (1713 KG)

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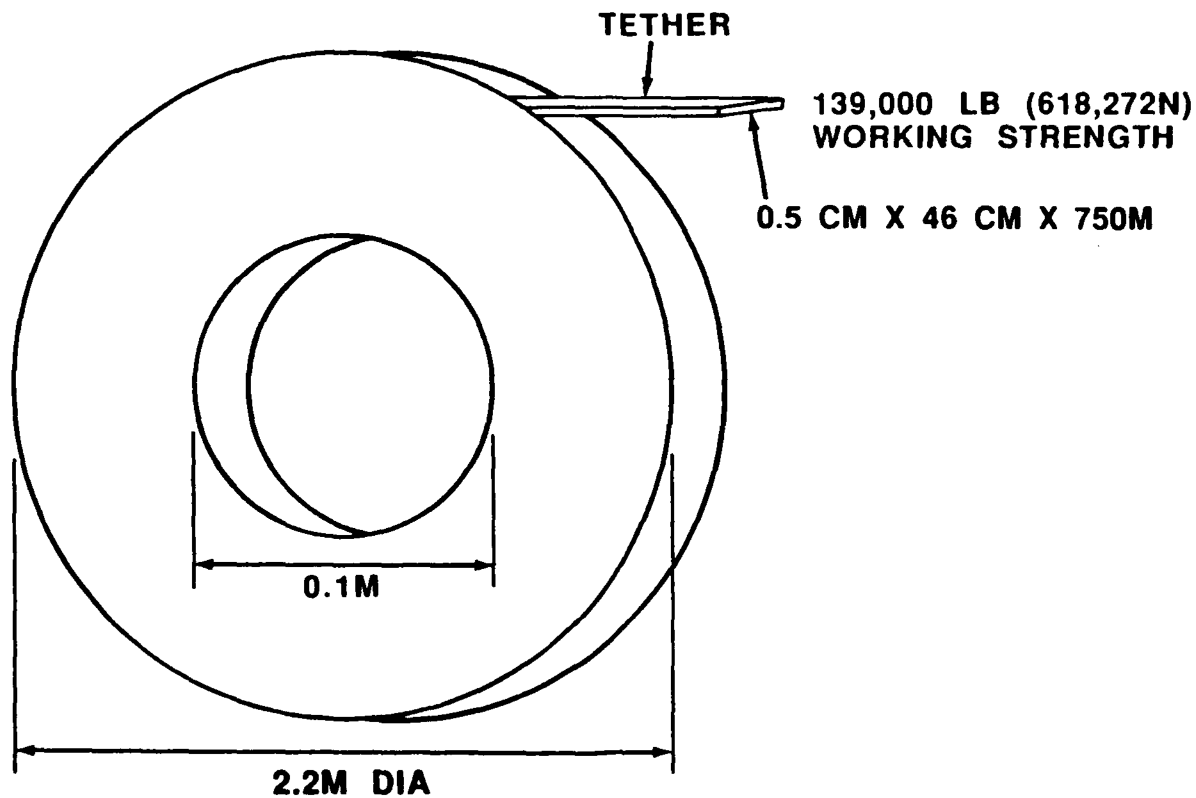
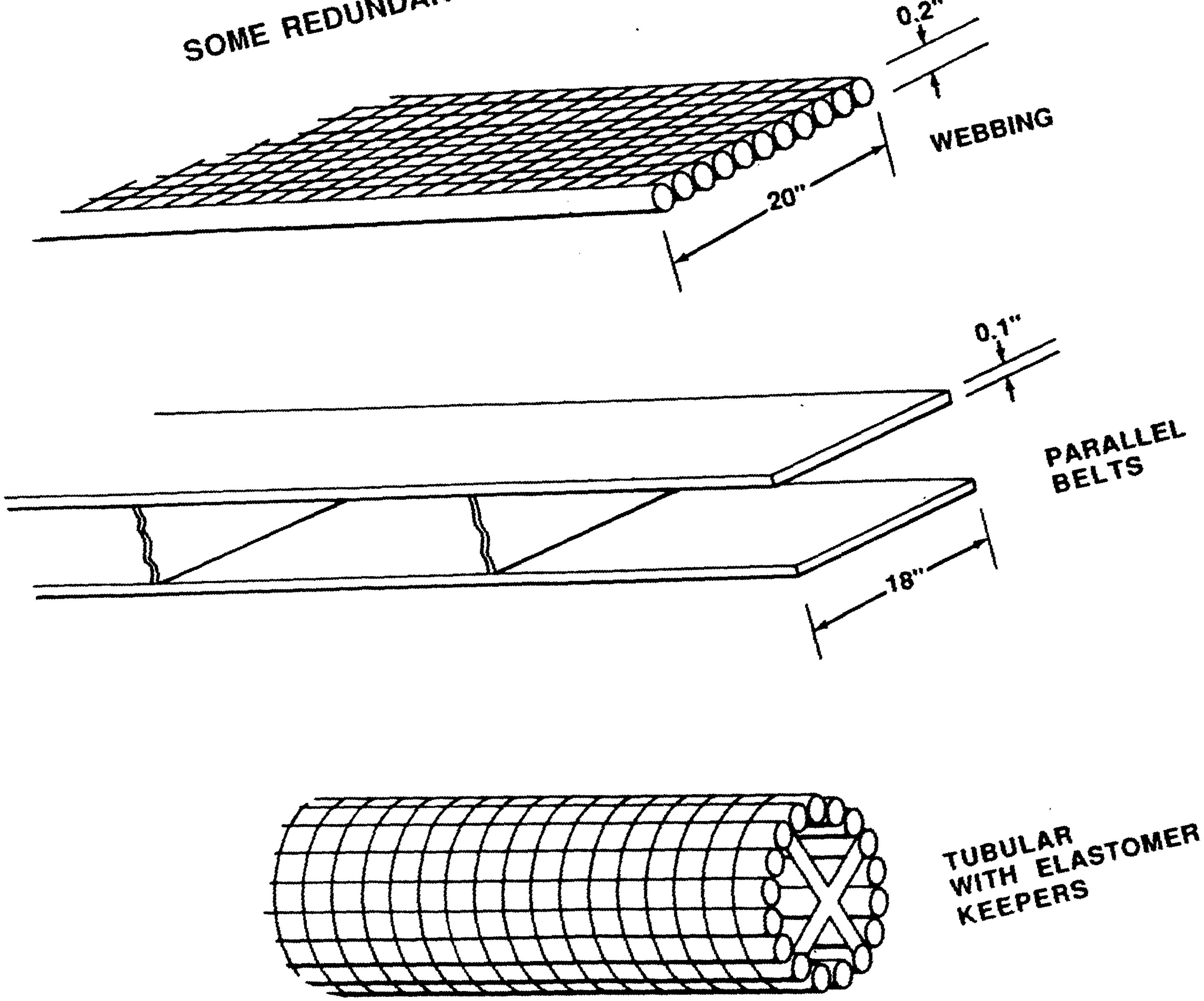
TETHER REEL

SAFETY FACTOR = 2 (DERATED STRENGTH/DYNAMIC LOAD)
= 7 (BREAK STRENGTH/DYNAMIC LOAD)



TETHER MASS 3,770 LB (1714 KG)

SOME REDUNDANT TETHER IDEAS



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METEOROID ENVIRONMENT

PROBABILITY OF TETHER BEING HIT BY A METEOROID WITH -

MASS >	DIAMETER >	IMPACT PROBABILITY
1 GM	1.6 CM	.000061
.1 GM	.72 CM	.001
.01 GM	.34 CM	.016
.001 GM	.16 CM	.23

ASSUMPTIONS: 420 DAY MISSION (EXPOSURE TIME)

690M^2 AREA (STRAP .46M X 750M X 2 SIDES)

METEOROID DENSITY .5GM/CM^3

MARS MISSION PILOTED VEHICLE PARAMETRIC MASS MODEL

- Round-trip interplanetary mission costs driven strongly by round-trip mass

- Primary effects on mission mass of including artificial gravity occur on piloted vehicle

=> Fractional mass change on piloted vehicle due to artificial gravity is good first-order estimate of overall mission cost change due to artificial gravity

MARS MISSION PILOTED VEHICLE PARAMETRIC MASS MODEL

$$M_{ag} = M_{0g} + \Delta \qquad (1)$$

$$M_{ag} = M_{0g} + \sigma M_{ag} + \varepsilon M_{0g} \qquad (2)$$

$$M_{ag} = \left[\frac{1 + \varepsilon}{1 - \sigma}\right] M_{0g} \qquad (3)$$

$$\Delta \approx \left[\sigma + \varepsilon\right] M_{0g} \qquad (4)$$

MARS MISSION PILOTED VEHICLE PARAMETRIC MASS MODEL

$$\Delta \approx 2.679\left[\frac{N\,F}{\omega\, I_{sp}}\right] + 8.341*10^{-4}\left[\frac{F}{\omega}\right]^2 + .075$$

N = number of spin-up/spin-down cycles
F = maximum operating artificial gravity level (fraction of 1-g)
ω = spin rate, radians/sec
I_{sp} = specific impulse of thruster propellant (LH2/LOX)

EXAMPLES:

A) $N = 4$, $F = 1$, $\omega = .2094$, $Isp = 460 \Rightarrow [\,\sigma+\varepsilon\,] = .20$ (MSFC study)

B) $N = 2$, $F = .5$, $\omega = .26175$, $Isp = 460 \Rightarrow [\,\sigma+\varepsilon\,] = .10$

POWER

VALGORA

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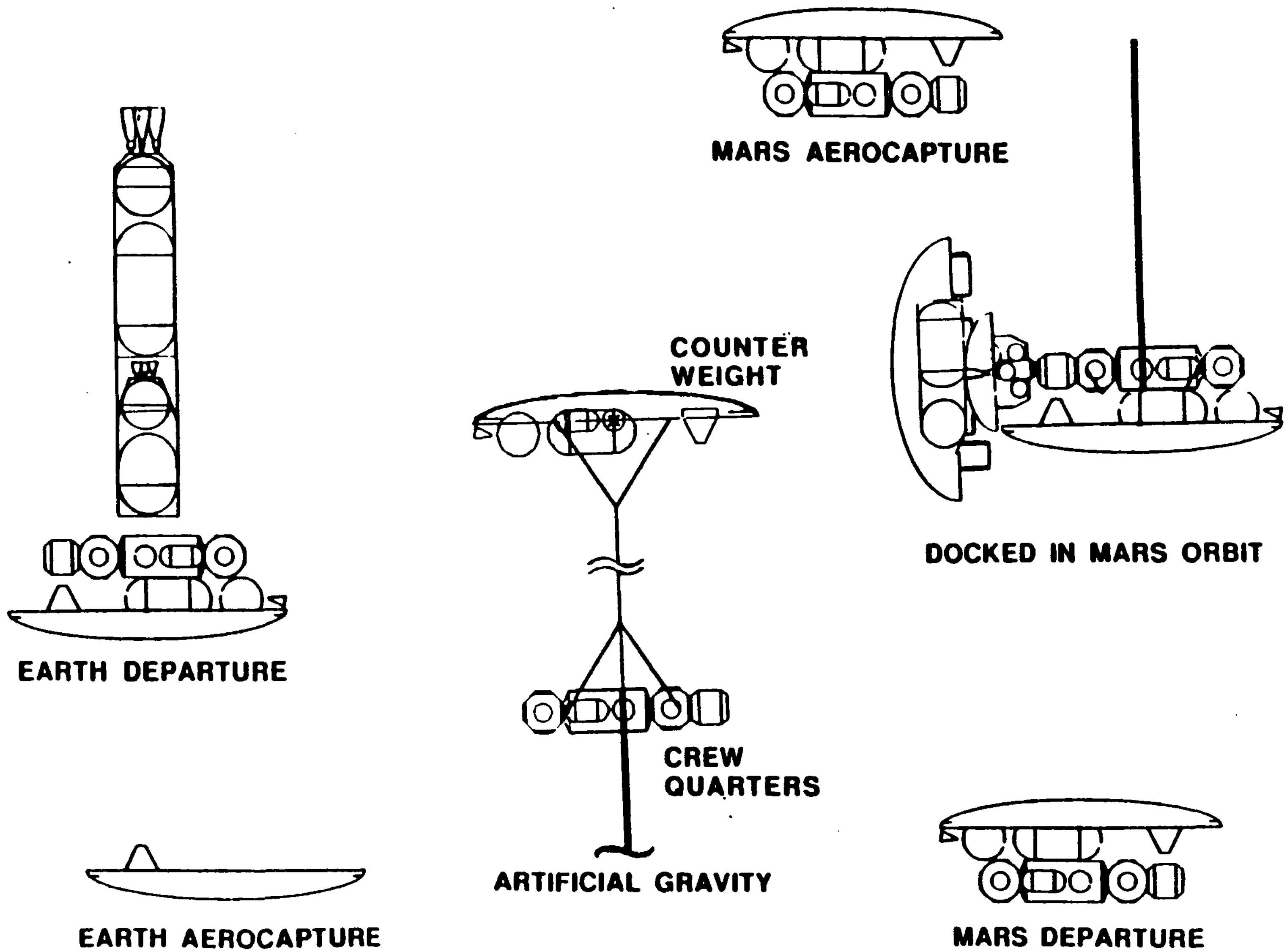
SPACE POWER CANDIDATES

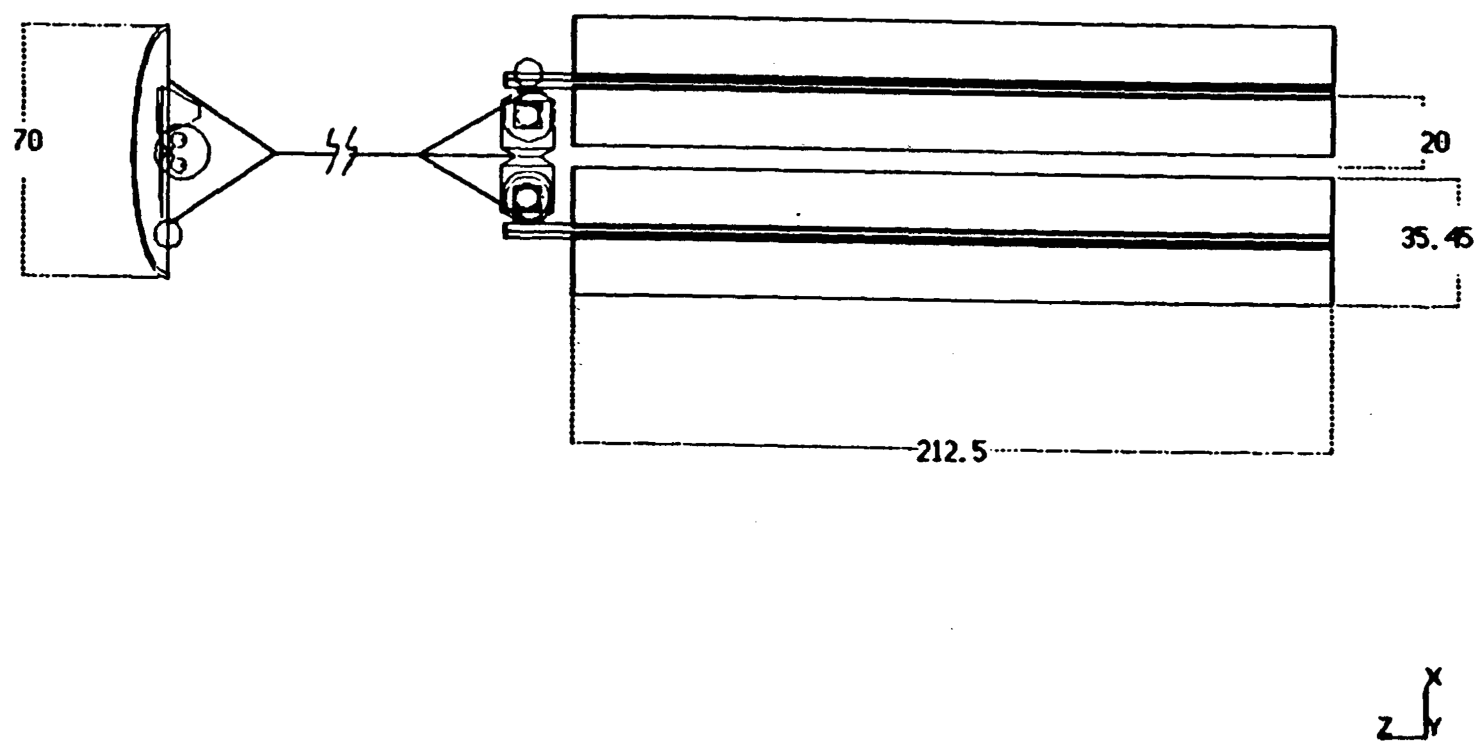
- PHOTOVOLTAIC/ENERGY STORAGE
- SOLAR DYNAMIC
- ISOTOPE DYNAMIC
- FUEL CELLS
- REACTORS

GAH MSFC PD23

MANNED MARS MISSIONS - ARTIFICIAL GRAVITY ACCOMMODATION
FLIGHT CONFIGURATIONS



MANNED MARS ARTIFICIAL G VEHICLE - DEPLOYED



POWER SYSTEM ELEMENTS

NEAR TERM TECHNOLOGY (SPACE STATION ERA)

- IMPROVED S.S. DEPLOYABLE/RETRACTABLE SOLAR ARRAY*
- AgZn BATTERIES (SHADE/RETRACTED PERIODS)*
- NiH2 BATTERIES (SHADE PERIODS)
- REGENERATIVE FUEL CELLS (4 HR. RETRACTED PERIOD ONLY)
- CARBON/EPOXY FLYWHEELS (POWER PULSES - 4/ MIN.)*

ADVANCED TECHNOLOGY

- LIGHTWEIGHT DEPLOYABLE/RETRACTABLE SOLAR ARRAY AMORPHOUS SILICON OR THIN FILM GaAs*
- VERY LIGHT WEIGHT ROLLOUT SOLAR ARRAY ARTIFICIAL GRAVITY DEPLOYED*
- HIGH ENERGY DENSITY LITHIUM BATTERIES*
- ADVANCED FLYWHEELS*

* YIELD MINIMUM SYSTEM MASS

MARS MISSION SCENARIO FOR POWER SYSTEM

PERIOD	REMARKS	TIME INTERVAL	POWER LEVEL	
BUILD-UP	ATTACHED TO S/S	7 MONTHS	UNKNOWN	
LEO C/O DEPART.	54/36 MIN-SUN/SHADE	TBD	25 KW	
SPIN UP AND DEPLOYMENT		3 HRS.	18 KW	
MARS TRANSFER	SOLAR DISTANCE .7 TO 1.38 A.U.	224 DAYS	25 KW	
RETRACTION/DESPIN		3 HRS.	18 KW	ENERGY STORAGE DRIVER
MARS BRAKING/ ORBIT CIRCULARIZATION		1 HR.	18 KW	
CARGO RENDEZVOUS		20 HRS.	18 KW	
MARS ORBIT	1000 KM ORBIT 82/42 MIN-SUN/SHADE	30 DAYS	18/15 KW	SOLAR ARRAY DRIVER
MARS C/O & DEPARTURE		1-2 DAYS	18 KW	
SPIN UP AND DEPLOYMENT		3 HRS.	18 KW	
EARTH TRANSFER	SOLAR DISTANCE 1 TO 1.4 A.U.	166 DAYS	25 KW	
RETRACTION & DESPIN		3 HRS.	18 KW	

EXTERNAL POWER SYSTEM
MASS SUMMARY

CASE	SPACE STATION ERA TECHNOLOGY	ADVANCED TECHNOLOGY
"0" G (FOR COMPARISON) SUN POINTED	1564 Kg (3441 Lb)	466Kg (1025 Lb)
BASELINE("0" & "1" G) SUN POINTED	2032 Kg (4470 Lb)	668Kg (1470 Lb)
BASELINE ("0" & "1" G) SPIN AXIS ⊥ TO SUN	3848 Kg (8466 LB)	798 Kg (1756 Lb)

CONCLUSIONS

- PHOTOVOLTAIC/ENERGY STORAGE CAN SATISFY THIS APPLICATION
- SPIN AXIS AT RIGHT ANGLE TO SUN DOUBLES MASS
- NEAR-TERM TECHNOLOGY (SPACE STATION ERA) IS HEAVY
- ADVANCED TECHNOLOGY COULD REDUCE MASS ABOUT 70-80 %

THERMAL/COMMUNICATIONS/DATA MGMT.

LIVINGSTON

THERMAL CONTROL

- TWO-PHASE RADIATOR SYSTEM UNSUITABLE FOR GRAVITY ENVIRONMENT
- POTENTIAL STRUCTURAL PROBLEMS WITH DOUBLE-SIDED RADIATORS
- USE SINGLE-PHASE BODY-MOUNTED RADIATORS
  - CAN REJECT ABOUT 15KW PER SPACE STATION MODULE (OR EQUIVALENT) IN LEO
  - ADEQUATE CAPACITY FOR PRESENTLY ANTICIPATED REQUIREMENTS
- NEED BETTER DEFINITION OF REQUIREMENTS
  - POWER LEVEL
  - POWER SPLIT BETWEEN TWO ENDS OF CONFIGURATION

DATA MANAGEMENT

- SAME HARDWARE AS SPACE STATION SYSTEM

COMMUNICATIONS AND TRACKING

ASSUMPTIONS

- TWO OPTIONS CONSIDERED
  - FULL-MOTION (30 FRAMES/SEC) COLOR VIDEO = 44 MBPS
  - COLOR VIDEO AT 3 FRAMES/SEC = 4.4 MBPS
- 1.3 AU MAXIMUM DISTANCE
- 12 WATT POWER AMPLIFIER
- UPGRADE 70M GROUND ANTENNAS FOR K_A BAND PER JPL DESIGN
- COST INCLUDES ONE ANTENNA
- COMMUNICATION WITH CARGO VEHICLE NOT INCLUDED
  - NOT A DISCRIMINATOR
- CARGO VEHICLE COMMUNICATIONS AND TRACKING SYSTEM NOT INCLUDED
- VEHICLE PROVIDES STABLE DESPUN PLATFORM

COMMUNICATIONS AND TRACKING

ALTERNATIVES TO HIGH-GAIN ANTENNA SYSTEM

- o POINTING DIFFICULTIES MAKE LASERS UNATTRACTIVE
- o ELECTRONICALLY-STEERED PHASED ARRAYS PROHIBITIVE
  - COMPLEXITY
  - POWER

COMMUNICATIONS AND TRACKING

CONCLUSIONS

- FULL-MOTION VIDEO REQUIRES 7.1M ANTENNA
  - ESTIMATED COST $203M (87$)
- LIMITED MOTION (3 FRAME/SEC) REQUIRES 2.25M ANTENNA a $175M
- FULL-MOTION VIDEO MAY BE REQUIRED (PUBLIC AFFAIRS)
  - LOWER TRANSMISSION RATE ACCEPTABLE IF REAL-TIME TV NOT REQUIRED
- POTENTIAL MAJOR CONFIGURATION DRIVER
  - ANTENNA SHOULD BE LOCATED AT MODULE END OF CONFIGURATION
  - DESPUN PLATFORM REQUIRED; NOT INCLUDED IN COST ESTIMATES
  - PLATFORM COULD BE SHARED BY SCIENTIFIC INSTRUMENTS
  - VIEW ANGLES, MUTUAL BLOCKAGE ARE ISSUES
- ANTENNA DESIGNED FOR 1-G LOADING

SPACE STATION IMPLICATIONS

PRITCHARD

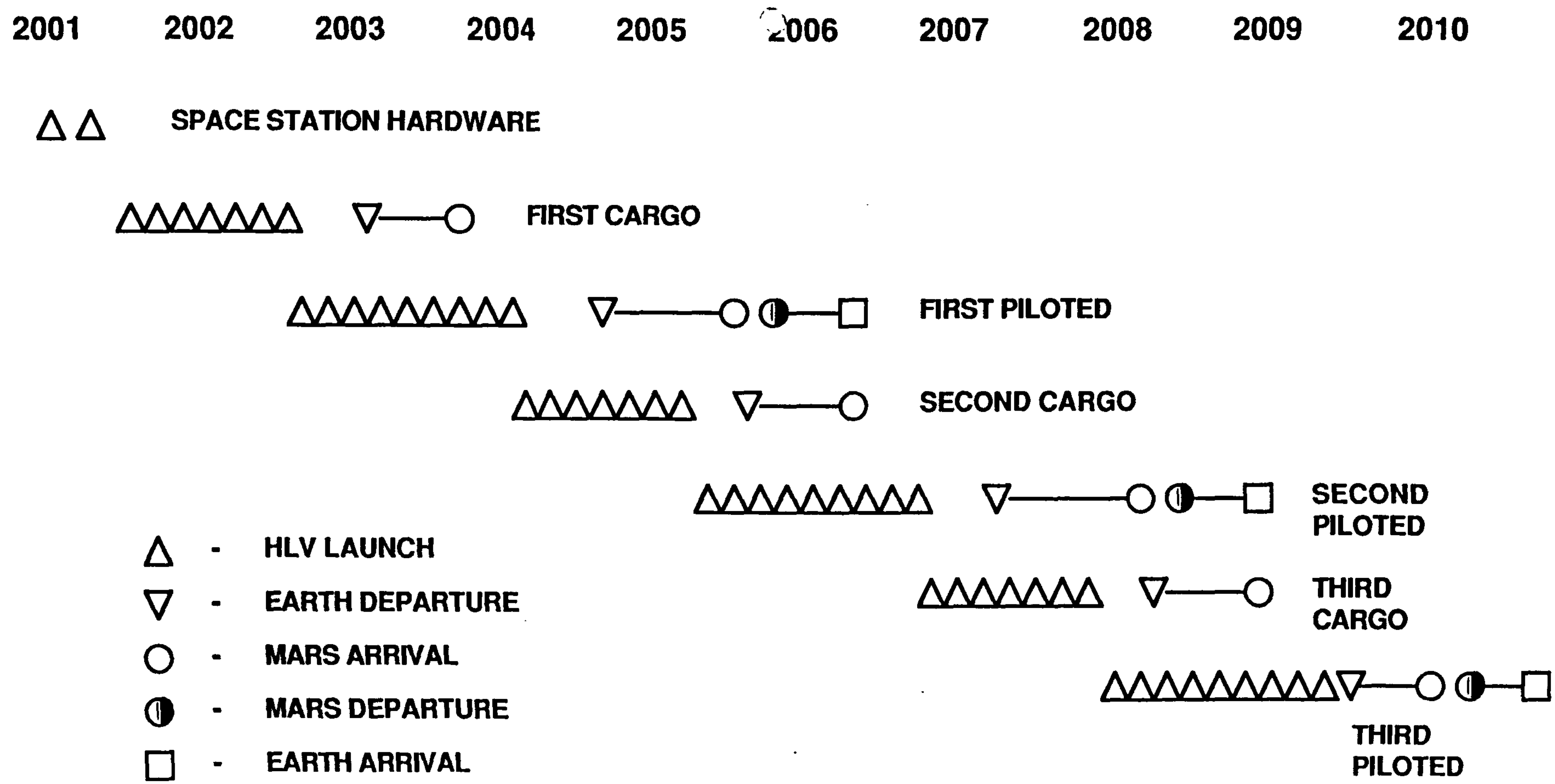
TYPICAL MANNED MARS MISSION SPACECRAFT MASS IN LOW EARTH ORBIT

AEROBRAKING AT MARS AND EARTH

MISSION CLASS	MARS STAY TIME	TRIP TIME	MASS
CONJUNCTION (MINIMUM ENERGY)	ONE YEAR PLUS	3 YEARS	10^6 LB
OPPOSITION	30 TO 60 DAYS	2 YEARS	1.5 - 2 X 10^6 LB
SPRINT	30 TO 60 DAYS	1 YEAR	5 X 10^6 LB
SPLIT SPRINT (CARGO VEHICLE ON NEAR MINIMUM ENERGY TRAJECTORY 1 YEAR AHEAD OF CREW VEHICLE)	30 TO 60 DAYS	1 YEAR (CREW)	1.6 X 10^6 LB (CREW) 1.3 X 10^6 LB (CARGO)

SPACE STATION ACCOMMODATION OF MARS SPRINT MISSION

MISSION TIMELINE



MARS MISSION MANIFESTING TO SPACE STATION

LAUNCH	CARGO	MASS (LB)
HLV-1		
	MOI AEROBRAKE	77276
	FLIGHT SYSTEMS (WET)	21470
	INTERSTAGE	10500
	REUSABLE 1ST STAGE (DRY WET)	47500
	1ST STAGE AEROBRAKE	8820
		165566
HLV-2		
	MARS LANDER - DESCENT STAGE	66000
	MARS LANDER - ASCENT STAGE	66000
	MARS STORAGE TANKER	38200
		170200
HLV-3		
	PROPELLANT CARRIER	40000
	PROPELLANT	160000
		200000

MARS MISSION MANIFESTING TO SPACE STATION

(CONTINUED)

LAUNCH	CARGO	MASS (LB)
HLV-4		
	PROPELLANT CARRIER	40000
	PROPELLANT	160000
		200000
HLV-5		
	PROPELLANT CARRIER	40000
	PROPELLANT	160000
		200000
HLV-6		
	PROPELLANT CARRIER	40000
	PROPELLANT	160000
		200000
HLV-7		
	PROPELLANT CARRIER	40000
	PROPELLANT	60560
	ROBOTIC SCIENCE PAYLOAD	33000
	MOI AEROBRAKE	26848
	1ST STAGE REFURB	13570
		173978

MARS MISSION MANIFESTING TO SPACE STATION

(CONTINUED)

LAUNCH	CARGO	MASS (LB)
HLV-8		
	EARTH RETURN CAPSULE	14883
	3 RD STAGE	32174
	AVIONICS	393
	RCS SUBSYSTEM	15253
	MODULE STRUCTURE (2)	25864
	NODE STRUCTURE (4)	24157
	AIRLOCK	6000
	THERMAL	2554
	AVIONICS	11966
	CREW SYSTEMS	9449
	ECLSS	14768
	CONTINGENCY (15%)	14065
	SPARES	1438
		172964
HLV-9		
	FLUIDS, THERMAL	164
	FLUIDS, ELECTRICAL	64
	ECLSS CONSUMABLE	6394
	TETHER	3770
	TETHER DEPLOYER	5225
	CREW CONSUMABLES	11267

MARS MISSION MANIFESTING TO SPACE STATION

(CONTINUED)

LAUNCH	CARGO	MASS (LB)
HLV-9 CONT.	RCS SUBSYSTEM	17714
	MISSION SCIENCE	5168
	1ST STAGE AEROBRAKE	8820
	1ST/2ND STAGE AERO-BRAKE	15600
	2ND STAGE AEROBRAKE	5240
	REUSEABLE 2ND STAGE (DRY WT)	28200
	PROPELLANT & CARRIER	53460
		161086
HLV-10	PROPELLANT CARRIER	40000
	PROPELLANT	160000
		200000
HLV-11	PROPELLANT CARRIER	40000
	PROPELLANT	160000
		200000

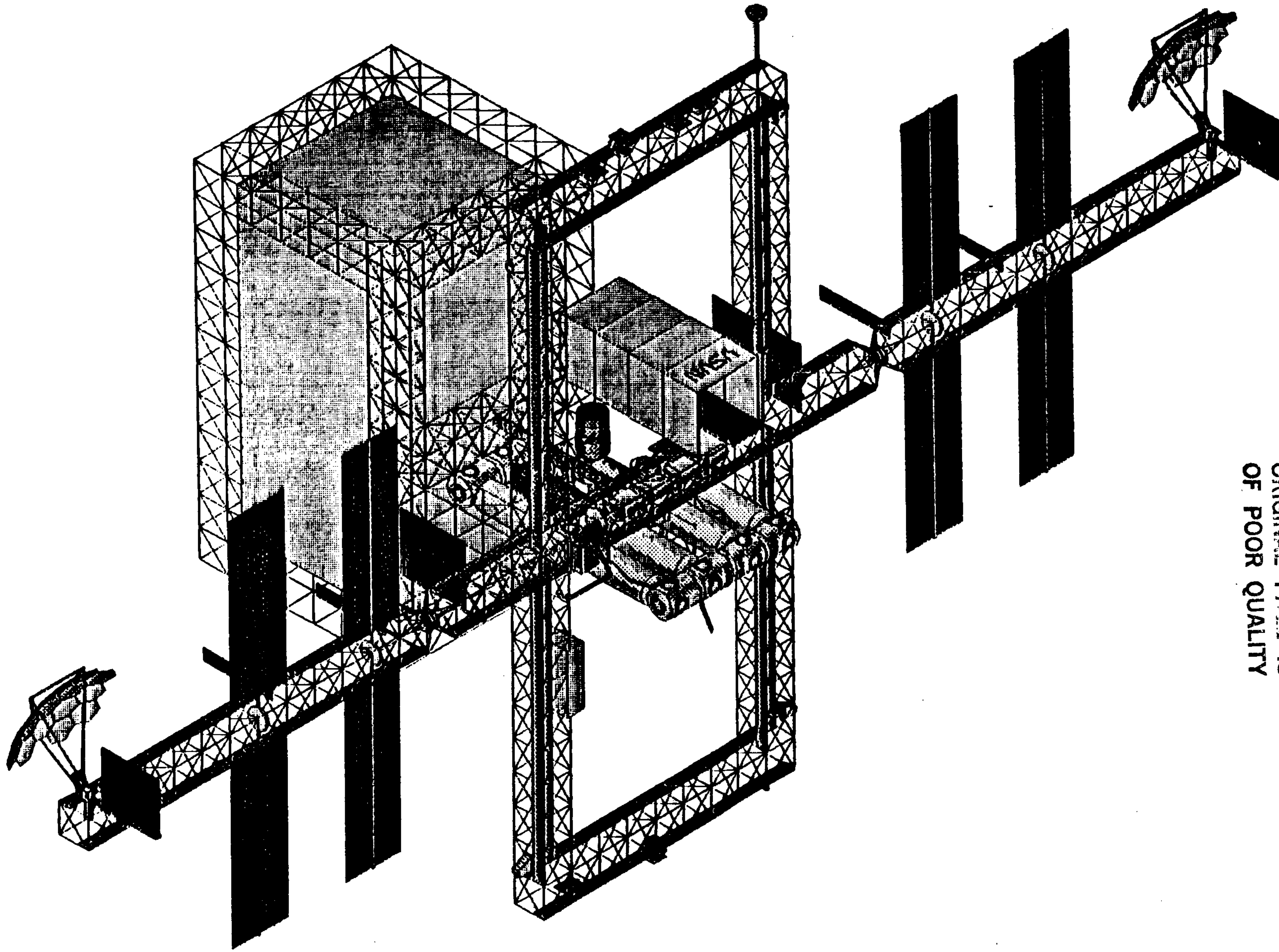
MARS MISSION MANIFESTING TO SPACE STATION

LAUNCH	CARGO	MASS (LB)
HLV-12	PROPELLANT CARRIER	40000
	PROPELLANT	160000
		200000
HLV-13	PROPELLANT CARRIER	40000
	PROPELLANT	160000
		200000
HLV-14	PROPELLANT CARRIER	40000
	PROPELLANT	160000
		200000
HLV-15	PROPELLANT CARRIER	40000
	PROPELLANT	160000
		200000
HLV-16	PROPELLANT CARRIER	40000
	PROPELLANT	118670
	1ST STAGE REFURB	13570
	2ND STAGE REFURB	8060
		180300

SPACE STATION ACCOMMODATION OF MARS SPRINT MISSION OPTIONS

1.	STATION BASED	- ALL MARS VEHICLE ACCOMMODATIONS, INCLUDING PROPELLANT, MAINTENANCE, AND STACKING FACILITIES ARE KEPT ON THE STATION.
2.	STATION BASED W/PTF	- VEHICLE PREP. AND MAINTENANCE FACILITIES ARE ON STATION, BUT PROPELLANT IS KEPT ON A CO-ORBITING PROPELLANT TANK FARM (PTF).
3.	TRANSPORTATION DEPOT (MAN TENDED)	- ALL VEHICLE AND OMV ACCOMMODATIONS (EXCLUDING HAB. MODULE AND CREW) ARE KEPT ON A CO-ORBITING PLATFORM.
4.	TRANSPORTATION DEPOT (PERMANENTLY MANNED)	- A COMPLETELY SEPARATE FACILITY IS BUILT TO ACCOMMODATE MARS VEHICLE, OMV'S, HAB. MODULE AND CREW.

MARS ACCOMMODATION STUDY



SPACE STATION TRANSPORTATION DEPOT MASS (LBS m) SUMMARY

AIRLOCK	4,430	RCS CLUSTERS	2,256
ALPHA JOINTS	2,640	RCS PROP. FARM	14,000
CMG'S	3,468	SD MODULES	30,972
CONTOL/ CHECKOUT EQ.	5,000	SERVICER, ELECTRO, GN&C	3,790
CUPOLA	3,200	TANK FARM SYS & ATTACH HARDWARE	45,400
DOCKING ADAPTERS	2,200	TDRSS & ANTENNA	2,160
ERECTOR SET	2,300	TELEROBOTIC SERVICER	5,238
NODES	20,000	THERMAL RADIATORS	8,080
HYPERBARIC AIRLOCK	6,137	TRUSS	22,100
IVA SERVICE ASSEMBLY LAB MODULE	69,350	UTILITY TRAYS	40,176
METEORITE/DEBRI: PROTECTION	45,700	TOTAL MASS	435,147
MSC/TRANSPORTER	10,800		
OMV & CREW CAB	18,900		
OMV PROP & TANICS	66,850		

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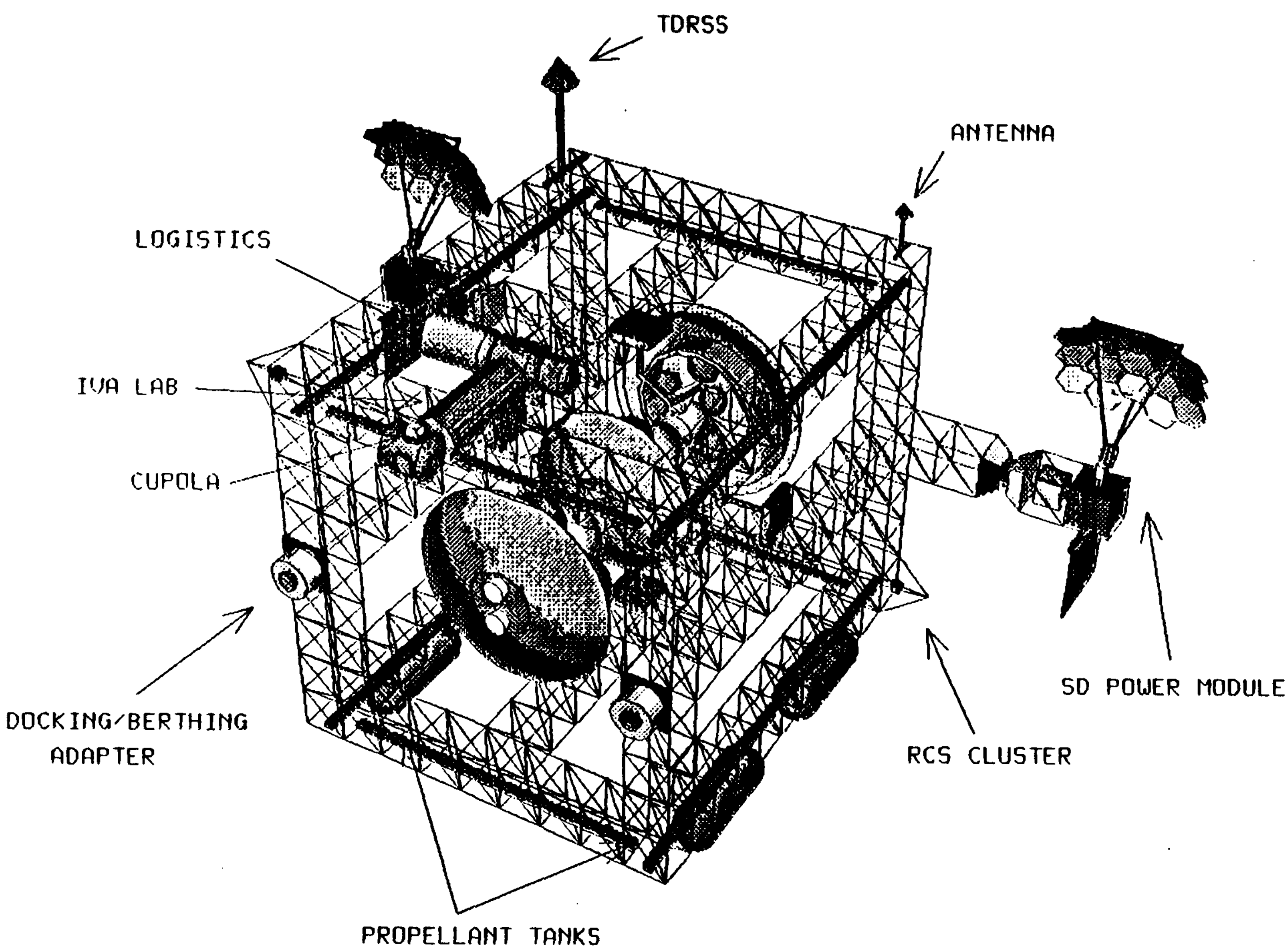
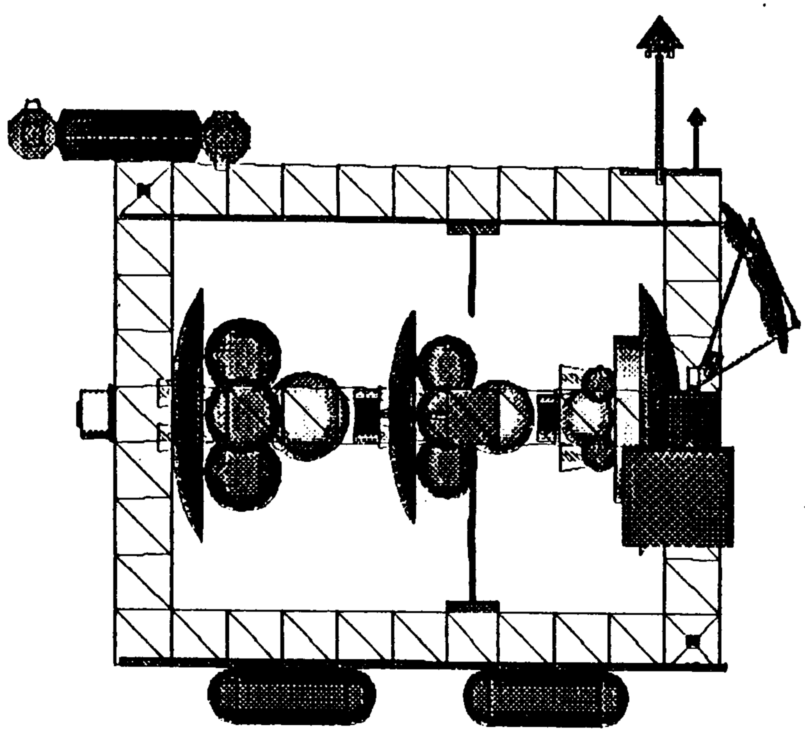
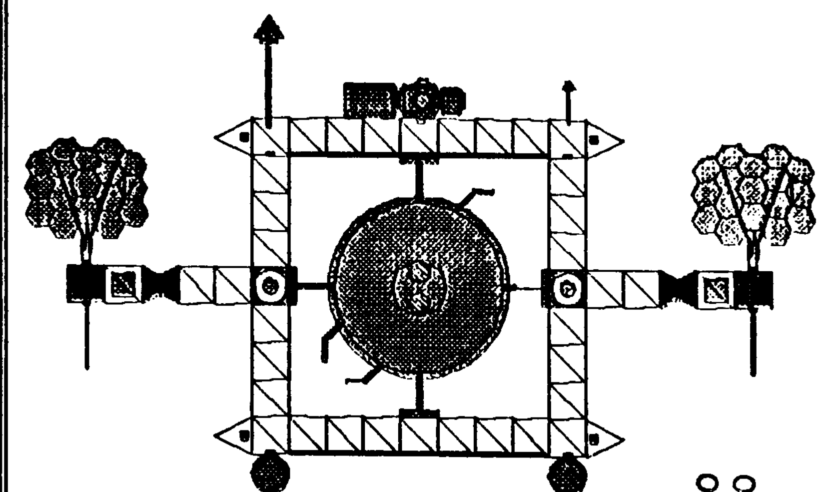
TRANSPORTATION DEPOT



MARS ACCOMMODATION STUDY

TRANSPORTATION DEPOT





MARS SPRINT MISSION ACCOMMODATION STUDY

MARS SPRINT MISSION AVERAGE CREW REQUIREMENTS

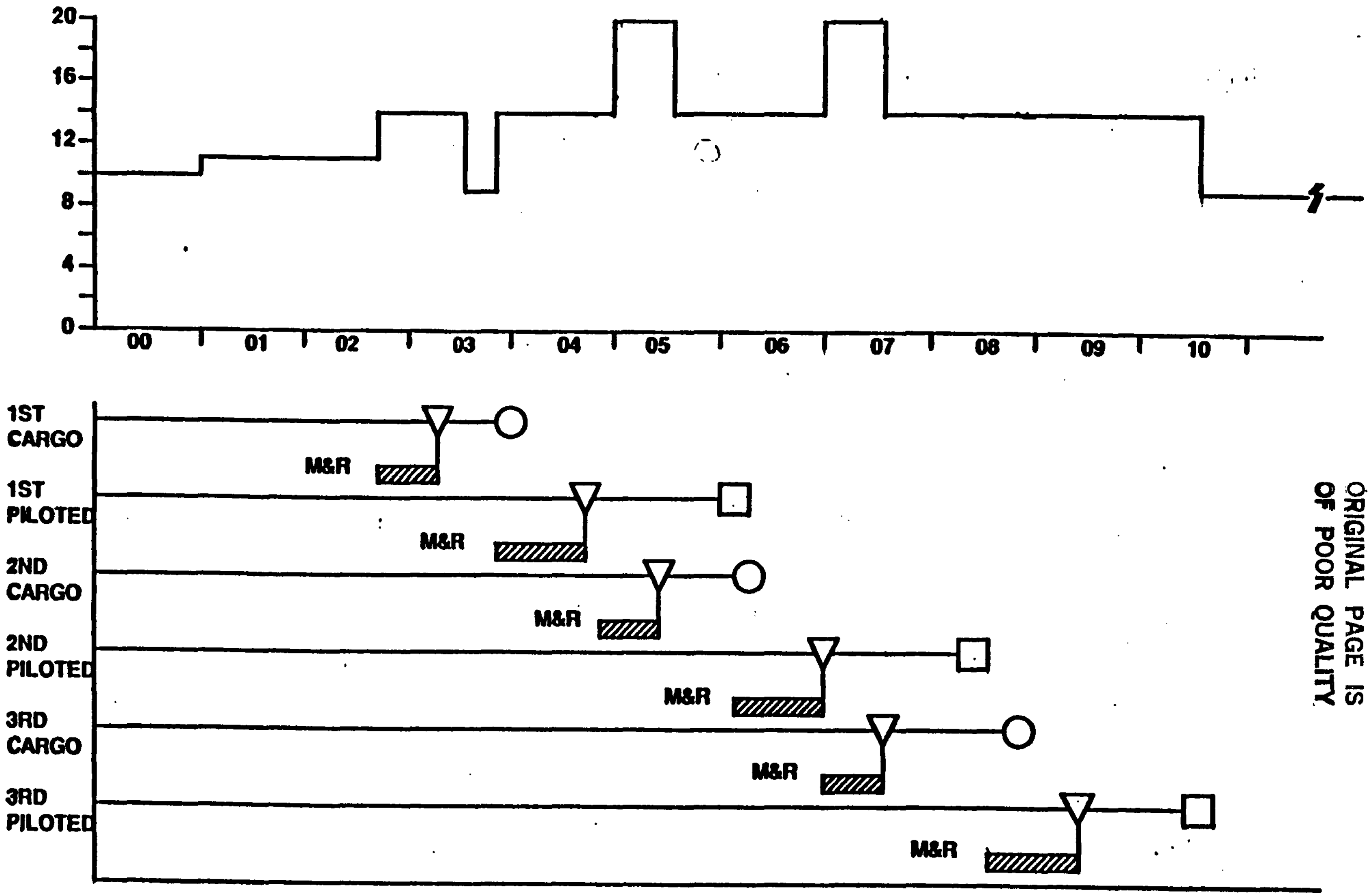
	'94	'95		'96		'97		'98		'99		'00		'01		'02	
BASELINE CREW [1]	8	8	8	8	8	8	8	8	8	8	8	8	8	8	8	8	8
LIFE SCIENCE	2.9	2.9	2.9	2.9	1.5	1.5	1.5	1.5	1.5	1.5	2.6	2.6	2.6	1.5	1.5	1.5	1.5
ON-ORBIT TECH. DEVELOPMENT	–	–	1	1	1	1	1	1	0.7	0.7	0.7	0.7	0.7	–	–	–	–
VEHICLE ASSEMBLY / C/O	–	–	–	–	–	–	–	–	–	–	–	–	–	3	3	3	3
TOTALS	10.9	10.9	11.9	11.9	10.5	10.5	10.5	10.5	10.2	10.2	11.3	11.3	11.3	12.5	12.5	12.5	12.5
TOTAL CREW AFTER ADJUSTMENT FOR BASELINE LIFE SCIENCES [2]	9.4		10.4		9				8.7		9.8			11 →			

1 - RESERVED FOR COEXISTING SCIENCE, COMMERCIAL AND TECHNOLOGY ACTIVITIES

2 - 1.5 CREW FOR LIFE SCIENCE MISSIONS EMBEDDED IN BASELINE CREW OF 8 (SAAX 307, 307A, 311).

MARS SPRINT MISSION ACCOMMODATION STUDY

QUARTERLY ON-ORBIT CREW REQUIREMENTS



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SPACE STATION ACCOMMODATION OF MARS SPRINT MISSION

CONCLUSIONS

- CURRENT BLOCK 2 CONFIGURATION CAN ACCOMMODATE THE MARS MISSION
- BRANCHING TO A TRANSPORTATION DEPOT IS THE "PROBABLE" ACCOMMODATION MODE - FURTHER IN-DEPTH STUDY REQUIRED
- HEAVY LIFT LAUNCH VEHICLE WITH 200,000 LB PAYLOAD TO SPACE STATION ORBIT IS REQUIRED
- FUNCTIONS REQUIRED OF SPACE STATION TO SUPPORT THE MANNED MARS MISSION ARE:
  - LIFE SCIENCE RESEARCH
  - TECHNOLOGY DEVELOPMENT AND QUALIFICATION
  - ASSEMBLY, CHECKOUT AND LAUNCH

COST

RUTHERFORD

MANNED MARS SPACECRAFT
COST GROUNDRULES

- DELTA COST BETWEEN ARTIFICAL-G AND ZERO-G.
- FY87 DOLLARS
- MISSION MODULE CONFIGURATION CHANGES FROM ZERO-G TO ARTIFICIAL-G OPTION CONSIST OF LARGER ELECTRICAL POWER REQUIREMENTS.
- COST INCLUDES DDT&E, FLIGHT UNITS, LAUNCH OPERATIONS, FACILITIES, AND WRAPAROUNDS (FEE/PROGRAM SUPPORT/CONTINGENCY).
- COST DOES NOT INCLUDE MISSION OPERATIONS.
- COUNTER WEIGHT FOR LEO VARIABLE G FACILITY (TANK WITH SS TRASH) ASSUMED TO BE AVAILABLE AT NO COST.

COST COMPARISON OF MANNED MARS MISSION OPTIONS

(BILLIONS OF 1987 DOLLARS)

	ZERO G	ARTIFICIAL G	DELTA
COUNTER WEIGHT SYSTEMS (MARS AEROBRAKE, EARTH RETURN CAPSULE & AEROBRAKE)	4.6	4.4	-.2
TETHER SYSTEM (TETHER, RCS, AVIONICS)	—	2.5	+2.5
MISSION MODULE (HABITAT, NODES, SOLAR ARRAY, AIRLOCK, MISSION SCIENCE)	10.7	11.1	+.4
PROPULSION STAGES (1ST, 2ND, 3RD & RETURN STAGES)	7.0	7.6	+.6
CARGO VEHICLE (1ST STAGE, PROP. STORAGE, AEROBRAKE, SCIENCE PROBES, MEM)	15.2	15.4	+.2
TRANSPORTATION (HLV)	2.1	2.7	+.6
TOTAL	39.6	43.7	+4.1

MANNED MARS MISSION
ARTIFICIAL G OPTIONS

(BILLIONS OF 1987 DOLLARS)

	ARTIFICIAL G BASELINE	ARTIFICIAL G W/MIN LEO FACIL	ARTIFICIAL G W/MAX LEO FACIL
COUNTER WEIGHT SYSTEMS	4.4	4.4	4.4
TETHER SYSTEM	2.5	1.5	1.5
MISSION MODULE	11.1	8.9	8.9
PROPULSION STAGES	7.6	7.6	7.6
CARGO VEHICLE	15.4	15.4	15.4
TRANSPORTATION (HLV)	2.7	2.7	2.7
SUBTOTAL	43.7	40.5	40.5
LEO VARIABLE G FACILITY		8.8	13.7
TOTAL		49.3	54.2

LEO VARIABLE G FACILITY COST

(BILLIONS OF 1987 DOLLARS)

	MINIMUM FACILITY	MAXIMUM FACILITY
MISSION MODULE	6.0	10.6
TETHER SYSTEM	2.5	2.7
TRANSPORTATION	.3	.4
TOTAL	8.8	13.7

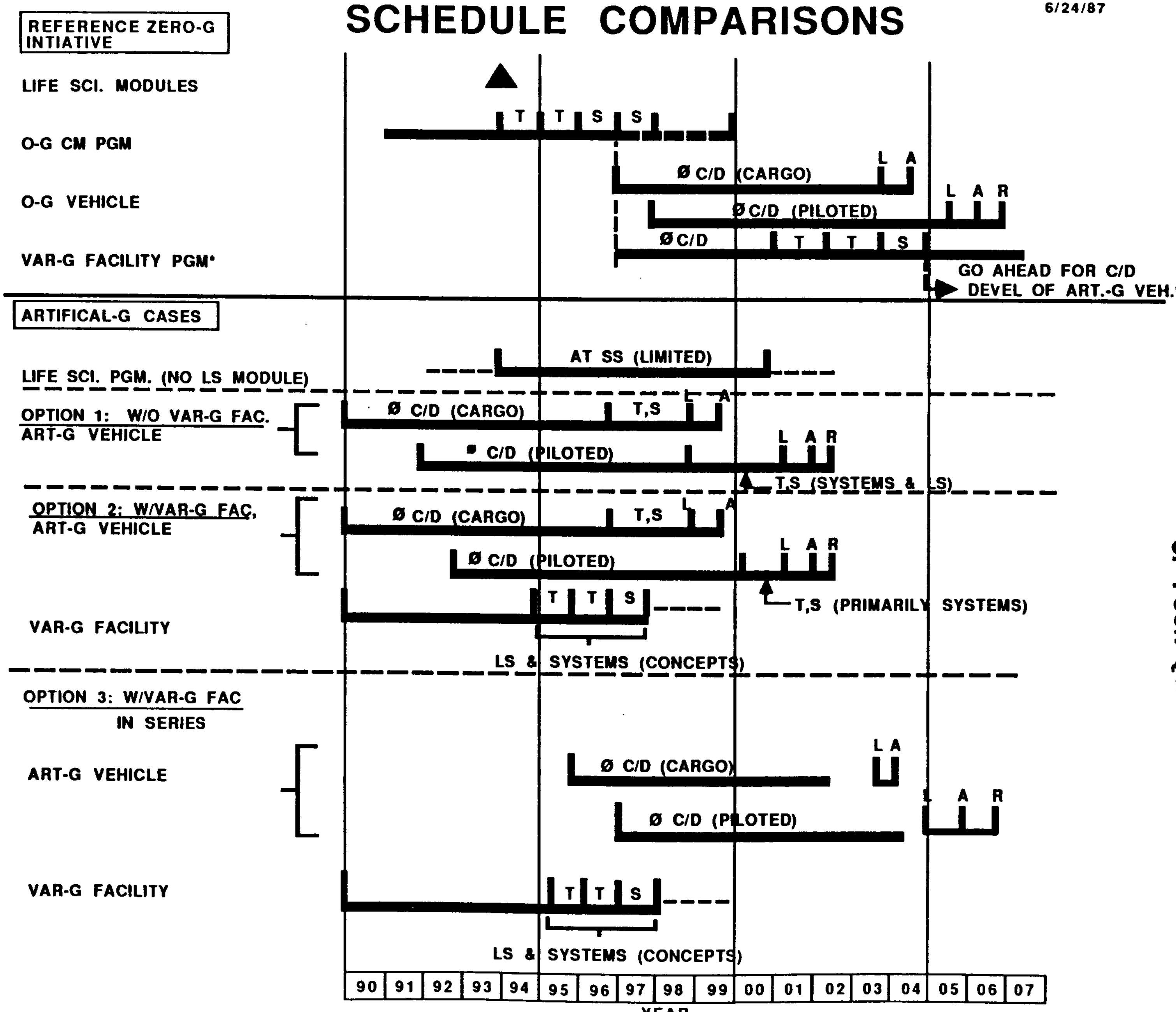
SCHEDULE/SUMMARY

BUTLER

1-3807-7-6D

SCHEDULE COMPARISONS

6/24/87



LS = LIFE SCIENCES * FALL-BACK MODE T = TEST S = SIMULATION L = LEAVE A = ARRIVE R = RETURN

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KEY ENABLING TECHNOLOGY/ADVANCED DEVELOPMENT AREAS FOR ARTIFICIAL-G MISSION

	FOR PRIMARY MODE	FOR BACKUP MODE
• LIFE SCIENCES		
– PHYSIOLOGICAL ATTRIBUTES OF ROTATIONAL g-FIELDS	√	
– ALLOWABLE ROTATION RATE	√	
– ALLOWABLE g-LEVEL	√	
– ALLOWABLE g-FIELD GRADIENT (HEAD-TO-FOOT)	√	
– ZERO-g COUNTER MEASURES		√ *
– ZERO-g MEDICAL TOOLS & TECHNIQUES		√ *
• ARTIFICIAL-g SYSTEMS	√	
– SYSTEMS IMPLICATIONS OF VARIABLES LISTED UNDER LIFE SCIENCES		
– TETHERS		
– SPINUP/SPINDOWN TECHNIQUES		
– SEVERED-TETHERED RECOVERY		
• MODULE SYSTEMS FOR OPERATION IN 0-g AND 1-g	√	√
– PRELIMINARY ASSESSMENT SHOWS AREAS NEEDING MODIFICATION ARE:		
• THERMAL CONTROL SYSTEM		
• SLEEP STATION		
• SHOWER & WASTE MGMT.		
• HUMAN FACTORS		
• HIGH-CYCLE-LIFE/"HIGH-FREQUENCY"(2 RPM) SYSTEMS	√	
– GIMBAL SYSTEMS FOR DESPUN PLATFORM/INSTRUMENTS		
– ENERGY STORAGE SYSTEM		

* TECHNOLOGY WORK PLANNED FOR CERTIFICATION OF >90-DAY STAY AT SPACE STATION MAY PROVIDE MOST OF THIS.

1-3947-7-9T

SUMMARY OBSERVATIONS

- AN ARTIFICIAL-G VEHICLE FOR MANNED MARS MISSIONS APPEARS TO BE FEASIBLE TECHNICALLY AND PROGRAMMATICALLY.

- THERE CAN BE A HIGH DEGREE OF COMMONALITY BETWEEN A ZERO-G AND AN ARTIFICIAL-G VEHICLE; THE DIFFERENCES ARE MOSTLY ADDITIONS OF EQUIPMENT.

- USING AN ARTIFICIAL-G VEHICLE INSTEAD OF A ZERO-G VEHICLE FOR THE PILOTED PORTION OF A SPLIT MISSION TO MARS:

  - ADDS ABOUT ~~430K~~ 478K LB. TO THE PILOTED VEHICLE WEIGHT AND ABOUT ~~205K~~ 230K LB. TO THE CARGO VEHICLE WEIGHT (~~25%~~ 26% TOTAL INCREASE).

  - ADDS ABOUT $4B (10% INCREASE) TO THE COST OF A PILOTED/CARGO SET, INCLUDING COSTS OF THE 5 ADDITIONAL EARTH-TO-ORBIT HLV LAUNCHES REQUIRED.

  - PROVIDES BENEFITS IN PHYSIOLOGICAL AND HUMAN FACTORS AREAS.

  - DOES NOT ELIMINATE REQUIREMENTS FOR ZERO-G COUNTERMEASURES RESEARCH (SINCE ZERO-G IS AN ABORT MODE), BUT SHIFTS THE PRIMARY EMPHASIS TO VERIFICATION OF ARTIFICIAL-G UNKNOWNS (POTENTIAL PHYSIOLOGICAL & ADAPTATION EFFECTS).

    - COULD POSSIBLY REDUCE SOME LIFE SCIENCE ACTIVITIES AT SPACE STATION
    - REQUIRES SOME TESTING & SIMULATION OF ARTIFICIAL-G MISSION, PROBABLY NECESSITATING A LEO VARIABLE-G FACILITY (COST ROUGHLY ESTIMATED AT 8.8B TO 13.7B).

  - DOES NOT IMPOSE SCHEDULE IMPACTS

Planetary Surface Systems Elements Catalog

Prepared by
L. A. Pieniazek

Lockheed Engineering & Sciences Company
October, 1988

INTRODUCTION

The attached catalog provides summary descriptions of the various elements used to define the Planetary Surface Systems for the FY 1988 Office o. ..xploration (OEXP) Case Studies. Entries are primarily point designs; however, parametrics and additional information are included as available. The element descriptions were extracted from studies performed by various groups, most notably the Special Assessment Agents and the Lunar Base Systems Study. The descriptions are intended as potential input to the Mission and Supporting Elements Data Base. The information provided reflects that requested in Appendix C, Data Base Input Requirements, of the OEXP *Study Requirements Document*, Version 2.0. The descriptions (including graphics) are currently being entered into a data base implemented with Double Helix II on a Macintosh Personal Computer.

The following sections provide the following:

Field Description — Description of the format of catalog entries.

References — List of source documentation from which information was extracted.

Catalog Summary — Summary list of elements and key parameters. The list is arranged alphabetically with respect to the major functional groupings and the short alphabetic element identifier. The Summary columns correspond to the following fields of an element entry (See below for field descriptions and content codes).

Column Id	Field Name	Description
ID-#	ID#	Record number
Class	Element Type	Element Type
Elt Id	Element Id	Element Identifier
Mass	Mass-kg	Mass in kg
Power	Avg Power kW	Average Power in kW
Vol	Volume-m3	Volume in m^3
Deploy	EVA Deploy mh	EVA time to Deploy in mh
Service	EVA Maintain	EVA time to Maintainin mh

Element Catalog — Element descriptions. Arranged by the numeric identifier code for the element. The code can be found in the first column of the Catalog Summary.

FIELD DESCRIPTION

Figure 1 depicts the standard format for an element entry. A field with no entry should be interpreted as undefined. In addition to the standard fields, an entry may contain other information extracted from the source study, e.g., tables and diagrams. The standard fields are described below.

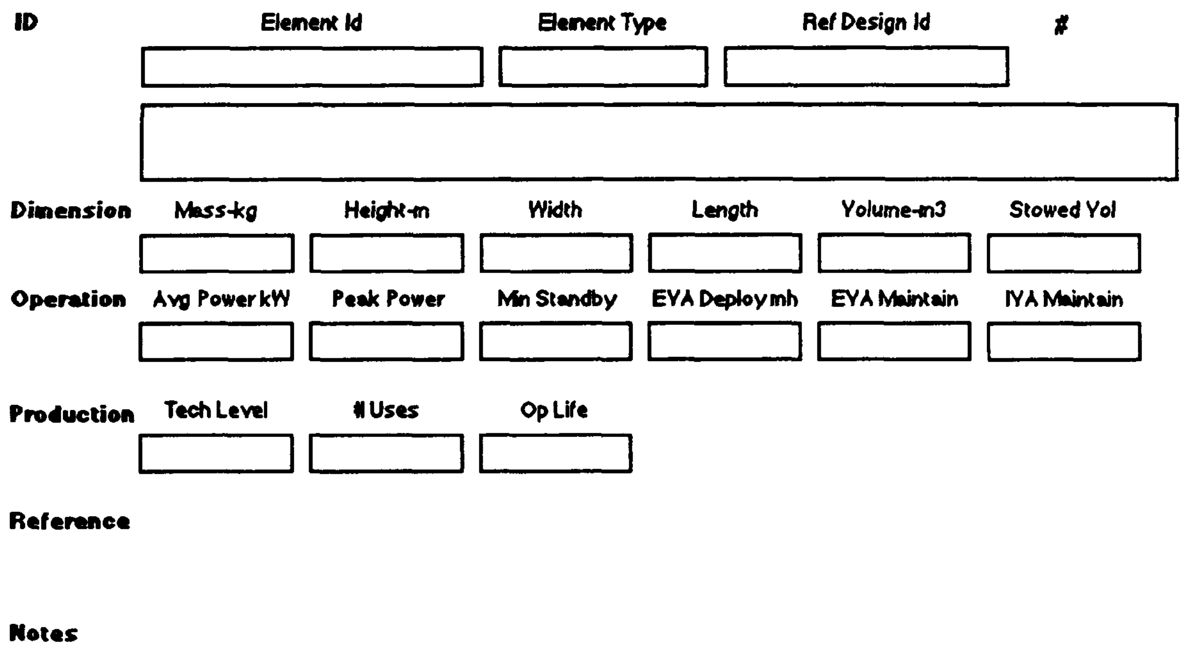

Figure 1. Sample Element Entry Format.

Form Name	Description
ID	
Element Id	*Brief Identification Code for referencing the element*
Element Type	General category or type: **veh** = vehicle, **env**=environmental system, **power** = power system, **EVA** = EVA system, **misc** = miscellaneous, **resource** = in situ resource utilization
Ref Design Id	Identification code of general design information on which element is based. Not yet implemented in the data base.
#	Record number of entry in data base.
	Name of element.
Dimension	
Mass-kg	Mass of single unit in kg.
Height-m	Height of unit in meters.
Width	Width of unit in meters.
Length	Length of unit in meters.
Volume-m3	Erected or deployed volume of a unit in cubic meters.
Stowed Vol	Packed or transported volume of a unit in cubic meters.
Operation	
Avg Power kW	Nominal or average power in kilowatts associated with a unit. Generally is the amount consumed, but may be the amount produced (power supplies), etc.
Peak Power	Maximum power in kilowatts associated with a unit.
Min Standby	Minimum standby power in kilowatts associated with a unit.
EVA Deploy mh	Estimate of the total number of EVA man hours needed to deploy a unit.
EVA Maintain	Estimate of the number of EVA man hours needed to perform a single Maintenance operation on a unit.
IVA Maintain	Estimate of the number of IVA (in vehicle, rover, or habitat) man hours needed to perform a single Maintenance operation on a unit.
Production	
Tech Level	Technology Readiness Level.
# Uses	Estimate of the number of uses of a unit. Expressed as order of magnitude: **s** = 1 to 10, **da** (deka) = 10's, **h** (hecto) = 100's, **k** (kilo) = 1000's.
Op Life	Estimate of the operational life of a unit.
Reference	
	Source or documentation for element.
Notes	
	Additional information.

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The following abbreviations are used:

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LBS-88-n	Preprint n from the Lunar Bases & Space Activities in the 21st Century Symposium, 5-7 April 1988.

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Simonsen, Lisa C., "Conceptual Design of a Thermal Control System for a Lunar Base Shack, Inflatable Habitat, and Manned Rover", in *Lunar Base Systems Study, A Status Report*, Presented at Joint JSC/LaRC Lunar Base Study Review, September 29, 1988.

Simonsen, Lisa C. and Mark Persons, "Conceptual Design of a Thermal Control System for the Lunar Shack, Inflatable Habitat, and Rover Vehicle", preprint, Summer, 1988.

Simonsen, Lisa C. et al, "Conceptual Design of a Lunar Base Thermal Control System", LB-88-225.

Catalog Summary

ID	Class	Elt Id	Mass kg	Power kW	Vol cu m	Deploy Man-hr	Service Man-hr
23	env	ECLS-Lmod	4803.0		27.00	24.0	
40	env	ECLSS - L inf6	16820.0	8.5	49.30		
41	env	ECLSS-Lshack 4/10	2800.0	4.2	14.54		
22	env	Hab-Inf 16m	18500.0		2145.00		
39	env	TCS-L inf	6122.0	40.0	21.80		
24	env	TCS-Lmod	4315.0	50.3	18.00	24.0	
19	EVA	EMU-LS	229.2		1.04		
20	EVA	EMU-MS	229.2		1.04		
18	EVA	EMU-Ph	135.9		1.04		
3	EVA	EVA Retriever	480.2		0.15		
4	EVA	EVA Tools	100.0				
1	EVA	MMU	153.1		1.27		
2	EVA	MMU-FSS	114.8		3.62		
5	EVA	PhD Aids	100.0		1.00		
21	EVA	RadPG	170.0	0.0	0.22		
25	misc	bagger	250.0				
26	misc	Pad Markers	10.0		0.03	2.0	
32	power	NR-1MW est	24000.0				
37	power	NR-825 kWe Stirling	20004.0	825.0			
38	power	PV/RFC-L50kW	19744.0	50.0			
35	power	PVA/RFC-L50kW (est)	15000.0	50.0			
33	power	PVC-aSi	500.0	50.0			
36	power	Pwr Cable	500.0				
34	power	RFC	17000.0	50.0			
28	resource	Pilot Plant-LLOX	22.4			256.0	
30	resource	Plant - PhProp	107900.0	1067.0			
27	resource	Plant-LLOX	45700.0			543.0	
29	resource	Plant-MProp	80000.0	744.0			
15	veh	Cord Cart	820.0		3.08		
12	veh	Crane	1900.0		273.00		
10	veh	digger	1900.0	5.0	94.50		
16	veh	Fuel Cell Cart	1290.0		7.27		
7	veh	pLRV	5000.0	7.0	144.00		
9	veh	pMRV	6500.0	14.0	144.00		
14	veh	PRV-L40	14000.0		168.00		
13	veh	T-ramp	2788.0		63.00		
17	veh	TeleAsst	1290.0		7.27		
11	veh	Truck-L2.5	1900.0		43.75		
6	veh	upLRV	550.0	2.0	15.00	2.0	
8	veh	upMRV	1000.0	4.3	15.00		

ID	Element Id	Element Type	Ref Design Id	# 1
	MMU	EVA		

Manned Maneuvering Unit

Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	153.1	1.22	0.85	1.22	1.27	

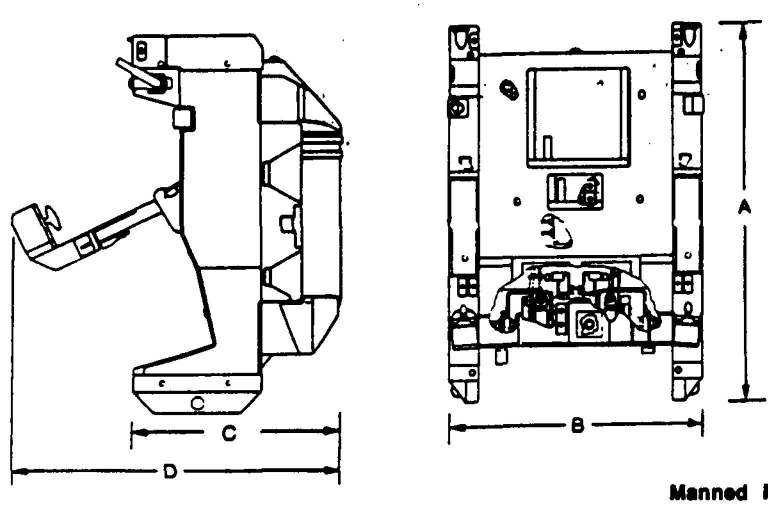
Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain

Production	Tech Level	# Uses	Op Life
	6	da	

Reference JSC-20466

Notes Based on STS MMU. Normally at least 2 MMUs are manifested, stowed on the MMU Flight Support Station (FSS).

Modular, self-supporting backpack with its own power, propulsion, and controls. Complete 6 DOF control and automatic attitude hold. Has attach points for accesories (tools, lights, cameras, and instrument sensors). Propellant is non-contaminating N2. Limited ability to maneuver free-flyers.



Manned Maneuvering Unit

Technical information	
Part Number	852MU000000
Weight	338 lb
Material	Aluminum
Control	Three modes of operation - manual, automatic attitude hold, and satellite stabilization Left hand controller - 3DOF translation Right hand controller - 3DOF rotation Acceleration - approximately 0.2 to 0.4 ft/sec^2 Redundant logic
Maximum Range	Early flights - approximately 300 ft Potential - approximately 3000 ft
Electrical Power	Two batteries: total power - 852 W-hr
Propellant	Gaseous nitrogen Reservicing in less than 10 min
Stowage	Forward cargo bay on FSS

Dimensional Data	
A	50.0 in.
B	33.3 in.
C	27.0 in.
D	48.0 in.

Performance Enhancements Required

ID	Element Id	Element Type	Ref Design Id	# 2
	MMU-FSS	EVA		

MMU Flight Support Station

Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	114.8	1.70	1.45	1.47	3.62	

Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain

Production	Tech Level	# Uses	Op Life
	7	da	

Reference *JSC-20466*

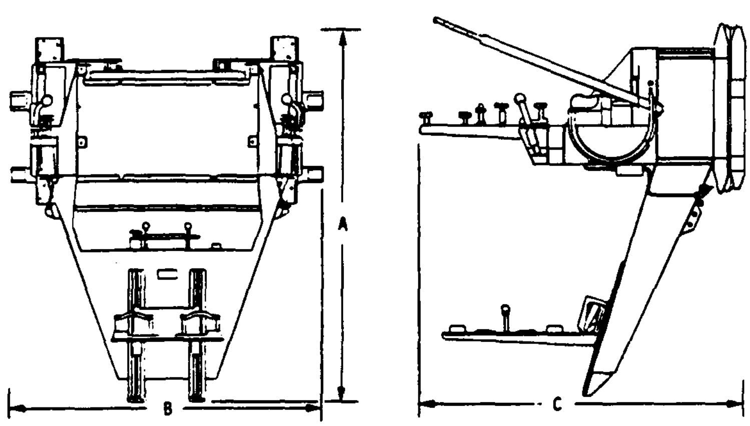
Notes Based on STS MMU FSS (253 lb, 67"x57"x57.9"). Normally at least 2 MMUs are manifested.

Servicing and mounting interface for MMU. Provides stowage, GN2 recharge, and power for 5 MMU heaters (thermal control via MLI and heaters on FSS propulsion components).

MMU FLIGHT SUPPORT STATION

Technical Information	
Part number	852MM000000V
Weight	253 lb
Material/con-struction	Structure - aircraft-style aluminum sheet metal Handrails and foot restraint platform - fiber-glass Exterior paint - white Interior surfaces - low emissivity coating Multilayer insulation - ten layers of 1/4 mil aluminized (both sides) mylar with Dacron net separators Heaters on all propulsion components Shock isolation system - attenuates FSS/MMU random vibrations
Foot platform	Adjustable over 14-in. range in 1-in. increments
Mounted	On port or starboard side of cargo bay near EVA airlock

Dimensional Data	
A	67.0 in.
B	57.0 in.
C	57.9 in.



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ID	Element Id	Element Type	Ref Design Id	# 3
	EVA Retriever	EVA		

EVA Retriever

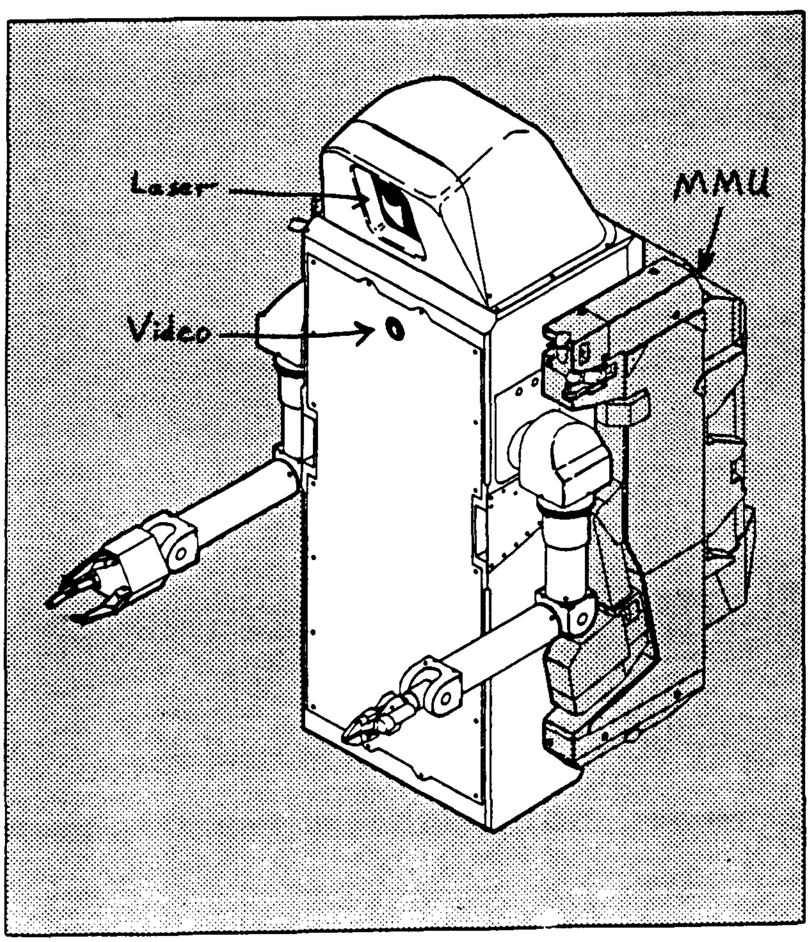
Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	480.2	1.78	1.02	0.08	0.15	

Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain

Production	Tech Level	# Uses	Op Life
	4-5	da	

Reference Reuter telecon

Notes Robot for on-orbit EVA support. Mass/volume based on current EVA Retriever under development.



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ID	Element Id	Element Type	Ref Design Id	# 4
	EVA Tools	EVA		

EVA Tools for Phobos

Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	100.0					

Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain

Production	Tech Level	# Uses	Op Life
	1-7	da?	

Reference JSC-20466, Christiansen pp52ff
Bufkin[88]

Notes Estimate of mass that appears typical of STS flights.
Requires much more study.

ID	Element Id	Element Type	Ref Design Id	# 5
	PhD Aids	EVA		

EVA Aids and Restraints for Phobos

Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	100.0				1.00	

Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain

Production	Tech Level	# Uses	Op Life
	0	s?	

Reference Al DuPont, Private Communication.

Notes Various mobility aids and restraints for exploration of Phobos/Deimos.
Rough guess for ample quantity of climbing gear: tethers, webbing, carabineers, pitons, chocks, friends, camalots, etc.

ID	Element Id	Element Type	Ref Design Id	# 6
	upLRV	veh		

Unpressurized Lunar Rover

Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	550.0	1.50	2.00	5.00	15.00	

Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain
	2.0	2.1	1.30	2.0		

Production	Tech Level	# Uses	Op Life
	6	da-h	

Reference EEI-188 pp126ff,

Notes Unpressurized cab and removable trailer. Based on LOTRAN with 0.65 vehicle to payload ratio for durability (Apollo LRV had 0.44). Total P/L 850 kg. Normally carries 2 crew and 490 kg P/L and can be outfitted for 4 crew. Max speed about 15 km/hr.

Locomotion power estimated by E = 0.08 Wh/kg/km • distance. Single lithium metal sulfide battery (48 kg, 36 V, 146 Amp h) gives total range of about 25 km. Power estimates are 2.15 kW maximum moving and 1.3 kW parking.

Technology needs include materials, greases, etc., for multi-use, dependable, easy to maintain vehicle.

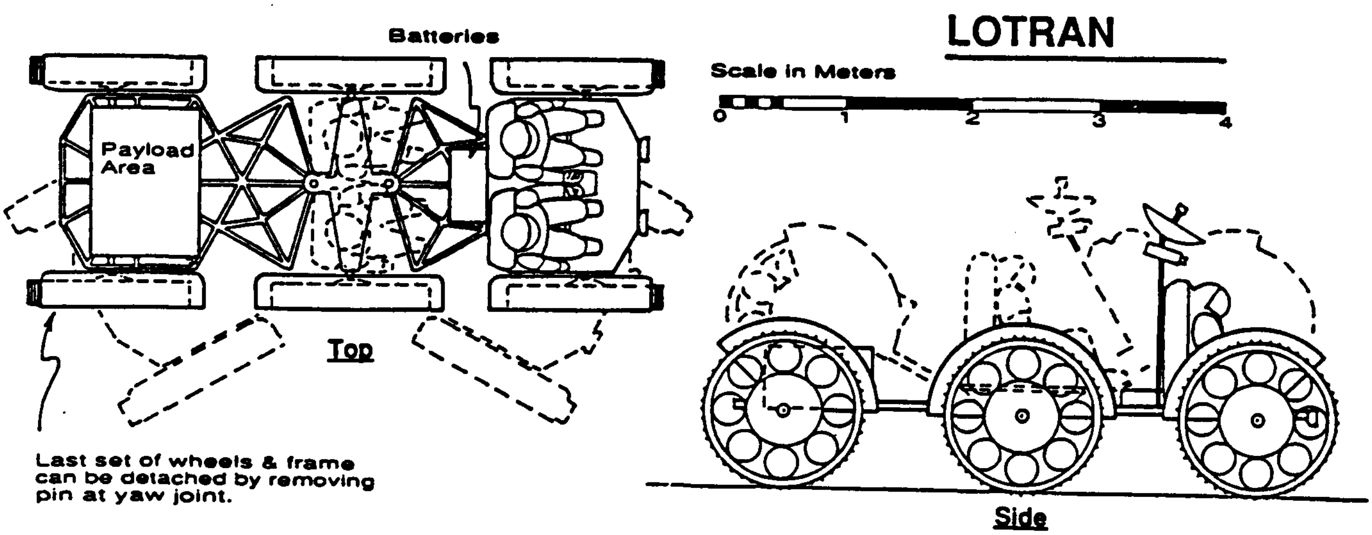

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Table 5.1.2.5-1 LOTRAN Electrical Energy Specification

LOTRAN FUNCTION	POWER (kw) MOVING	PARKED	TIME USED (hours)	ENERGY (kwh)
Locomotion	1.600	0	7	16.0
Control Electronics	0.08	0	7	0.6
Navigation	0.02	0	7	0.2
Lights	0.05	0.05	2	0.1
Thermal	0.1	0.1	8	0.8
Communications	0.15	0.15	8	1.2
Science and Applications	0.15		7	1.1
		1.0	1	1.0
TOTAL	2.15	1.3		21.0

Table 5.1.3-1 LOTRAN Configuration Definition

Three Articulated Sections:	Cab, Bed, and Trailer
Number of Wheels:	6
Wheel Type:	Metal-Elastic
Wheel Diameter:	1.35 m
Treadwidth:	1.8 m
Overall Width:	2.0 m
Suspension:	Passive + Active
Cab-to-Bed Wheelbase:	1.85 m
Bed-to-Trailer Wheelbase:	1.85 m
Total Length:	5.05 m
Detachable Section:	Trailer
Length Without Trailer:	3.2 m
Permanent Crew Stations:	Two on Cab
Installable Passenger Stations:	Two on Bed
Cargo Areas:	Bed and Trailer
Power Source:	Four Lithium-Metal Sulphide Batteries
Battery Specification (each):	36 volt, 146 Amp-hr, 48 kg
Total Energy Stored:	21 kwh
Locomotion Power Requirement:	1.6 kw
Maximum Power Requirement:	2.15 kw
Gross Payload Mass:	850 kg (2 Crew + 490 kg or 4 Crew + 130 kg)
Vehicle Mass:	550 kg
Total LOTRAN Mass (Loaded):	1,400 kg
Range:	100 km
Operational Speed:	15 km/hr
Maximum Driving Duration:	7 hrs

ID	Element Id	Element Type	Ref Design Id	# 7
	pLRV	veh		

Pressurized Lunar Rover Vehicle

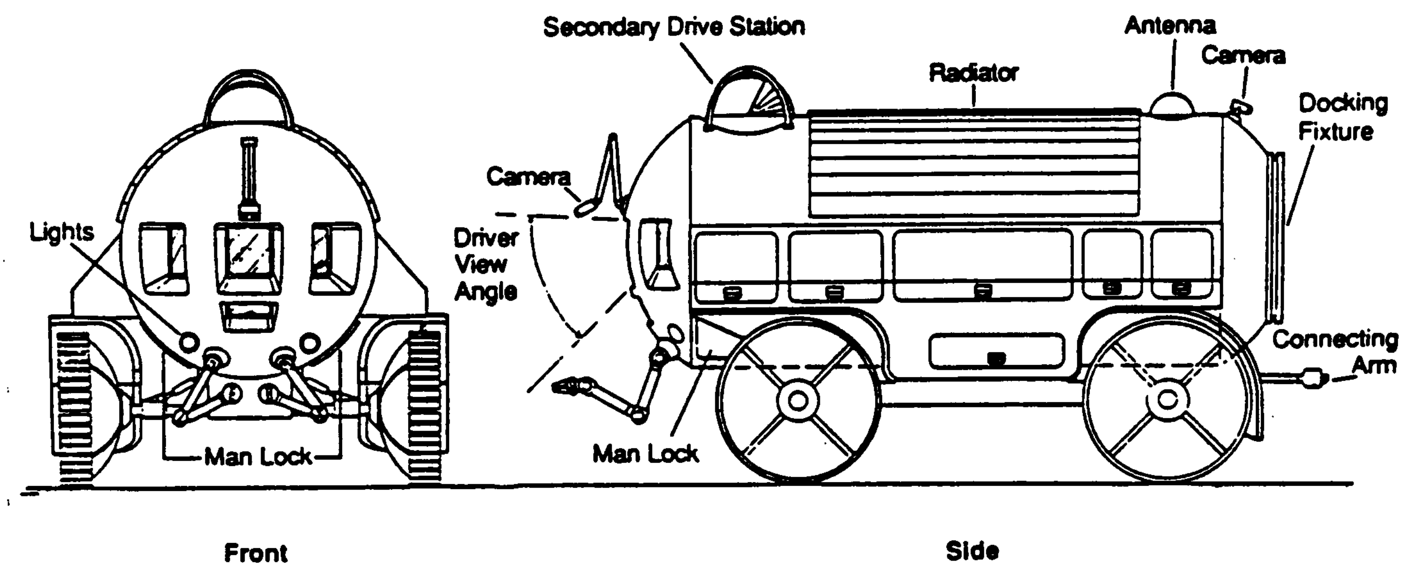
Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	5000.0	4.50	4.00	8.00	144.00	

Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain
	7.0	12.0	1.00			

Production	Tech Level	# Uses	Op Life
	1-2	da	

Reference EEI-188, sec 5.2, pp134ff

Notes Based on the PRCV (Primary Control Research Vehicle) element of MOSAP (MObile Surface APplications traverse vehicle), a lunar system design with modular elements that can be configured for various classes of missions. Capable of 50 km (100 km roundtrip) for 3 days with 2 crew, 500 kg experiments & payload. Additional capability can be obtained by adding other elements: Auxiliary Power Cart (APC), Habitation Trailer Unit (HTU), Experiment and Sample Trailer (EST).
Interior pressurized volume about 50 cu m. Provides for EVA with two airlocks that open downward. Contains teleoperations station; avionics and communication subsystems; galley, hygiene and waste stations; and living and sleeping quarters. Uses Shuttle fuel cells. Power needs assume 0.08 Wh/km/kg for locomotion and 1kW for other units. Thermal control is both passive (MLI, coatings) and active heating and cooling (1.4kW).



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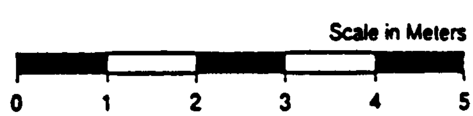


Figure 5.2.3.1-2 Primary Control Research Vehicle Interior Layout Drawing

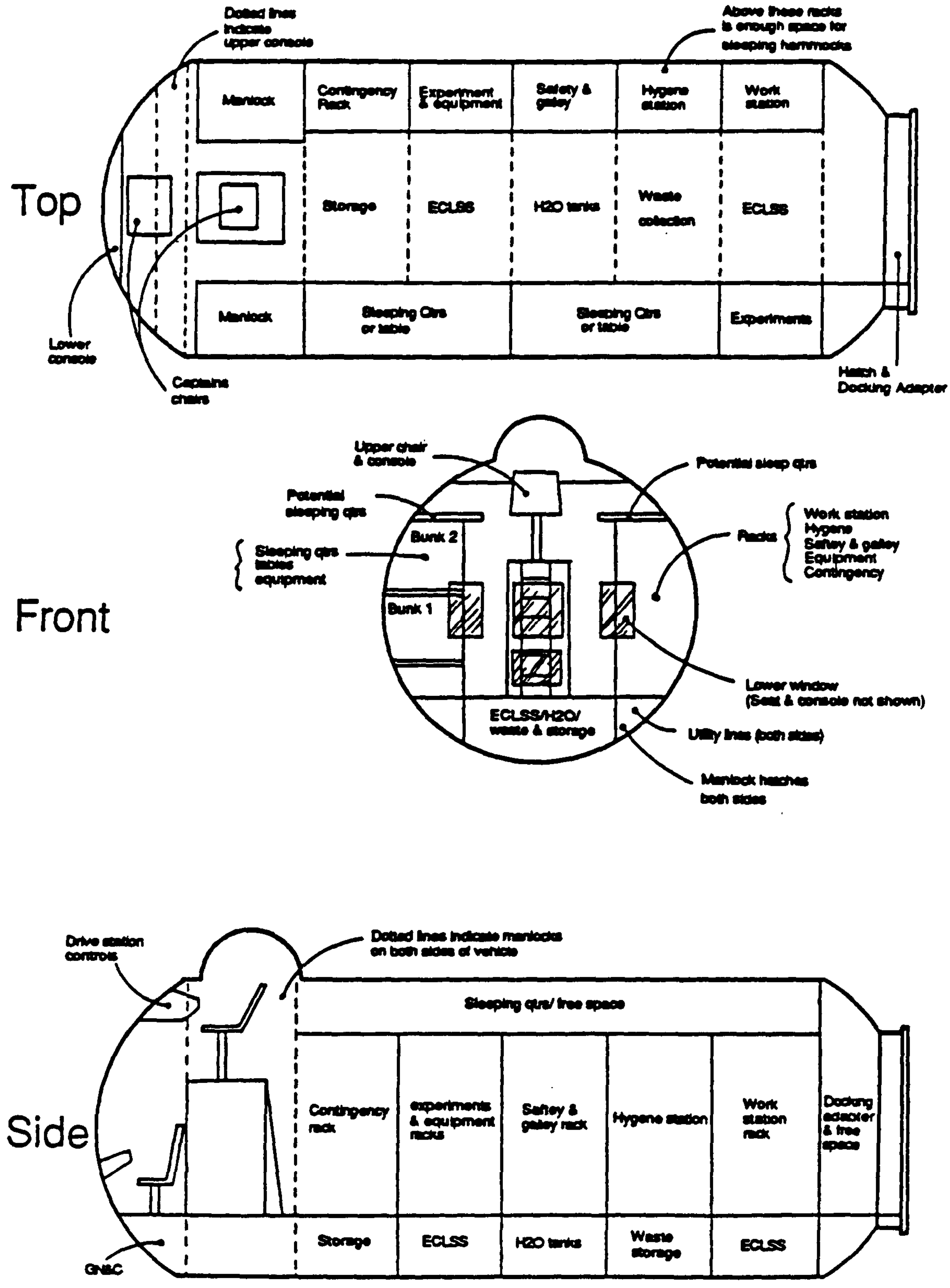


Table 5.2.3.1-1 Primary Control Research Vehicle Weight Statement

Structure and Pressure Vessel	
Inner Shell	490 kg
Outer Shell	500 kg
Other Structure	200 kg
Insulation	130 kg
Active Thermal System	
Radiator	160 kg
Pump	20 kg
Heat Exchanger	50 kg
Piping	100 kg
Refrigerant	300 kg
Power System	
Hydrogen Tanks	20 kg
Oxygen Tanks	15 kg
Water Tanks (incl. potable)	40 kg
Reactants	75 kg
Fuel Cell	90 kg
Power Distribution	100 kg
Wheels and Locomotion	300 kg
Man-locks	230 kg
Galley	70 kg
Personal Hygiene	90 kg
Emergency Equipment	30 kg
Avionics	90 kg
ECLSS	200 kg
Drive Stations	80 kg
Workstation	40 kg
Sleep Quarters	60 kg
Crew	360 kg
EMU's (3)	680 kg
Experiments and Payload	500 kg
TOTAL	5,020 kg

ID	Element Id	Element Type	Ref Design Id	# 8
	upMRV	veh		

Unpressurized Mars Rover Vehicle

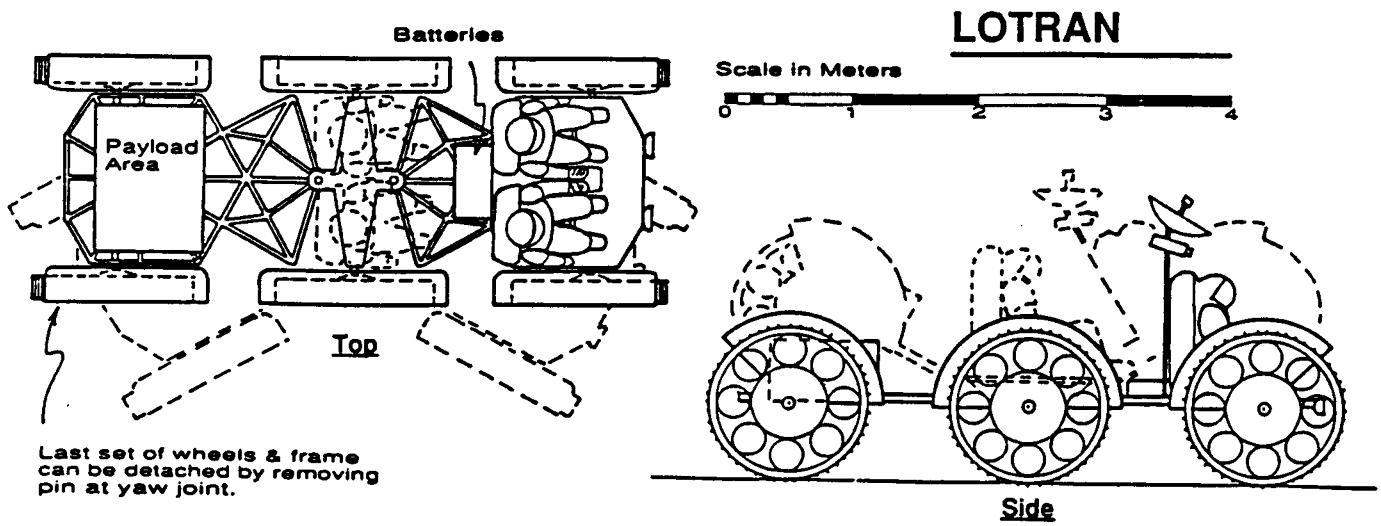
Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	1000.0	1.50	2.00	5.00	15.00	

Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain
	4.3	3.0	1.30			

Production	Tech Level	# Uses	Op Life
	4-7	s	

Reference EEI-188 sec 5.1, pp126ff.

Notes Limited use, unpressurized cab and trailer with max range of 25 km with one battery. Total payload 1140 kg. Rough upscaling of lunar design where structural mass and locomotion power are multiplied by 2 for increased gravity. Locomotion Energy estimated as 0.16 Wh/kg/km * distance.



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ID	Element Id	Element Type	Ref Design Id	# 9
	pMRV	veh		

Pressurized Mars Rover Vehicle

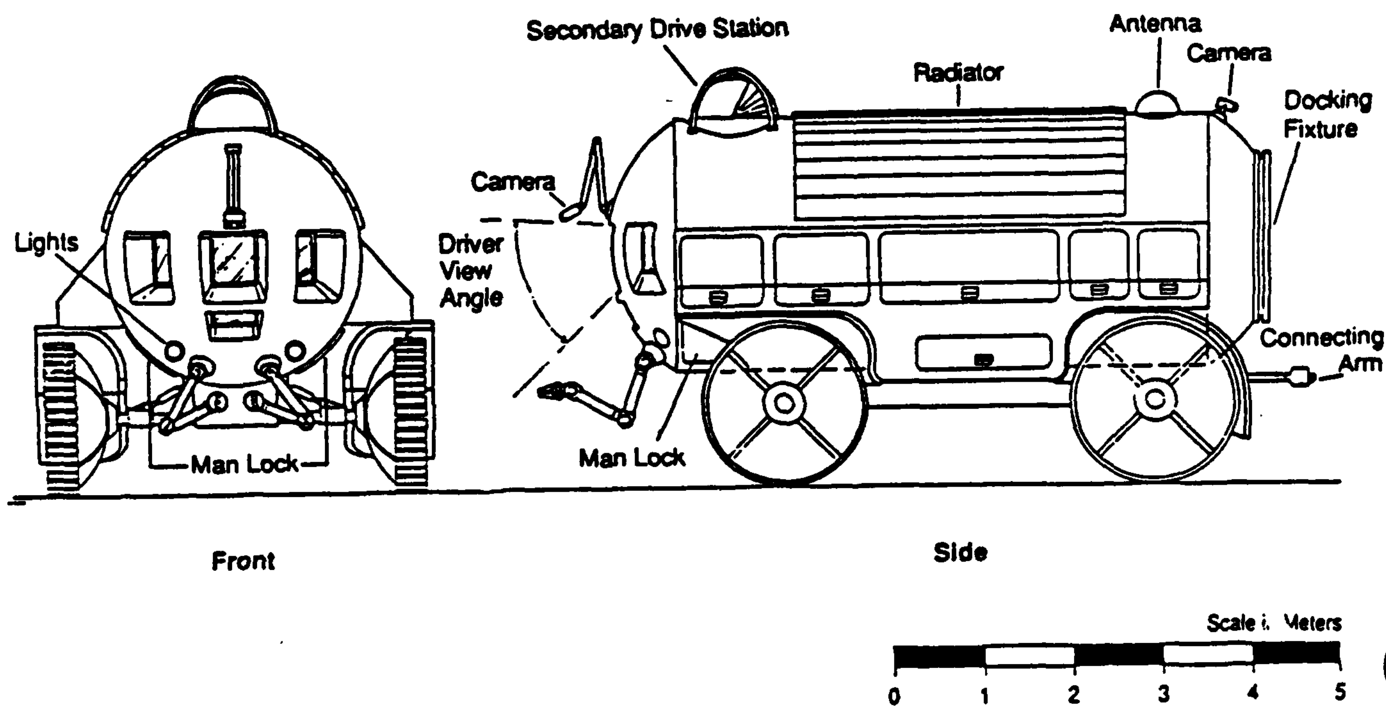
Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	6500.0	4.50	4.00	8.00	144.00	

Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain
	14.0	24.0	2.00			

Production	Tech Level	# Uses	Op Life
	1	da	

Reference EEI-188, sec 5.2 pp134ff;

Notes Based on pressurized LRV with rough upscale of structures by factor of 2 to account for increased Martian gravity.



ID	Element Id	Element Type	Ref Design Id	# 10
	digger	veh		

Excavator/Digger

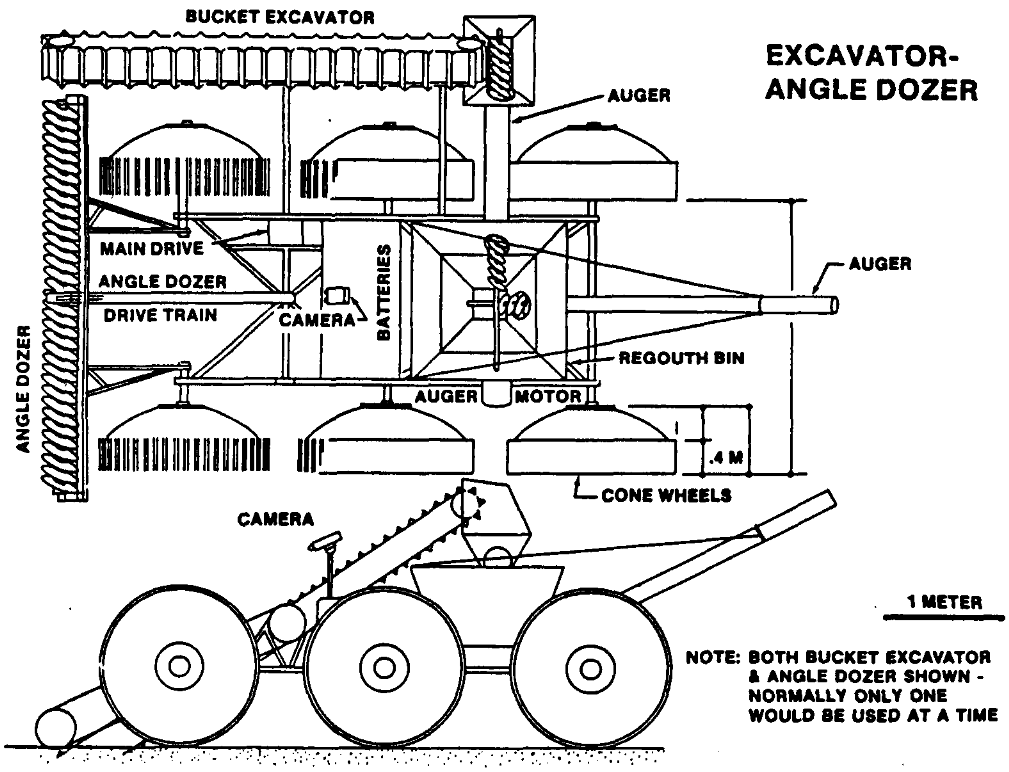
Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	1900.0	3.00	4.50	7.00	94.50	

Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain
	5.0					

Production	Tech Level	# Uses	Op Life
	2	da-h	

Reference Graf, p8

Notes Lunar design. Bucket wheel excavator mounted next to large excavator vehicle for excavtion and dozing. Can excavate 5000 kg/hr. Stable to 10° slope. Power is 2.3 kW locomotion, 2.7 kW excavator.



ID	Element Id	Element Type	Ref Design Id	# 11
	Truck-L2.5	veh		

Lunar Truck, 2.5 t

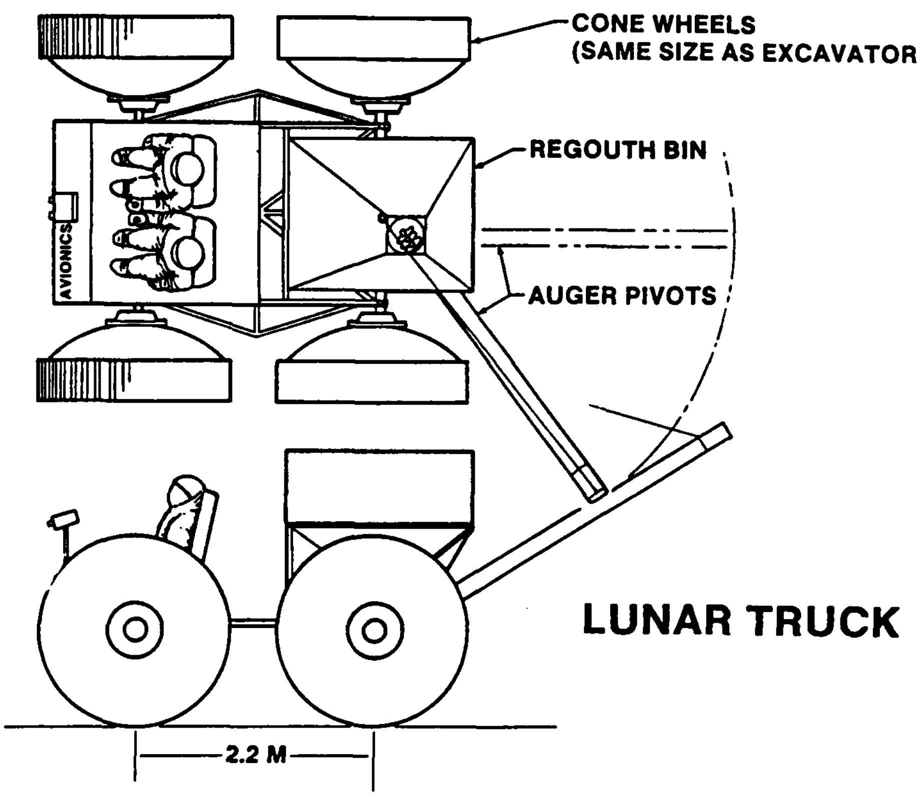
Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	1900.0	2.50	3.50	5.00	43.75	

Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain

Production	Tech Level	# Uses	Op Life
	2	da-h	

Reference Graf pp7f

Notes 5 km/hr offroad, 10 km/hr on smooth surface. For hauling and loading regolith. Manual and remote operation. 2.5 t payload.



LUNAR TRUCK

ID	Element Id	Element Type	Ref Design Id	# 12
	Crane	veh		

Lunar Crane/Hoist

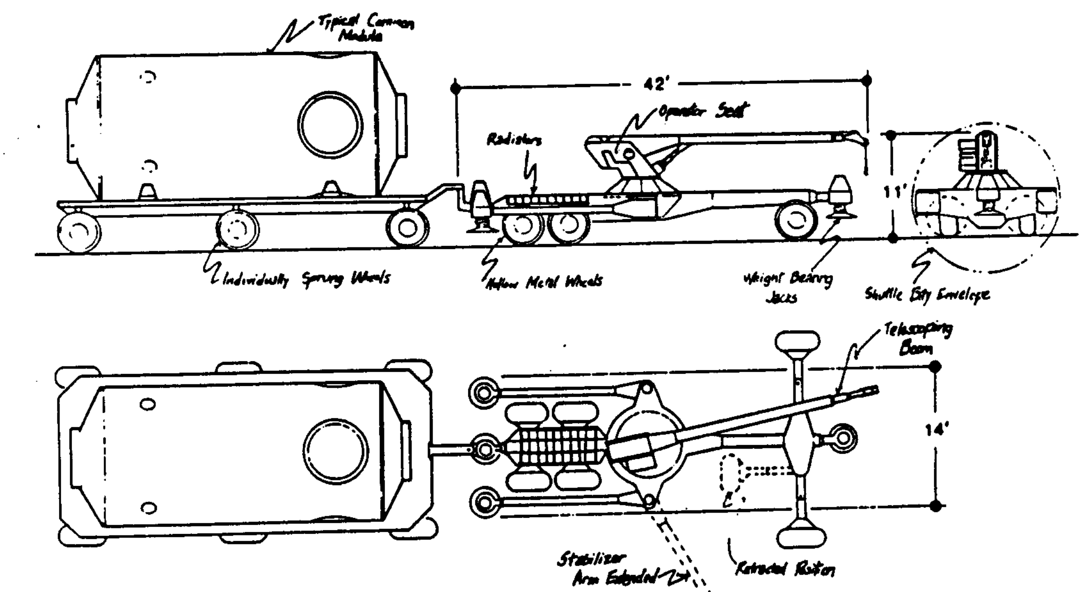
Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	1900.0	3.50	6.00	13.00	273.00	

Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain

Production	Tech Level	# Uses	Op Life
	2	da-h	

Reference EEI-85-109B pp5-7

Notes Designed to pack in 4.5m x 11m STS envelope. Includes 3 stabilizers (2 deployable, 1 fixed). Stabilizers use Acme threaded jacks as does the telescoping boom. Cable is winch operated. Wheels are hollow metal, and driven by electric motors. Trailer hitch at back for towing cargo. Operated by crewman seated near base of boom.



Lunar Crane
& Trailer with Common Module

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ID	Element Id	Element Type	Ref Design Id	# 13
	T-ramp	veh		

Pressurized Tunnel Ramp

Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	2788.0	3.50	2.00	9.00	63.00	

Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain

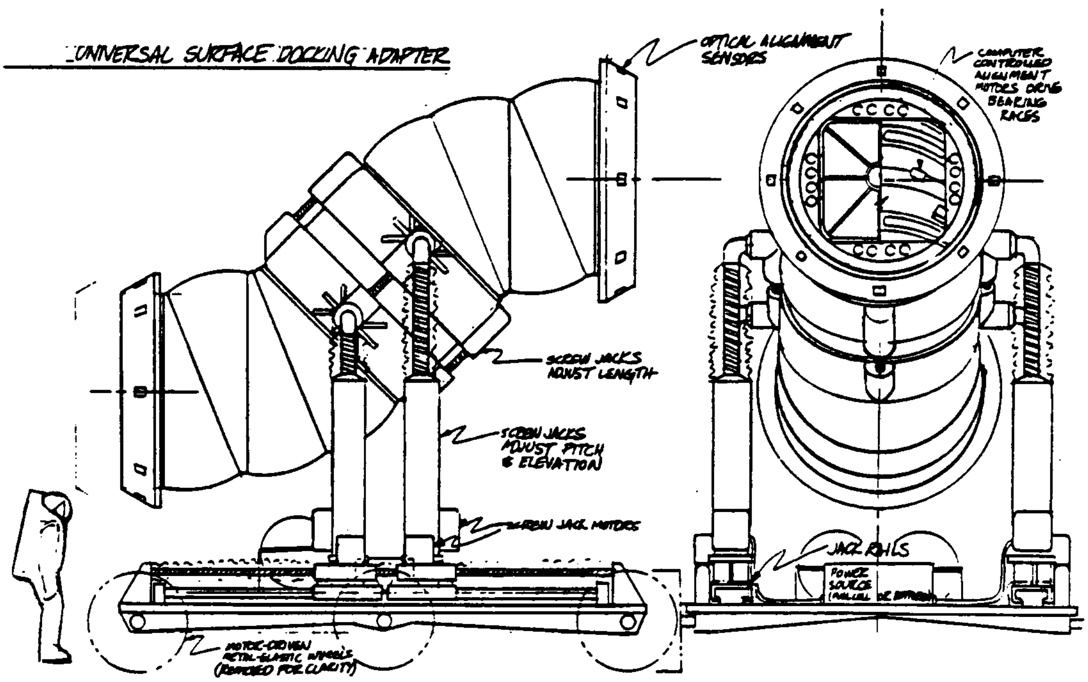
Production	Tech Level	# Uses	Op Life
	2	da-h	

Reference EEI-178 pp69ff

Notes Provides for pressurized Transfer of crew to and from landers. Trailer with special pressurized tunnel and universal docking adapters/hatches.

Component mass estimates (kg) are:

Tunnel	1200
Docking Adaptors	430
Cart	900
Actuators	150
Power&Controls	100



ID	Element Id	Element Type	Ref Design Id	# 14
	PRV-L40	veh		

Propellant Refil Vehicle

Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	14000.0	4.00	7.00	6.00	168.00	

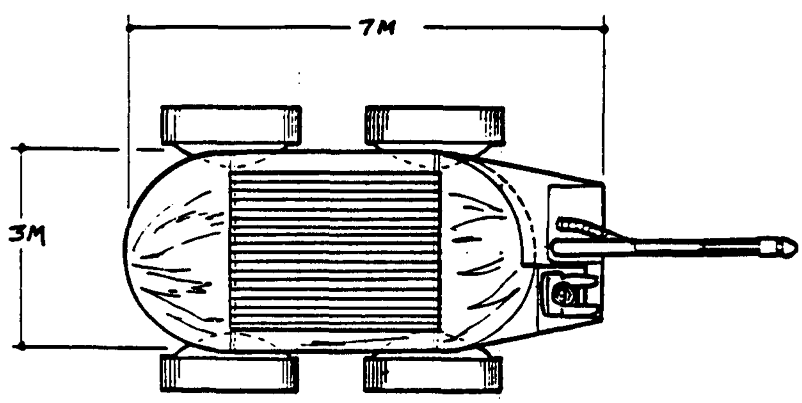
Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain

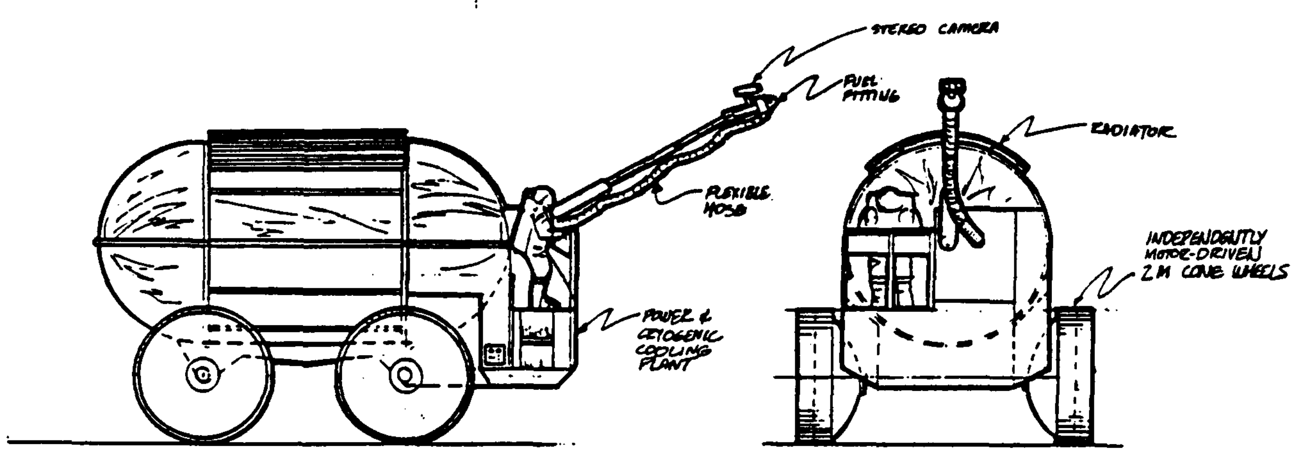
Production	Tech Level	# Uses	Op Life
	2	da	

Reference EEI-178 pp71ff

Notes Consists of storage tank, transfer support equipment, and equipment to run the vehicle (fuel cells, telemetry, etc). Several may be needed depending on LEM size. Carries 35 cu m of propellants (2.5 t LH2, 40 t LOX). Has boil-off line directed through refrigeration unit for reliquefaction.

Mass estimates are very rough.





LANDER PROPELLANT TANKER

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ID	Element Id	Element Type	Ref Design Id	# 15
	Cord Cart	veh		

Lunar Power Cord Cart, 1 km

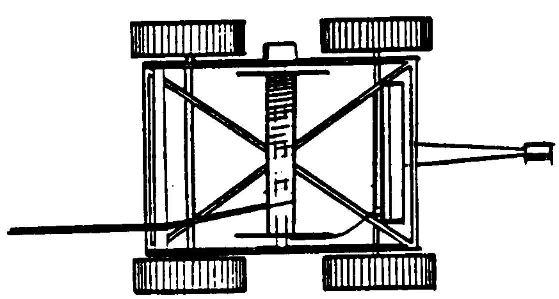
Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	820.0	1.10	1.40	2.00	3.08	

Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain

Production	Tech Level	# Uses	Op Life
	2		y

Reference EEI-178 pp74ff

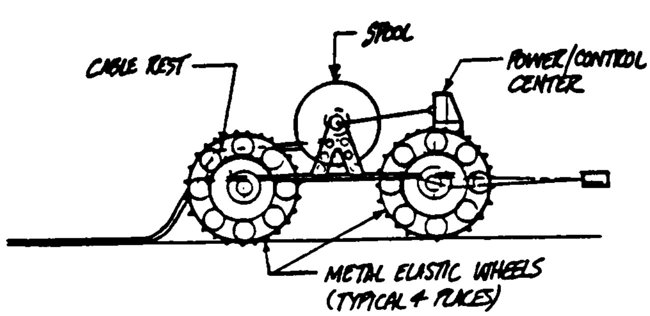
Notes Provides supplemental power to LEM from central unit at base via 1 km long cord on a spool mounted to 4 wheeled vehicle. Provides some power conditioning (transformer, rectifier).

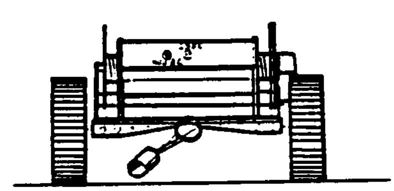


ELECTRIC CORD SYSTEM (1 kilometer)

Conductor	490 kg
Insulation	250 kg
Power Conditioner	20 kg
Cart	90 kg
TOTAL	820 kg

Dimensions 2.0 m Long 1.4 m Wide 1.1 m High





CORD CART

ID	Element Id	Element Type	Ref Design Id	# 16
	Fuel Cell Cart	veh		

Lunar Fuel Cell Power Cart

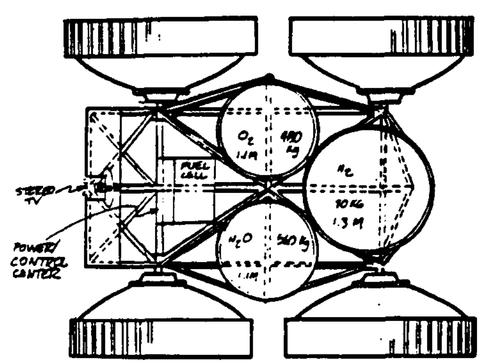
Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	1290.0	1.30	1.30	4.30	7.27	

Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain

Production	Tech Level	# Uses	Op Life
	2	da	y

Reference EEI-178 pp76f

Notes Alternative to Cord Cart/Base Power. Provides supplemental power to LEM via 7 kWe STS fuel cell system (FCS). Doubling power capacity to 14 kW would increase mass about 7%.

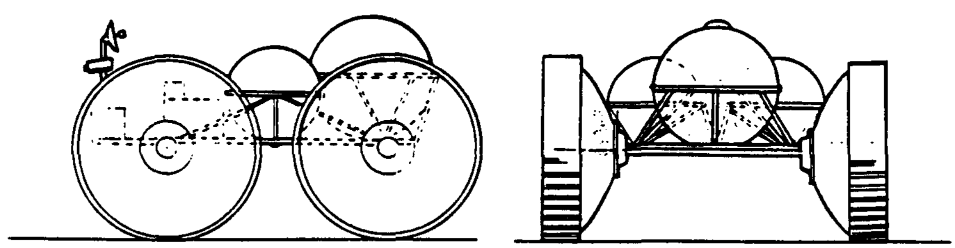


FUEL CELL POWER CART (2 kilowatts, 28 days)

Tanks		
	Hydrogen	190 kg
	Oxygen	130 kg
	Water	130 kg
Fuel Cell		90 kg
Solar Panel (1 kw)		40 kg
Cart		150 kg
DRY MASS		730 kg
Reactants		560 kg
TOTAL		1,290 kg

Dimensions 4.3 m Long 1.3 m Wide 1.3 m High

Tanks		
	Hydrogen	1.3 m Diameter
	Oxygen	1.1 m Diameter
	Water	1.1 m Diameter



POWER CART
(4x4 TRUCK CHASSIS)

C - 6

ID	Element Id	Element Type	Ref Design Id	# 17
	TeleAsst	veh		

Teleoperated Assistant

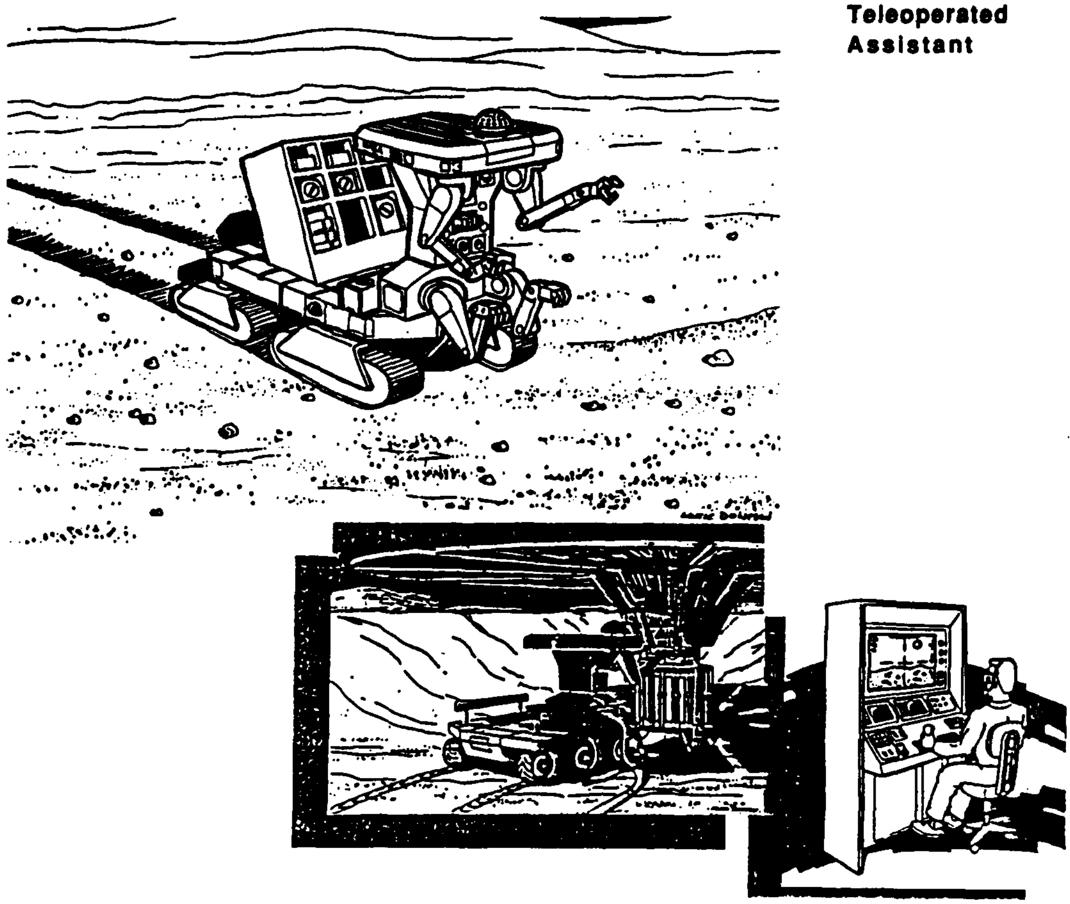
Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	1290.0	4.30	1.30	1.30	7.27	

Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain

Production	Tech Level	# Uses	Op Life
	1-2	da	y

Reference

Notes Mobile, semi-intelligent, general purpose robot for supporting surface operations. Can be teleoperated from Planetary surface. Only broad concept defined. Hence mass and volume estimates are extremely crude guess.



Teleoperated Assistant

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ID	Element Id	Element Type	Ref Design Id	# 18
	EMU-Ph	EVA		

Phobos Extravehicular Mobility Unit (EMU)

Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	135.9	1.92	0.75	0.72	1.04	

Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain

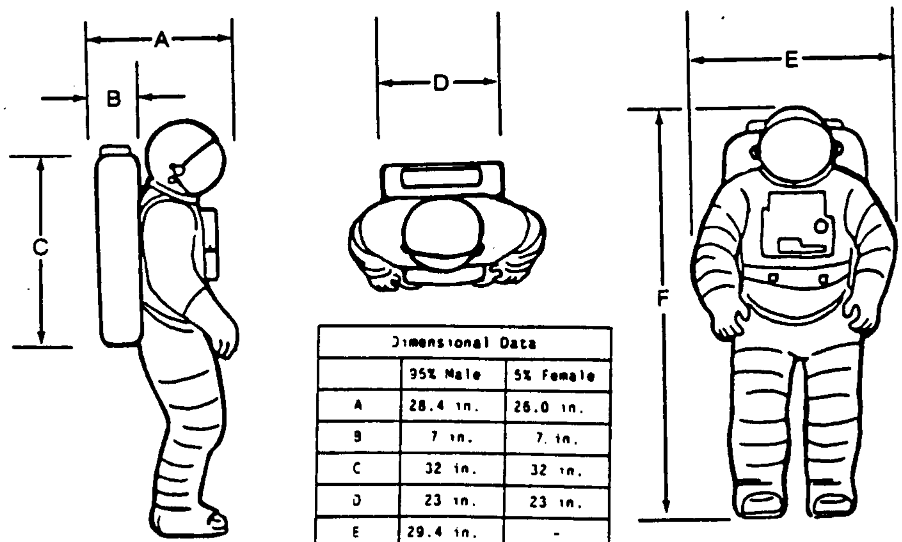
Production	Tech Level	# Uses	Op Life
	7	da	

Reference Buffkin[88]
JSC-20466

Notes For "weightless" use. Based on the STS EMU in the EVA catalog. Reflects 95 percentile male size (28.4"x29.4"x75.5", 300 lb). Includes 19.3 kg consummables for two 6-hour EVAs. Standard interface attachments: MMU, mini-workstation, tool caddies, EMU television system, EMU lights, wrist and waist tethers.

For SS design, Bufkin et al give fully charged SS EMU as 500 lb. SS CETF annual estrimates for weekly EVAs are (annual/weekly average):

Consummables	300 kg/6.4	0.344 cu m/0.007
Limited Life Items (batteries, filters, etc.)	300 kg/12.9	0.344 cu m/0.02



Dimensional Data		
	95% Male	5% Female
A	28.4 in.	26.0 in.
B	7 in.	7 in.
C	32 in.	32 in.
D	23 in.	23 in.
E	29.4 in.	-
F	75.5 in.	67.5 in.

ID	Element Id	Element Type	Ref Design Id	# 19
	EMU-LS	EVA		

Lunar Surface Extravehicular Mobility Unit

Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	229.2	1.92	0.75	0.72	1.04	

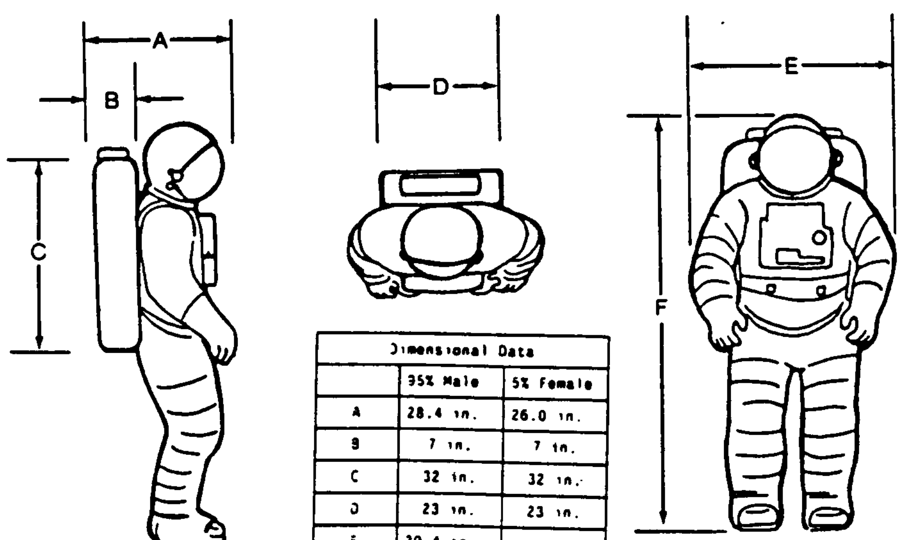
Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain

Production	Tech Level	# Uses	Op Life
	6	da	

Reference Bufkin[88]
EEI87-172
JSC-20466

Notes Mass based on EEI Lunar Surface Ops Study and includes 160 lb suit assembly and 346 lb Portable Life Support System (PLSS). Volume same as STS EMU. Includes consumables for two 6-hour EVAs.

The SS SSA accommodates from 50 percentile female to 95 percentile male with two HUT (Hard Upper Torso) sizes and various fabric elements. Estimates for spares are 50% of prime or 2 complete EMUs.



Dimensional Data		
	95% Male	5% Female
A	28.4 in.	26.0 in.
B	7 in.	7 in.
C	32 in.	32 in.
D	23 in.	23 in.
E	29.4 in.	-
F	75.5 in.	67.5 in.

ID	Element Id	Element Type	Ref Design Id	# 20
	EMU-MS	EVA		

Mars Surface Extravehicular Mobility Unit

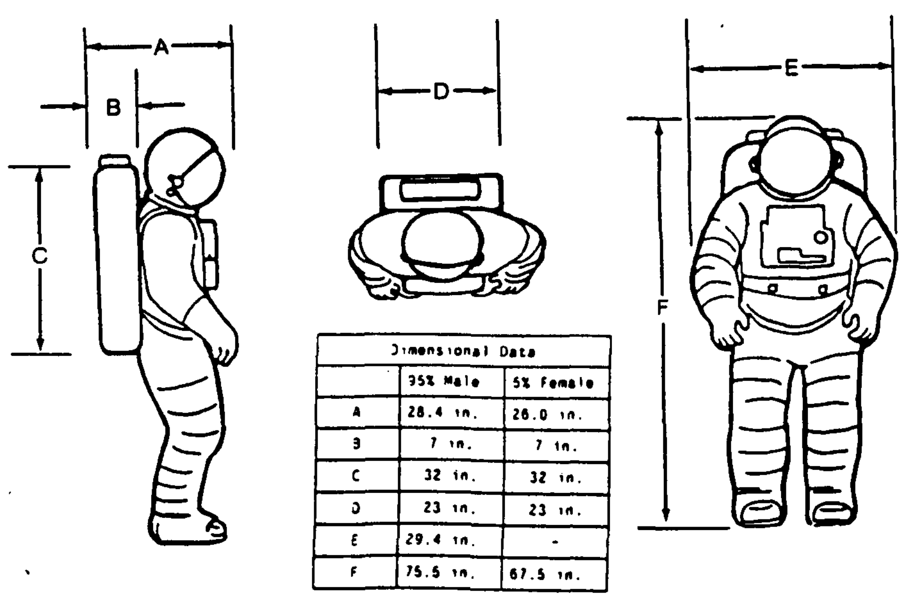
Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	229.2	1.92	0.75	0.72	1.04	

Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain

Production	Tech Level	# Uses	Op Life
	6	da	

Reference Bufkin[87]
Christiansen pp51ff
JSC-20466

Notes Estimates based on the Lunar EMU (EMU-LS). Requires further definition.



Dimensional Data		
	95% Male	5% Female
A	28.4 in.	26.0 in.
B	7 in.	7 in.
C	32 in.	32 in.
D	23 in.	23 in.
E	29.4 in.	-
F	75.5 in.	67.5 in.

ID	Element Id	Element Type	Ref Design Id	# 21
	RadPG	EVA		

Radiatiation Protection Garment

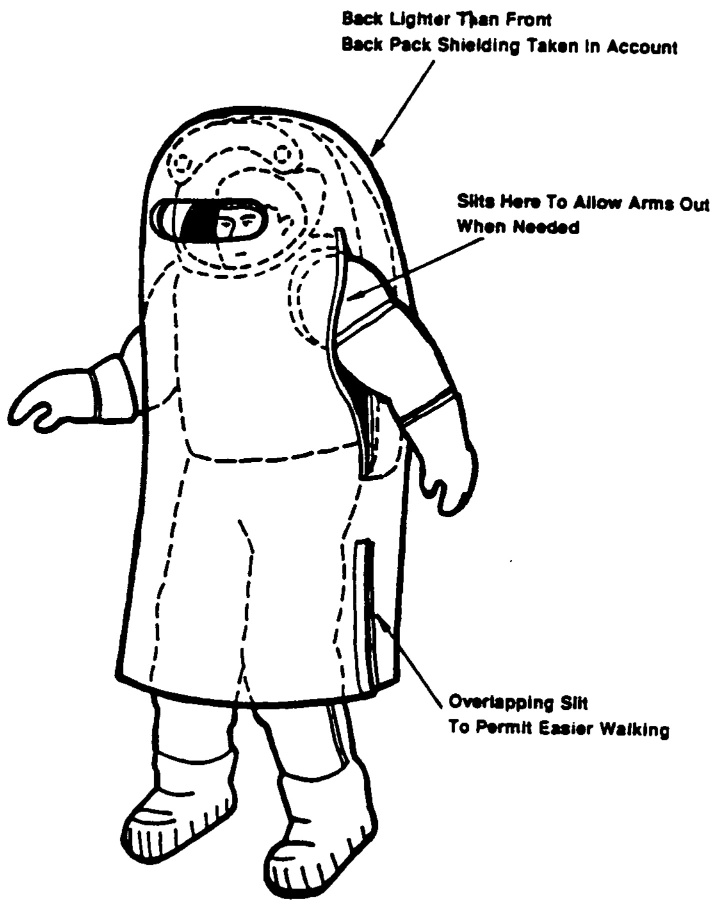
Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	170.0	1.91	0.76	0.15	0.22	

Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain
	0.0	0.0	0.00			

Production	Tech Level	# Uses	Op Life
	2	da	

Reference Gill pp45ff

Notes Emergency protection garment that provides 8 gm/ sq cm shielding to reduce dose by a factor of 5-7 from unshielded suit. Cannot support entire flare period (assuming 25 REM maximum emergency dose).
Allows emenrgency dose of about 5-7 REM compared to 36 over a 3 hour period for a flare event like the 6 hr peak of the Aug 72 event (123 RAD or 12 REM/hr assuming RBE=1.15).
Sized for 99 percentile male: Standing Ht 75", Sitting Ht 39", Shoulder wd 22", Knee to buttocks 23.4", Foot Length 12".
Can be made of 20 oz per yard Carbon fiber cloth (118 layers, 3" thick). Might be reduced in back to account for backpack shielding.



ID	Element Id	Element Type	Ref Design Id	# 22
	Hab-Inf 16m	env		

Inflatable Habitat, 16 m diameter

Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	18500.0	16.00	16.00	16.00	2145.00	40.00

Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain

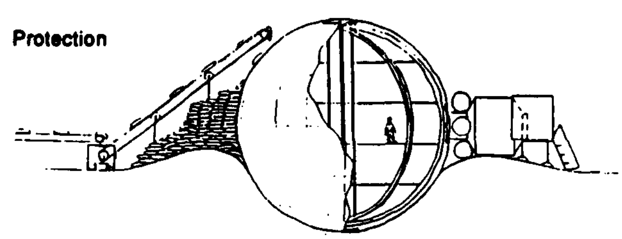
Production	Tech Level	# Uses	Op Life
	2	s	

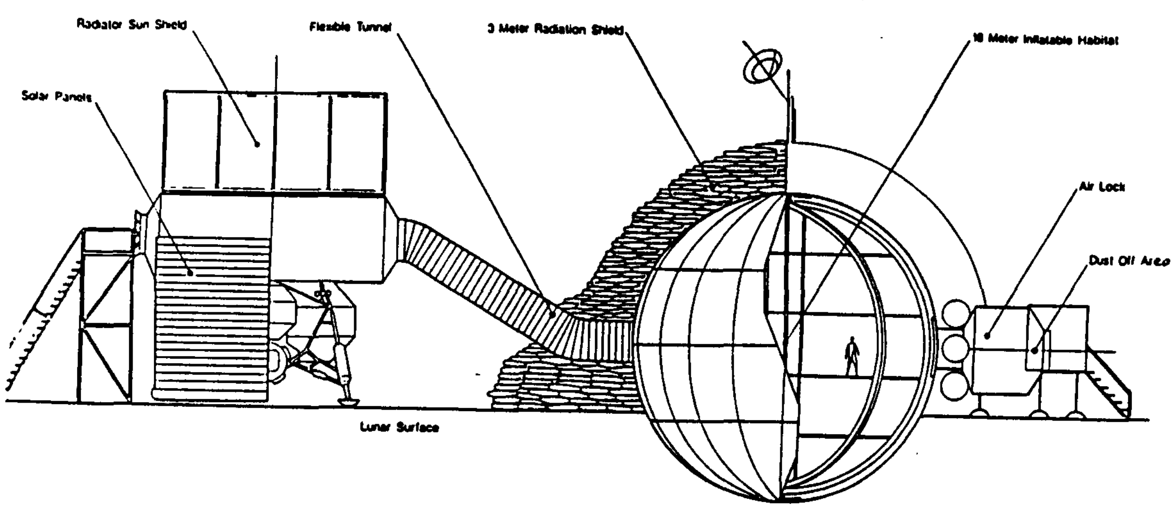
Reference Roberts[88]

Notes 16 m diameter, 2143 cu m open volume, 101.4 kPa internal pressure (standard Earth at sea level). Sized for 12 crew with 178.25 cu m/crewman. 594 sq m floorspace on 4 floors. Envelope is high strength multi-ply fabric with impermeable inner layer and thermal coated outside. Using parameters for DuPont Kevlar-29, thickness of 0.144 mm has breaking strength of 525 N/m and structural safety factor of 5. Structural layer thickness assumed to be 5 mm. Stowed volume estimated using 10 to 1 packaging ratio. Interior is cage with curved beams that run about the envelope. Covered with regolith for radiation protection. Component mass estimates (t): Envelope 2.2, Primary Structure 9, Flooring 6, Walls 1.3. Rough scalings of 7m diam/14t for crew of 8, and 5.6m diam/9t for crew of 4, obtained by using volume of 178.25 cu m per crewman, and scaling mass as ratio of surface areas of spheres. Construction time estimated (very rough, does not include excavation/outfit) as 3 crew for 48 hrs to erect + 12 hrs to inflate.

Inflatable Habitat

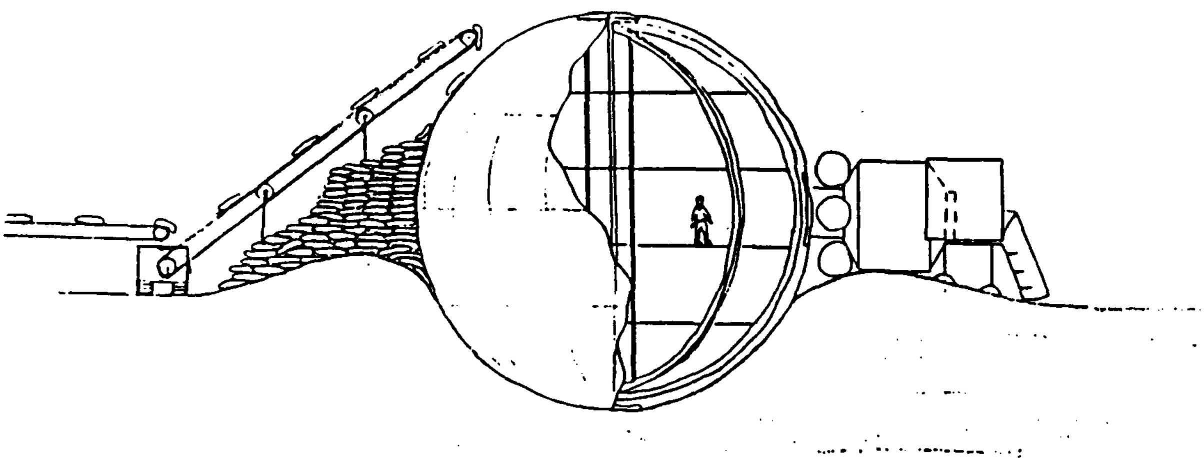
- Bagged Regolith Provides Radiation Protection
- Excavation By Blasting





TASK SUMMARY CHART

TASK	EQUIP-MENT	NON-REUSABLE HARDWARE	EXCAVATION VOLUME	OPERATIONS	EVA TIME crew X duration (hrs)
Bury Habitat Module (1 module)	• Truck • Excavator • Conveyor • Bagger	Truss 2400 kg Bags 510 kg	310 m^3	• Prepare Platform • Deploy Truss Structure • Excavate for Bagging • Place Bags	2 X 6 2 X 12 - 4 X 12
Prepare Landing Site (6 sites)	• Rover • Excavator • Truck • Angle Dozer	Explosives 25 kg Landing Aides 40 kg	910 m^3	• Survey, Map, Select Site • Excavate for Filling • Fill Craters • Smooth Surface	2 X 16 - 2 X 8 -
Transfer Payload (1 cargo lander 15000 kg)	• Truck	Truss 250 kg Block and Tackle 40 kg	-	• Deploy Truss • Off Load	2 X 3 2 X 10
Smooth Roads (3.3 km)	• Rover • Excavator • Angle Dozer	Nav. Aides 100 kg	3400 m^3	• Select Path, Set Nav. Aides • Excavate for Backfill • Fill Craters • Smooth Surface	2 X 6 - 2 X 6 -
Cover Inflatable (14.3 m sphere)	• Truck • Drill Core • Excavator • Bagger • Conveyor	Explosives 100 kg Bags 950 kg	710 m^3 bagged (≈ 500 m^3 blasted)	• Drill Core, Blast Hole • Excavate for Backfill • Backfill • Place Bags	2 X 4 - 2 X 6 4 X 24
Excavate for Oxygen (2 MT/month)	• Excavator • Truck	- -	1470 m^3	• Excavate • Haul to Sizer • Haul to Processor • Return Spent Fines	- - - -



ID	Element Id	Element Type	Ref Design Id	# 23
	ECLS-LSB	env		

Lunar Base ECLS and consummables resupply

Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	4803.0				27.00	

Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain
		10.5		24.0		

Production	Tech Level	# Uses	Op Life
	3-4	s	

Reference Hypes[88b]

Notes

Lunar design. Regenerative system sized for crew of 4. Supports 90% water reclamation, partial O2 reclamation. Food from Earth. Based on SS ECLS elements.
90 day resupply for crew of 4 is 2705 kg, 9.8 cu m.

Increasing crew to 8 gives the following deltas:

Launch	9042 kg	33.4 cu m
Resupply	2235 kh	11.4 cu m

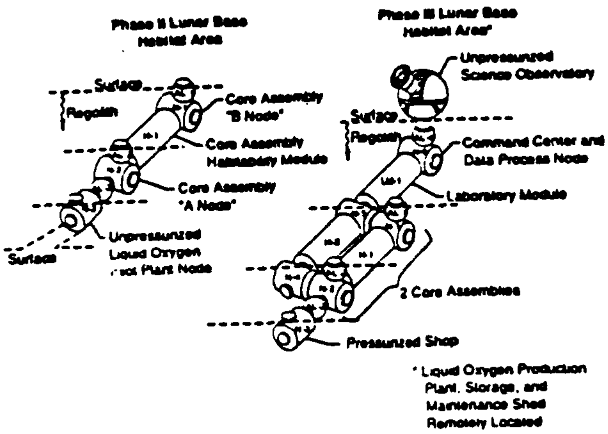


Figure 2. Concepts of a Phase II and Phase III Lunar Base Habitat Area

Table 8 - ECLS System Tankage Requirements

(a) Initial Launch and 1st 90 Days

Tankage	Number of tanks	Unit (Dry) Weight, lb	Unit (Dry) Volume, ft^3	Subtotal (Dry) Weight, lb	Subtotal (Dry) Volume, ft^3
Potable H_2O storage	5	50	3.1	250	15.5
Hygiene and wash H_2O collect and hold	14	50	3.1	700	43.4
Urine and flush H_2O collect and hold	14	25	1.6	350	22.4
Condensate collect and hold	12	25	1.6	300	19.2
Oxygen	2	75	10.0	150	20.0
Nitrogen	6	175	20.0	1050	120.0
			TOTAL	2800	240.5

(b) 90-Day Resupply

Tankage	Number of tanks	Unit (Dry) Weight, lb	Unit (Dry) Volume, ft^3	Subtotal (Dry) Weight, lb	Subtotal (Dry) Volume, ft^3
Water resupply	25	50	3.1	1250	77.5
Oxygen resupply	2	75	10.0	150	20.0
Nitrogen resupply	3	175	20.0	525	60.0
			TOTAL	1925	157.5

Table 6.- Phase II Lunar Base ECLS System Weight, Volume, and Power Totals

	Module 1	Node 1	Node 2	Node 3	Airlock 1	Airlock 2	Airlock 3	Total
Launch weight, lb	4880	2766	2357	-	256	256	75	10590
Launch volume, ft^3	486	221	222	-	16	10	-	955
Resupply weight, lb	4985	268	623	-	42	42	12	5972
Resupply volume, ft^3	277.65	26.73	30.15	-	4.59	4.59	2.07	345.78
Power, kW	2.365	4.015	2.189	-	0.700	0.700	0.525	10.494

Table 7.- Phase III Lunar Base ECLS System Weight, Volume, and Power Totals

	Phase II totals	Additional Required to Evolve to Phase III*								Phase III Totals
		Habitability module 2	Laboratory module 1	Node 3	Node 4	Node 5	Node 6	Airlock 4	Subtotal	
Launch weight, lb	10590	7034	3378	1355	2932	3209	1977	50	19935	30525
Launch volume, ft^3	955	582	123	21	191	204	49	8	1178	2133
Resupply weight, lb	5972	3418	553	288	251	142	266	10	4928	10900
Resupply volume, ft^3	346	303	42	18	7	14	17	2	403	749
Power, kW	10.494	2.365	1.120	0.930	1.300	3.580	1.100	0.300	10.695	21.189

*Observatory, LOX Plant, Storage and Maintenance Shed are unpressurized and contain no ECLS system components.

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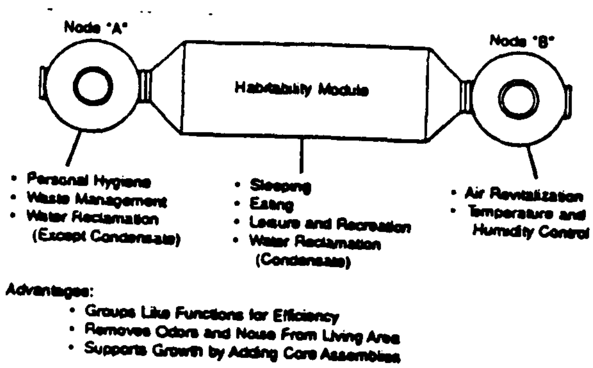

Figure 1. Node/Module/Node Core Assembly

Table 4 - Processes and Subsystems of the Space Station ECLS System

Water Reclamation
- Urine - TIMES
- Wash - RO
- Condensate - MF

Air Revitalization
- CO_2 Removal - MS
- CO_2 Reduction - BOSCH
- O_2 Generation - H_2O Electrolysis
- Contaminant Control - Catalytic Oxidation
- O_2/N_2 Make Up - Cryogenic

Temperature and Humidity Control
- Forced Air Circulation
- Water/Glycol Loop
- Condensing HX

Waste Management
- Shredder/Dryer

Trash Management
- Compaction

Food
- Freeze Dried, Frozen
- Disposable Utensils

Clothing
- Reusable

Habitability
- Shower
- Washer/Dryer

EVA
- 3 psia Suits, 100% O_2

Table 5 - ECLS System Distribution Within the Node/Module/Node Core Assembly

Unit or component	Habitability module	Node A	Node B
Air Revitalization Subsystem			
• CO_2 Collection			✓
• CO_2 Reduction			✓
• O_2 Recovery			✓
• Contaminant Control			✓
• Atmosphere Composition Control (Instrument)	✓		
• O_2/N_2 Make-Up	✓		
• Temperature/Humidity Control	Intake & Filter	Intake & Filter	Cond. HX
Water Management Subsystem			
• Potable H_2O Storage (Incl. Reclaimed Condensate)	✓		
• Humidity Condensate Reclamation Unit	✓		
• Wash H_2O Reclamation Unit and Processed H_2O Storage		✓	
• Urine H_2O Reclamation Unit and Processed H_2O Storage		✓	
Waste Management Subsystem			
• Feces Collection		✓	
• Feces Processing and Storage		✓	
• Urine Collection		✓	
• Trash Compaction and Storage	✓		
Personal Hygiene Subsystem			
• Shower		✓	
• Clothes Washer/Dryer		✓	
• Sink		✓	
Nutritional Support			
• Galley	✓		
• Food Storage	✓		
Crew Accommodations*			
• Bunks	✓		
• Lockers	✓		
• Clothing Storage	✓		
• Recreation	✓		
• Exercise Devices	✓		

*Crew accommodations components are not included in the ECLS system weight, volume, and power totals.

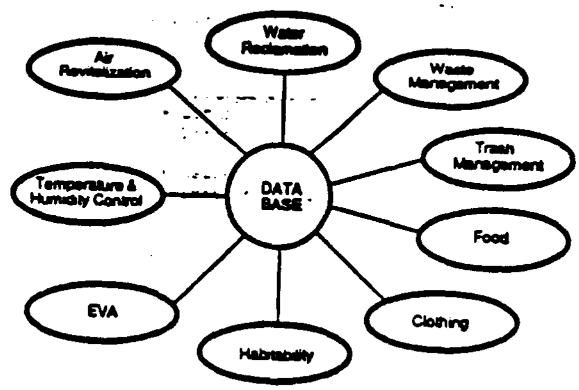

Figure 3. Functional Groups of the ECLS System Candidate Technology Data Base

ID	Element Id	Element Type	Ref Design Id	# 24
	TCS-L	env		

Lunar Base Thermal Conrol System

Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	4315.0				18.00	

Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain
	50.3			24.0		

Production	Tech Level	# Uses	Op Life
	2-3		

Reference Simonsen[88]

Notes Derived from SS and sized for crew of 4 at initial lunar base. Regolith provides insulation from surface so that heat rejection is the major concern. Two loop, cascaded Vapor Cycle System. Two loops operated for heat rejection during day (130° C surface temperature) and a bypass prevents over rejection at night (-150° C surface temperatures). Rejection temperatures of 13° C for metabolism and -6° C for equipment.

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TABLE 4: Acquisition Summary: Initial Phase

Item	HM-1	N-1	N-2	AL-1	AL-2	AL-3
WEIGHT (kg):						
Air Temperature & Humidity Control	149	143	57	57	57	57
Equipment Heat Acquisition 2° C	355	68	104	100	100	100
21° C	390	200	191	195	195	195
Support Loop (2° C & 21° C)	81	73	146	0	0	0
TOTAL	975	484	498	352	352	352
VOLUME (m^3):						
Air Temperature & Humidity Control	1.73	.71	.40	.40	.40	.40
Equipment Heat Acquisition 2° C	.21	.07	.09	.08	.08	.08
21° C	.22	.10	.10	.10	.10	.10
Support Loop (2° C & 21° C)	.04	.03	.07	0	0	0
TOTAL	2.20	.91	.66	.58	.58	.58
POWER (kW):						
Air Temperature & Humidity Control	.49	1.04	.15	.15	.15	.15
Equipment Heat Acquisition 2° C	.32	.02	.07	.07	.07	.07
21° C	.29	.18	.18	.19	.19	.19
Support Loop (2° C & 21° C)	.03	.03	.06	0	0	0
TOTAL	1.13	1.27	.46	.41	.41	.41

Table 5: Acquisition Summary: Growth Phase

Item	HM-2	LM-1	N-3	N-4	N-5	N-6	AL-4	OBS
WEIGHT (kg):								
Air Temperature & Humidity Control	149	149	57	57	143	143	57	57
Equipment Heat Acquisition 2° C	355	355	0	104	68	68	0	104
21° C	390	390	0	191	191	191	0	191
Support Loops (2° C & 21° C)	81	81	0	0	0	146	46	0
TOTAL:	975	975	57	352	402	548	103	352
VOLUME (m^3):								
Air Temperature & Humidity Control	1.73	1.73	.40	.40	.71	.71	.40	.40
Equipment Heat Acquisition 2° C	.21	.21	0	.09	.07	.07	0	.09
21° C	.22	.22	0	.10	.10	.10	0	.10
Support Loops (2° C & 21° C)	.04	.04	0	0	0	.07	.02	0
TOTAL:	2.20	2.20	.40	.59	.88	.95	.42	.59
POWER (kW):								
Air Temperature & Humidity Control	.48	.48	.15	.15	1.04	1.04	.15	.15
Equipment Heat Acquisition 2° C	.32	.22	0	.07	.02	.02	0	.07
21° C	.29	.29	0	.18	.18	.18	0	.18
Support Loops (2° C & 21° C)	.03	.03	0	0	0	.06	.03	0
TOTAL:	1.13	1.13	.15	.40	1.26	1.30	.18	.40

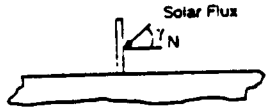
7: Transport and Rejection Summaries for the Initial and
h Phases at Lower Latitudinal Regions

Phase	Weight (kg)	Volume (m^3)	Power (kW)
nitial:			
Refrigeration	75	1	50.0
Transport	902	7	0.33
Rejection	3338	10	0
Total	4315	18	50.33
rowth:			
Refrigeration	150	2	100.0
Transport	2044	15	0.66
Rejection	6979	21	0
Total	9173	38	100.66

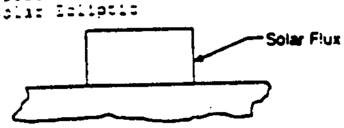
The rejection capability of the radiators was estimated using the following equation:

$$q = \epsilon \sigma (T_{rad}^4 - T_{sink}^4)$$

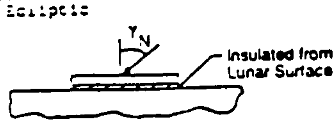
where q is the radiator heat rejection capability in W/m^2 and T_{sink} is the effective environmental temperature (K).



a) Vertical Two-Sided Radiator Perpendicular to Plane of Solar Ecliptic



b) Vertical Two-Sided Radiator Parallel to Plane of Solar Ecliptic



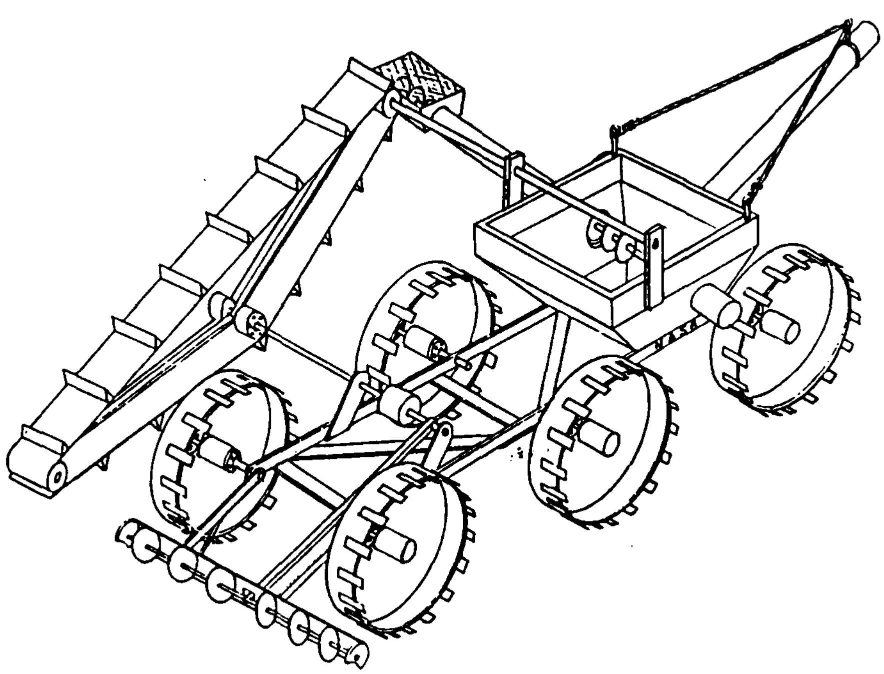
c) Horizontal One-Sided Radiator

Figure 5: Radiator Orientations

ID	Element Id	Element Type	Ref Design Id	# 25		
	bagger	misc				
	Regolith Bagger					
Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	250.0					
Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain
Production	Tech Level	# Uses	Op Life			
	2	da				

Reference Graf p9

Notes Receives soil, places it in bag, seals bag, and sets aside bag. Bag size about 0.3 m x 0.3 m x 1 m. Power should be low.



ID	Element Id	Element Type	Ref Design Id	# 26
	Pad Markers	misc		

Lunar Pad Markers

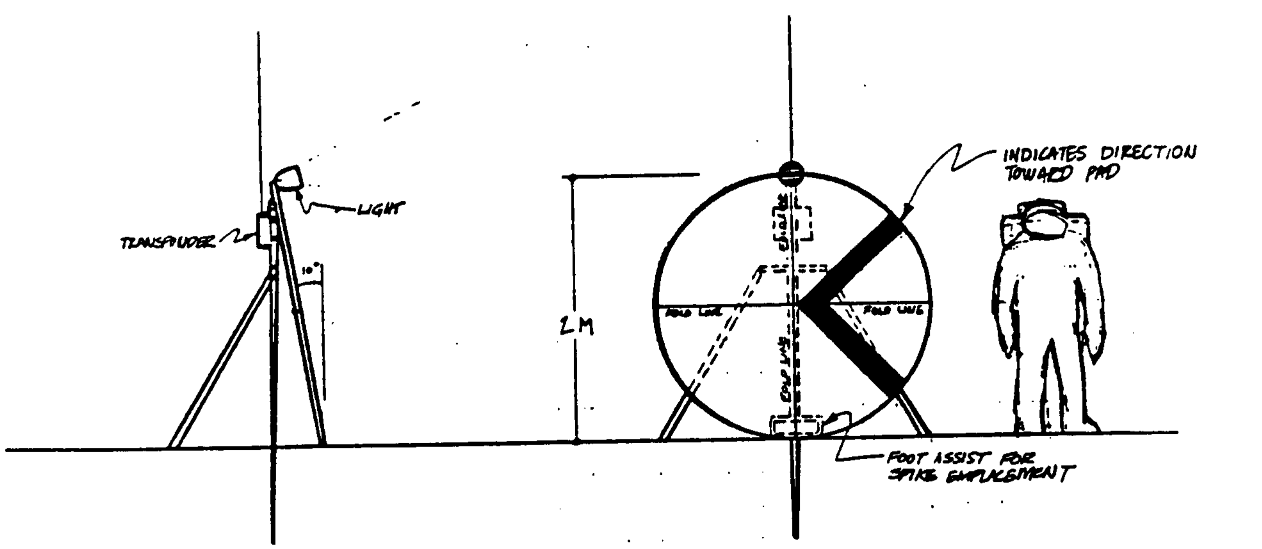
Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	10.0	0.50	0.50	0.10	0.03	

Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain
				2.0		

Production	Tech Level	# Uses	Op Life
	2		

Reference EEI-178 pp65ff

Notes Marker and Nav aid for lunar landing/launch facility. Contains transponder, visual marker, and light. Top has retroreflector for laser ranging. Power low since short range and operation time (5-10 min).



LANDING PAD MARKER

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ID — # 27

Element Id	Element Type	Ref Design Id
Plant-LLOX	resource	

Lunar Propellant Plant

Dimension

Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
45700.0					

Operation

Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain
			543.0		

Production

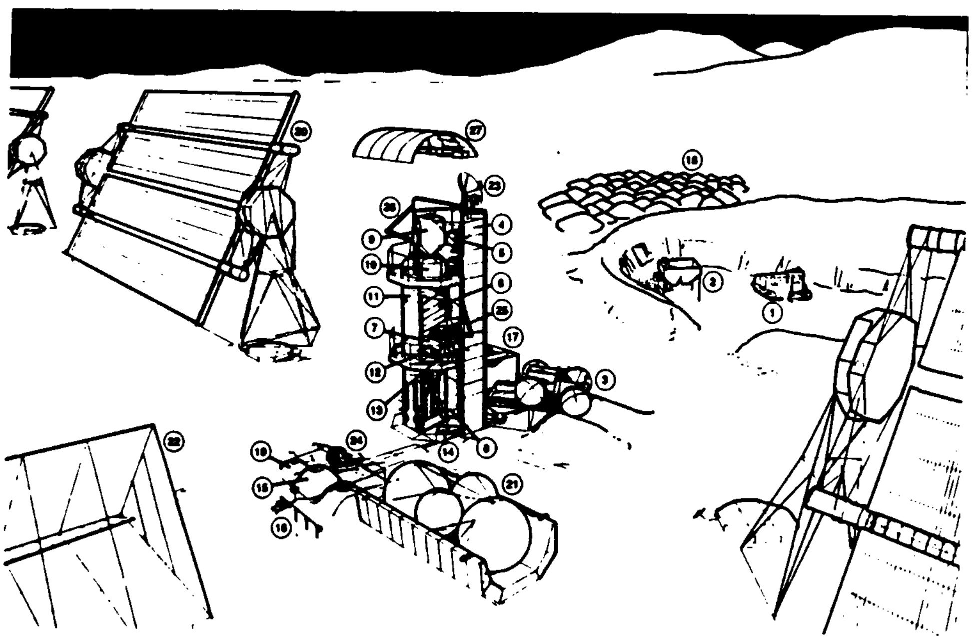
Tech Level	# Uses	Op Life
1-2		

Reference Christiansen

Notes Ilmenite reduction process plant producing 12.5t/mo (150t/yr) of LOX from soil feedstock. Uses nuclear power (included in mass). 90% duty cycle. Requires about 306 kg soil per kg of LOX. Mass represents a complete package: plant, power, mining, benefication, etc). Mining vehicle teleoperated from Earth.
For 200-1500t/mo class plants with nuclear power, 90% continuous duty, and soil feedstock, scaling is
Mass_in_t = 0.22 * Capacity_t_per_yr + 12.7.
Construction manpower estimated at 4 crew x 8 days x 8 hrs/day for each 22.4 t of plant mass.

Note: Picture depicts PV/RFC rather than nuclear power.

Conceptual Design Call-Out



Location: Lacus Veris (87.5°W, 13°S)
View Facing North-East

1. Excavator (Front-End Loader)
2. Pit Scalper
3. Hauler
4. Support Structure (Payload Bay Pallet)
5. 3-Stage Crushing/Grinding Circuit
6. Vibratory Screen (Fines Removal)
7. Hold-up Bin
8. High-Intensity Magnetic Separator
9. Low-Pressure Reactor Feed Hopper
10. High-Pressure Reactor Feed Hopper
11. 3-Stage Fluidized Bed Reactor
12. Electric Gas Heater
13. Solid-State Electrolysis Cell
14. Oxygen Liquefier
15. Buried Oxygen Storage Tanks
16. Liquid Oxygen Loading Station
17. Tails Discharge Bin
18. Tailings Piles
19. Makeup Hydrogen Storage Tank
20. Photovoltaic Power System (Sun-Tracking)
21. Regenerative Fuel-Cell Gaseous-Reactant Storage Tanks
22. Radiator with Fixed Sun-Screen
23. Communications: High- and Low-Gain Antennas
24. Telerobotic Servicer on Lunar Surface Mobile Platform
25. Telerobotic Servicer on Remote Manipulator Arm
26. Spare Remote Manipulator Arm
27. Equipment Repair and Spares Storage Shed

ID	Element Id	Element Type	Ref Design Id	# 28
	Pilot Plant-LLOX	resource		

Pilot Plant for Lunar Oxygen Production

Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	22.4					

Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain
				256.0		

Production	Tech Level	# Uses	Op Life
	1-2		

Reference Christiansen

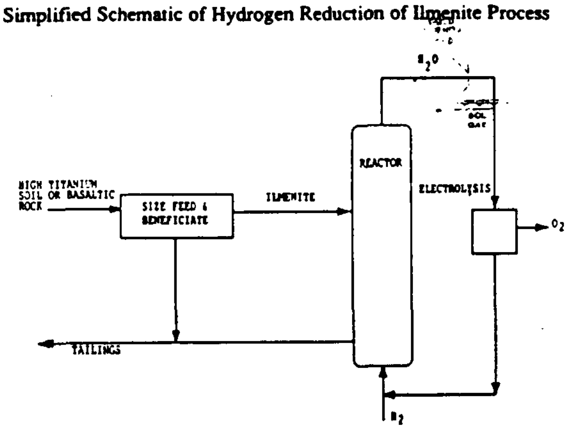
Notes Ilmenite reduction process plant producing 2t/month of LOX from soil feedstock. Power from PV arrays/RFC. Mining components have 35% duty cycle, operating during day with hot standby at night. Mass represents a complete package: plant, power, mining, benefication, etc). Mining vehicle teleoperated from Earth.

For 1-5t/mo class plants with PVA/RFC, 35% duty, and soil feedstock, scaling is

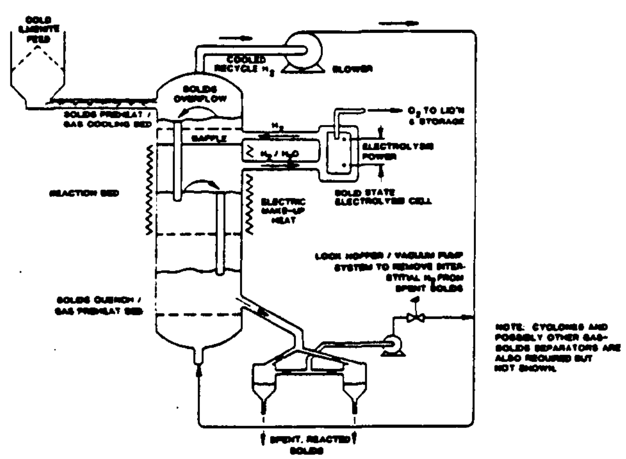
Mass_plant_in_t = 5.5 * Capacity_t_per_yr + 11.4.

Construction manpower estimated at 4 crew x 8 days x 8 hrs/day for each 22.4 t of plant mass.

Simplified Schematic of Hydrogen Reduction of Ilmenite Process



Three-Stage Fluidized Bed Reactor Concept for Ilmenite Reduction



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ID Element Id: Plant-MProp | Element Type: resource | Ref Design Id: | # 29

Mars Propellant Plant

Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	80000.0					

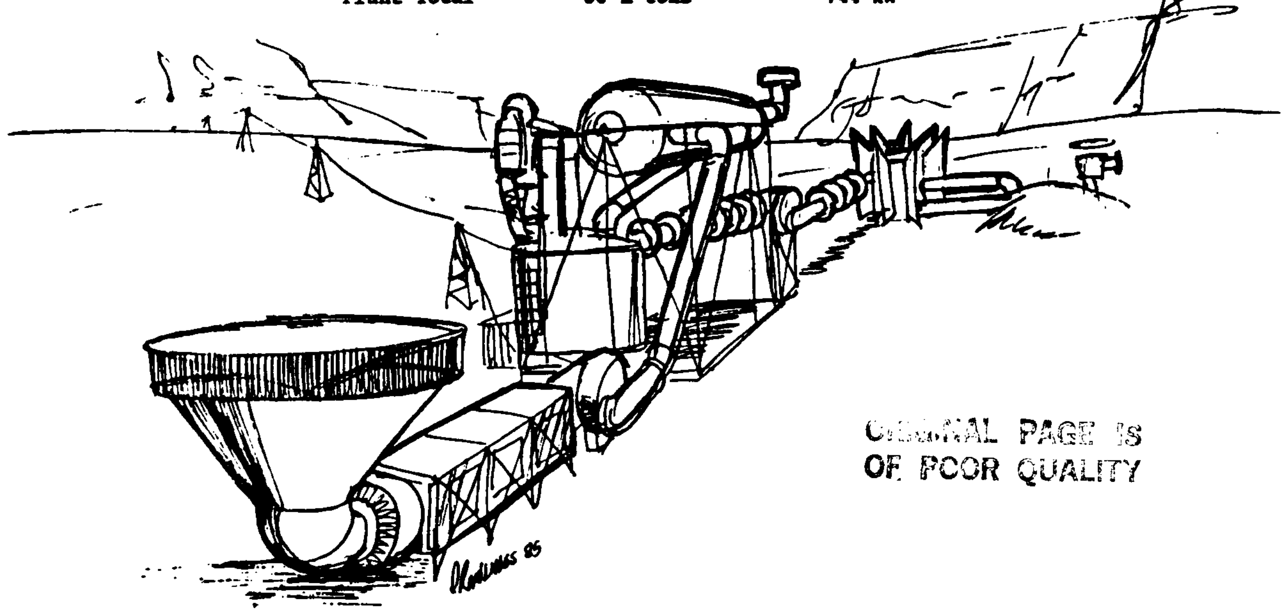
Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain
	744.0					

Production	Tech Level	# Uses	Op Life
	1-2		

Reference EEI-85-109B p6-13

Notes 300 t/year of LOX from CO2 in atmosphere. Mass for complete package (includes 20t for nuclear power). Produces O2 from CO2 by an electrolytic process. Blower forces air through filter tor remove particulates. Gas compressed and preheated to 950° K, then enters electrolytic unit that operates at 1273° K. CO2 dissociaties into O2 and CO and membranes isolate O2. Exhaust gas preheats inlet before it is vented. Liquefied O2 stored in buried tanks.

UNIT	MASS, m tons	POWER, kW
Filter	1	
Blower	1	10
Compressors (2)	19	74
Heat Exchanger	4	
Electrolytic Unit	14	600
Radiators (120 m^2)	1	
Cryogenic Unit	9	60
Piping & Insulation	8	
Tanks (3.6 m dia)	3	
Plant Subtotal	60	
Power	20	
Plant Total	80 m tons	744 kW



ID	Element Id	Element Type	Ref Design Id	# 30
	Plant - PhProp	resource		

Phobos Propellant Plant

Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	107900.0					

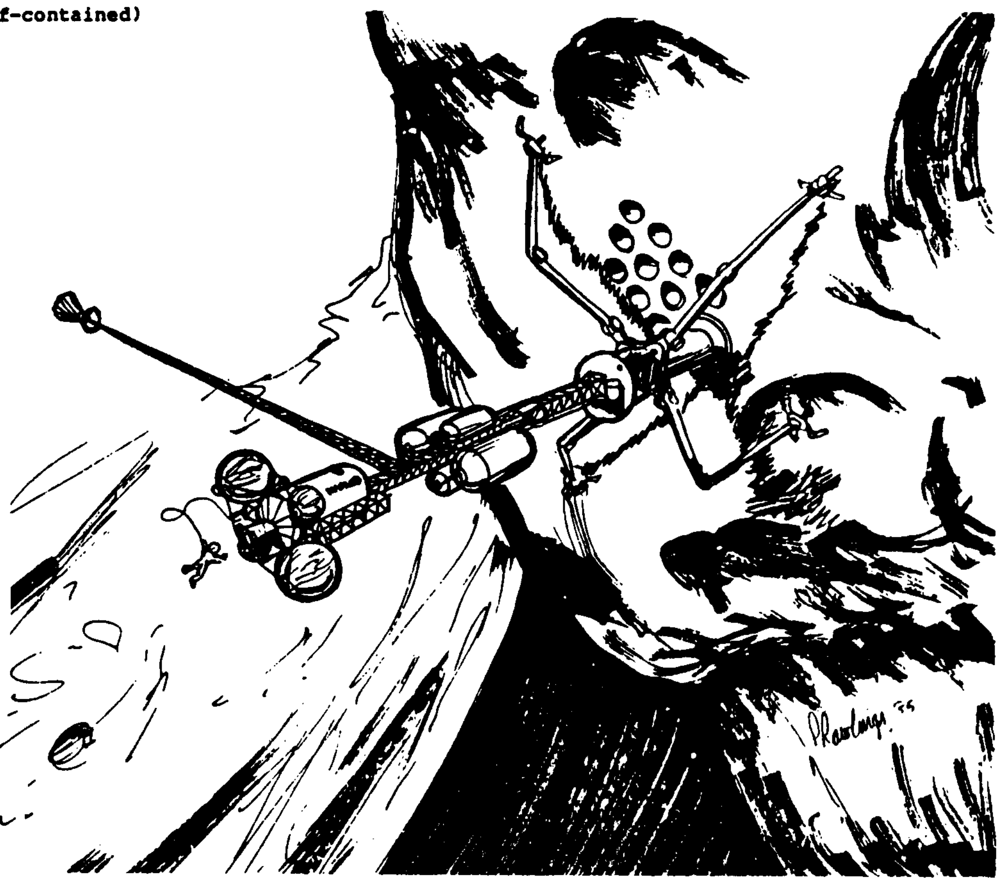
Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain
	1067.0					

Production	Tech Level	# Uses	Op Life
	1-2		

Reference EEI-85-109B p6-7ff

Notes 600 t/yr of water (533 LOX, 67 LH2) from Phobos rock and soil. Includes complete package (plant, power, tankage, crew quarters, etc). Assumes regolith is 5% water. Based on rock-penetrator developed at Las Alamos. Uses rock-melter as coring device. Impermeable glasslike lining forms and seals volatiles in borehole. Subsequent processing removes impurities (e.g., CO, CO2, HS) and dissociates water by electrolysis. O2 and H2 then liquefied and strored. Moves to new boresites via legs with end-effectors after rasing plant with hydraulic jacks.

UNIT	MASS, m tons	POWER, kW
Penetrator	4.9	(self-contained)
Radiation Shield	11.8	
Filters (2)	4.3	
Condensors (2)	3.1	
Electrolytic Unit	2.7	335
Cryogenic Unit	1.5	450
Habitat (& ECLSS)	19.8	36
Crew & equipment	1.3	
Boom & structure	2.0	
Hydraulics & legs	1.0	
Tanks (4)	8.9	
Radiators (300 m^2)	6.1	
Plant Subtotal	**67.4**	**821**
30% contingency	20.2	246
Nuclear reactor	2.4	
Power converter	2.3	
Shield (man-rated)	12.0	
50 m boom	0.7	
Radiator	2.9	
Power total	**20.3**	
Plant Total	**107.9**	**1067**



ID	Element Id	Element Type	Ref Design Id	# 32
	NR-1MW est	power		

Nuclear Reactor- 1MW class SP-100 Derivative Estimates

Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	24000.0					

Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain
		1000.0				

Production	Tech Level	# Uses	Op Life
	1-2	1	

Reference ASAO

Notes Rough characterization of 1 MW class SP-100 type reactor. Fuel is U235 with trace U234. Assumes surface material used as shield. With 4 pi, man-rated shield from Earth, total mass is 25.7 t.
Estimate based on following data.

Power (kW)	Specific Mass (kg/kW) (native shield / Earth shield)
100	40 / 119
500	24 / 41
2000	12.5 / 18

ID	Element Id	Element Type	Ref Design Id	# 33
	PVC-aSi	power		

Photovoltaic Cells, thin film amorphous silicon

Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	500.0					

Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain
	50.0					

Production	Tech Level	# Uses	Op Life
	3-4		

Reference Brinkler[88]

Notes Thin film amorphous silicon solar blankets with total output of 50kWe, specific power 1kW/kg. Large area cell on flexible substrate. Deployment involves unrolling array. May loose about 1% efficiency per year after deployment but output levels should eventually flatten out. Very resistant to radiation (better than GaAs and InP). Space applicability yet to be demonstrated. Technology estimates are

Tech Level	Year	W/kg	W/sq m	Efficiency (%)
demo	1988	120	60	6
		200	110	10
4	1990	600		
3	1993	600		

ID	Element Id	Element Type	Ref Design Id	# 34
	RFC	power		

Advanced Regenerative Fuel Cells

	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
Dimension	17000.0					

	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain
Operation	50.0					

	Tech Level	# Uses	Op Life
Production	4-7		

Reference Prokopius[88]

Notes Advanced cells including tanks, reactants, fuel cell, electrolyzer stacks, and radiator. Sized for lunar application with 50 kW level during lunar night (16800 kWh = 50 kW x 336 h), EEF = 65%. Scaled from Prokopius data on 25 kW RFC.

Spefic Storage (Wh/kg)

	Storage(h)	SOA (55% EEF)	Near term (<5yr) (55% EEF)	Advanced (8 yr away) (65% EEF)
Mars	12	100	300	450
Moon	336	200	550	1000

ID	Element Id	Element Type	Ref Design Id	# 35
	PVA/RFC-L50kW (est)	power		

Lunar 50 kW Power System, Photovoltaic Array with Regenerative Fuel Cells

Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	15000.0					

Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain
	50.0					

Production	Tech Level	# Uses	Op Life
	4-7		

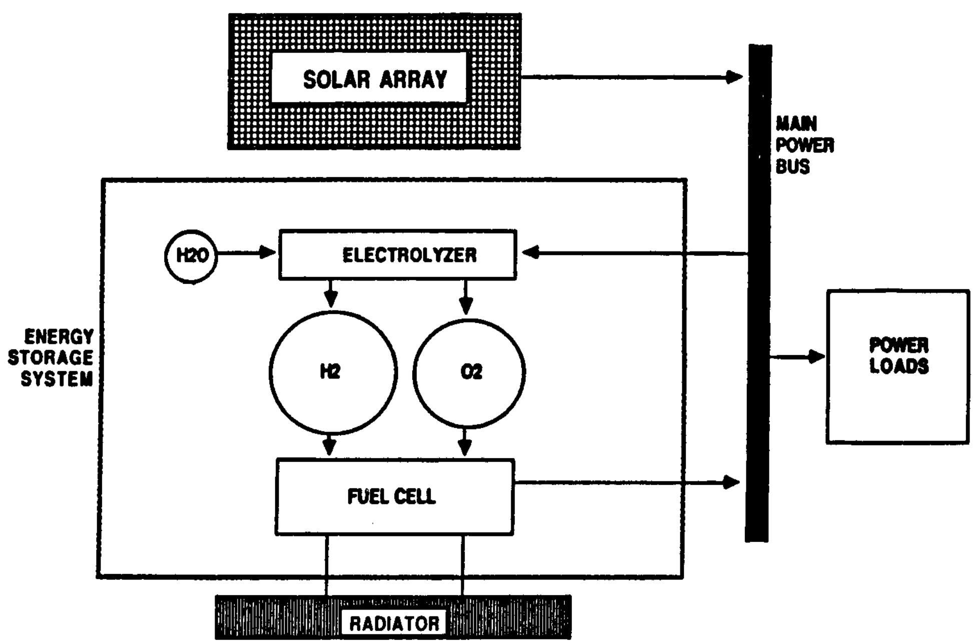
Reference Prokopius[88]

Notes

Rough estimate of complete system scaled from data in Prokopius for 25 kW class:

25 kW class System	W/kg 2wk Lunar	W/kg 12hr Mars
Adv Solar Dynamic/RFC	2-4	2-4
PV Array/RFC	2-4	15- 20
PV Array/Battery	0.1-0.5	5-8
Primary Fuel Cell	2-4	1-2
Nuclear (SP-100 Deriv)	7-15	7-15

REGENERATIVE FUEL CELL SYSTEM SCHEMATIC



ID	Element Id	Element Type	Ref Design Id	# 36
	Pwr Cable	power		

Power Transmission Cable, 1 km distance, 2 x 4/0 Al

Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	500.0			1000.00		

Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain

Production	Tech Level	# Uses	Op Life
	6		

Reference Engineering Handbook

Notes Rough estimate for two strands of 4/0 gauge Al cable. Cable gauge based on following assumptions: 80% pfa, 5% power loss, distances are 10's of km. These indicate about 3/0 Cu. Weights for spools, layers, etc, are not determined.

4/0 Al properties: Equivalent to 2/0 Cu; mass = 198.6 lb/ 1000 ft; wires = 19 x 0.1055" diameter, 7 x 0.1739" diameter; conductor diameter = 0.528" (19 wire), 0.522" (7 wire); DC resistance at 20° C = 0.08196 Ohm/1000 ft.

ID	Element Id	Element Type	Ref Design Id	# 37
	NR-825 kWe Stirling	power		

Nuclear Reactor - 825 kWe Stirling System

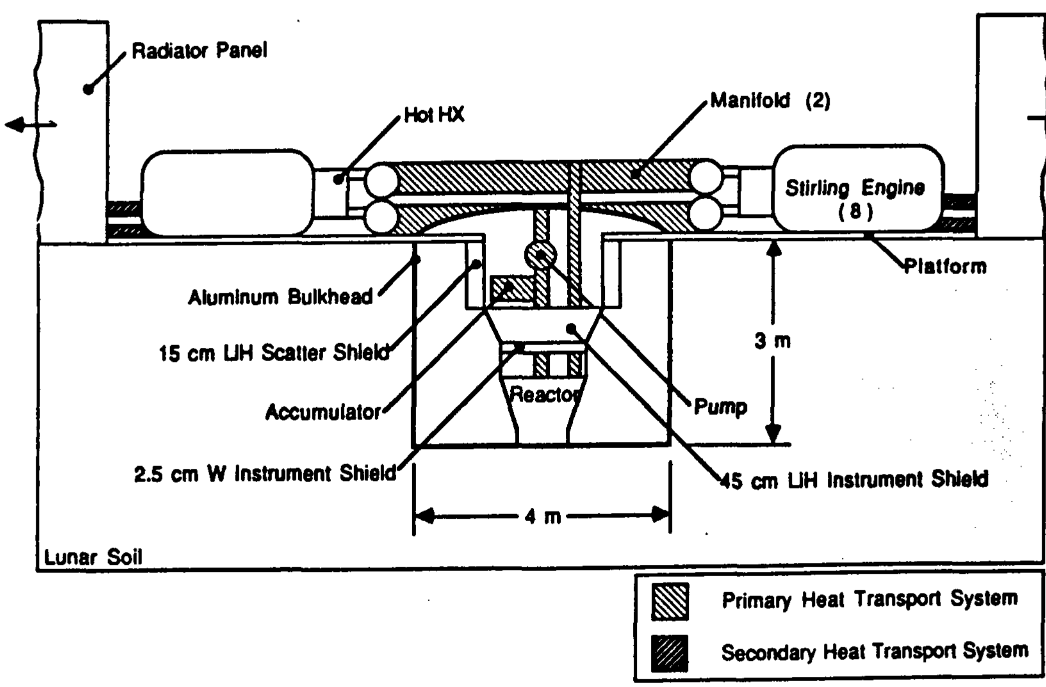
Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	20004.0					

Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain
	825.0					

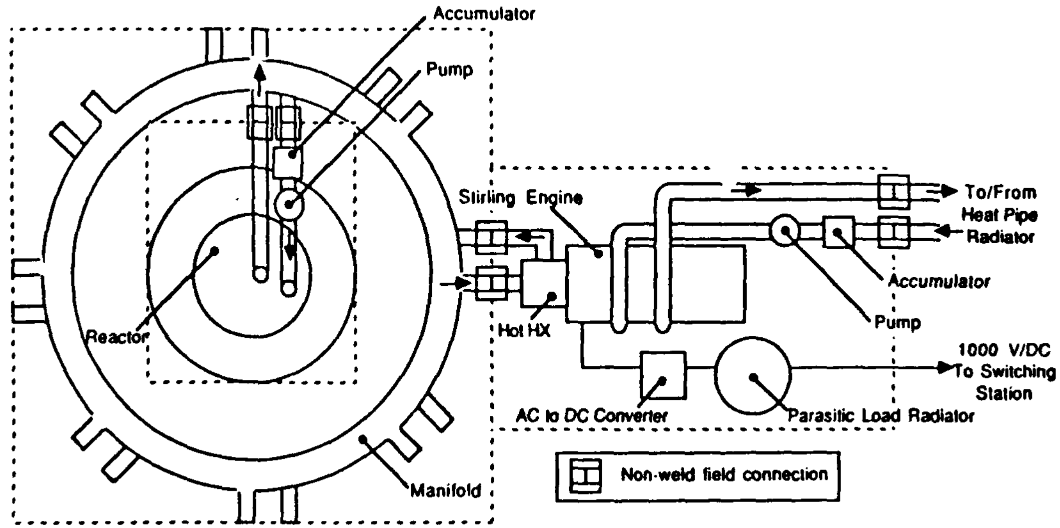
Production	Tech Level	# Uses	Op Life
	1-2		

Reference Mason[88]

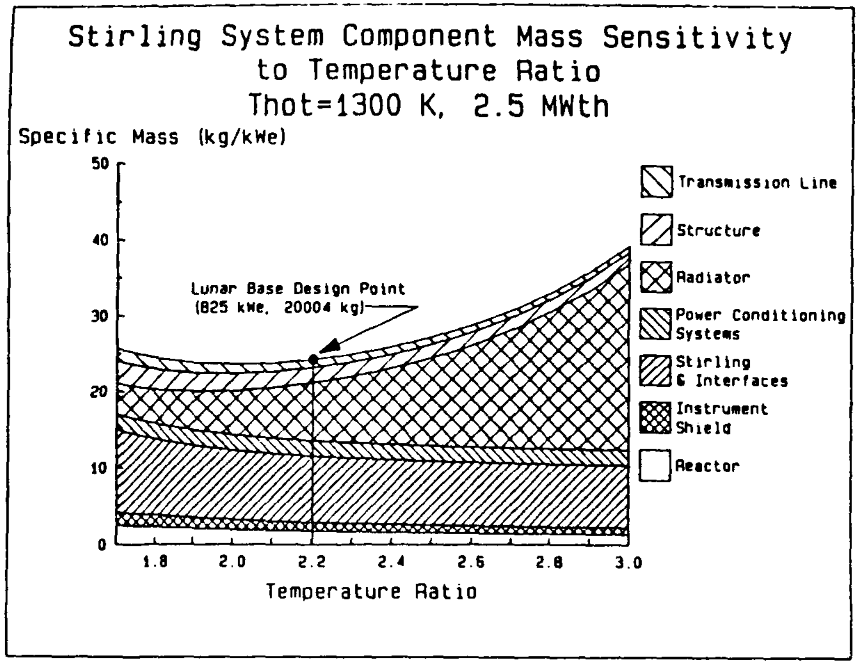
Notes SP-100 derivative reactor using 8 Stirling engines (6 @ 91.7% capacity, 2 reserve) for power generation. Reactor buried in cavity with aluminum bulkhead for dust isolation. Regolith provides shielding, reducing system mass, and permitting flexibility of placement and crew access to radiators. Dual manifold heat transport system designed for easy installation of Stirling engines at non-weld field connections. Area efficient vertical spoked radiator panels. Independent heat rejection loops for each engine simplify assembly and construction. Heat pipe radiator offers built-in redundancy. Apron designed to reduce surface temperatures from 375°K to 222°K.



Reactor Thermal Power	2500	kWt
Reactor Design Lifetime (@ full power)	7	yrs
Electrical Output (6 of 8 Engines)	825	kWe
Electrical Output/Operating Engine	138	kWe
Rated Electrical Output/Engine	150	kWe
Percent of Operating Engine Capacity	91.7	%
Thermal-to-Electric Efficiency	33.0	%
Stirling Heater Temperature	1300	K
Stirling Temperature Ratio	2.2	
Stirling Cooler Temperature	591	K
Radiator Surface Temperature	525	K
Total Heat Rejected	1675	kWt
Lunar Surface Temp. (w/Apron)	222	K
Lunar Sky Temperature	267	K
Radiator Emissivity	0.85	
Radiator Area (Spoked Wheel)	780	m2



Subsystem	Mass (kg)	Material
Reactor	755	Refractory Metal
Primary Heat Transport	342	Refractory Metal
Instrumentation & Control	359	
Shadow Shield	931	LiH/W
Stirling Engines (8; incl HXs)	5871	Refractory Metal
Reactor/Stirling Manifold (Dual)	423	Refractory Metal
Heat Rejection Loops (8)	648	Stainless Steel
EM Pumps (8)	160	Hiperco-27
Accumulators (8)	24	Refractory Metal
Radiator (Spoked Wheel)	6240	Stainless Steel/ Mercury Heat Pipes
Structure		
Reactor Excavation Bulkhead	679	Aluminum
Engine Support Platforms (8)	1005	Carbon-Carbon
AC-DC Converter (& PLR)	1650	
Transmission Lines (5051 m)	917	
Total	20004	kg
Total System Specific Mass	24.2	kg/kWe



ID	Element Id	Element Type	Ref Design Id	# 38
	PV/RFC-L50kW	power		

Photovoltaic/Regenerative Fuel Cell Lunar Power Supply - 50 kWe

Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	19744.0					

Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain
	50.0					

Production	Tech Level	# Uses	Op Life
	1-2		

Reference Mason[88]

Notes

- Amorphous Silicon Arrays - Technology Goals
  - 15% cell efficiency
  - 90% packing factor
  - 300 W/kg specific mass (cells, kapton substrate, and interconnect wiring)
  - Roll-Up storage capability
- Regenerative Fuel Cells - Technology Goals
  - 60% round trip efficiency
  - 1000 W-hr/kg energy density
  - High pressure filament wound storage tanks
- Power Management and Distribution
  - 92% Power Conditioning/Distribution efficiency
  - 20 kg/kWe specific mass

Power (kWe)	Array Area (m2)	Array Mass (kg)	100% Night Power RFC Mass (kg)*	Total Mass (kg)
25	397.6	241.5	9130	9872
50	795.2	483.1	18261	19744
75	1192.8	724.6	27391	29616
100	1590.4	966.2	36522	39488

* 336 Hour Lunar Night

ID	Element Id	Element Type	Ref Design Id	# 39
	TCS-L inf	env	

Lunar Thermal Control System - Inflatable Habitat

Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	6122.0				21.80	

Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain
	40.0					

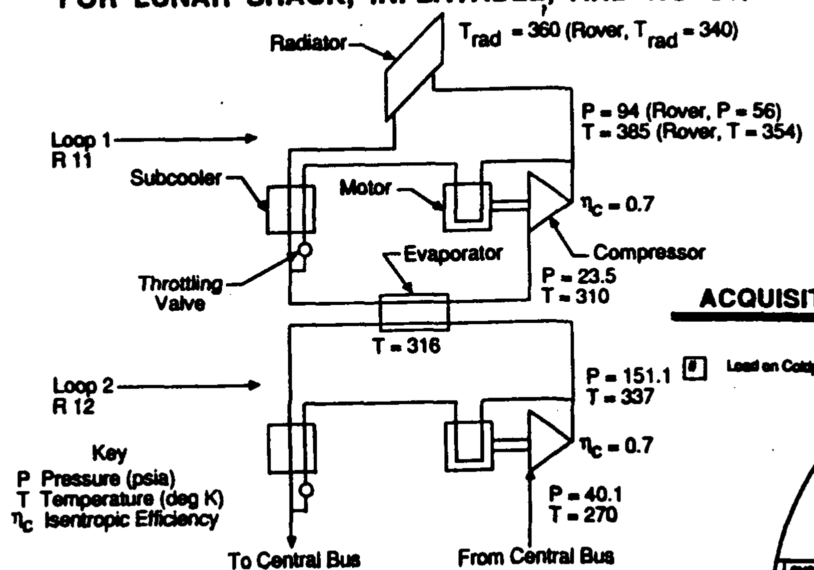
Production	Tech Level	# Uses	Op Life
	2-3		

Reference Simonsen[9/88]

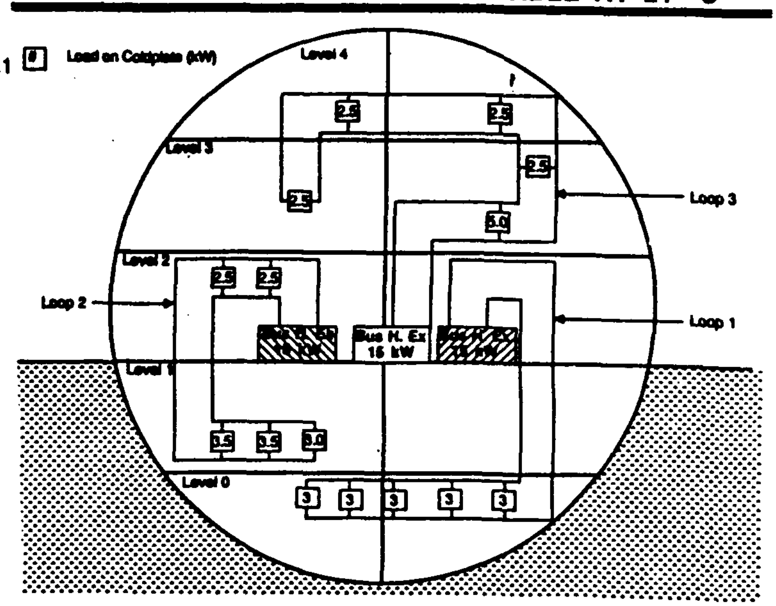
Notes Two stage Vapor Cycle Refrigeration System for Lunar Shack, Inflatable, and Rover. Acquisition: Single phase pumped water loops at 2° C and 21° C. Transport: Two-phase freon 12 at -3° C. Rejection: Two stage vapor cycle refrigeration system using SS ammonia heat pipe radiators. Sizing estimates:

Structure	Wt (lb)	Vol (cu ft)	Power (kW)	Load (kW)
Shack	4,945	300	15.96	25
Inflatable	13,514	767	39.94	65
Rover	2,685	172	6.20	9

TWO-STAGE VAPOR CYCLE REFRIGERATION SYSTEM
FOR LUNAR SHACK, INFLATABLE, AND ROVER



ACQUISITION LAYOUT FOR INFLATABLE AT 21° C



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ID	Element Id	Element Type	Ref Design Id	# 40
	ECLSS - L inf6	env		

Envionmental Control and Life Support System for Lunar Inflatable - 6 crew evolving to 12 crew

Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	16820.0				49.30	

Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain
	8.5					

Production	Tech Level	# Uses	Op Life
	2-3		

Reference Hypes[9/88]

Notes Uses 6-crew subsystems: compatible with mission models, practicle size, and growth compatible. Use LOX after available (assumed MM9 in model). Integrates with rover ECLS. SS and new technology. System for 6 crew for 42 day mission. Assumes use of lunar Shack for safe haven. Evolution to 12 crew/70 days summarized in tables. Evolution mission model as follows (crew/days): Lunar shack used in MM1-5, Inflatable used from MM6 on with shack as safe haven; MM1-3 4 /10, MM4 4/30, MM5 4/42, MM-6 4/42 (start using inflatable), MM7 6/42, MM8-9 6/70, MM10+ 12/70.

ECLS Function	MM-6 Crew 4	MM-7 Crew 6	MM-8 Crew 6	MM-9 Crew 6	MM-10 Crew 12	MM-11 thru 15 Crew 12
Air revitalization						
Technique for:						
• CO_2 removal	EDC	→	→	→	→	→
• CO_2 reduction	Bosch	→	→	→|	(N/A	→)
• O_2 supply	Cryogenic	Same	Same	Same	Same	Same
	SFWE	→	→	→|	(N/A	→)
• N_2 supply	Cryogenic	Same	Same	Same	Same	Same
• O_2/N_2 pres. cont. • Vent & relief • Fire Det. & supp. • Contaminant mon.	Space station Inheritance	→	→	→	→	→
• Contaminant cont.	Catalytic oxidizer	→	→	→	→	→
• Emergency press.	Gas storage	→	→	→|	Gas storage	→
Water Management						
Source of:						
• Drink & food prep.	Stored potable	Same	Same	Same	Same	Same
	MF-RO/cond.	MF-RO/cond. & excess		→	+ units 3 & 4	→
• Hygiene	Times/hygiene & urine	→	→	→	+ units 2	→
Waste Management						
Approach to:						
• Fecal & wipes	Shredder-dryer & storage (2)	→	→	→	+ units 3 & 4	→
• Trash	Compaction & Storage	→	→	→	+ units 2	→
Habitability						
Technique for:						
• Body cleaning	Shower (2)	→	→	→	+ units 3 &4	→
• Area cleaning	Vacuum & wipes	→	→	→	→	→
• Towels, linens, etc.	Disposable	Same	Same	Same	Same	Same
Food & Feeding						
Provisions for:						
• Food	Freeze dried &Frozen	Same	Same	Same	Same	Same
• Food lockers	Space station inheritance	→	→	→	+ units 2	→
• Storage & prep.	Refrig - freezer & galley	→	→	→	+ units 2	→
• Utensils	Disposable	Same	Same	Same	Same	Same
Clothing:	Disposable	Same	Same	Same	Same	Same
Provisions for:						
EVA						
Provisions for:						
• Suits						
• Env. control	PLSS units (6)	→	→	→	+ units 7 &12	→
Microbiological cont.						
Technique for:						
• Decontamination	Steam sterilizer	→	→	→	→	→

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JSC/LaRC LUNAR BASE STUDY

Guidelines (G) and Assumptions (A)

• General	
Manned missions	Max. 6 months apart (G)
Base LOX production	Begin MM - 9 (G)
Base LOX for ECLS	Begin MM - 10 (A)
Shack	Safe haven for habitat (G)
• Structure Volumes	
Habitat	77,684 ft^3 (G)
Shack	4,1[illegible] ft^3 (A)
Habitat to shack tunnel	609.5 ft^3 (A)
Habitat airlock	1,538.6 ft^3 (A)
Shack airlock	1,538.6 ft^3 (A)
• Repressurizations	
Shack	1/mission (A)
Habitat	2/total habitat life (G)
Tunnel	1/mission (A)
• Leak Rates	
Habitat	5 lb gas/24-hours (A)
Shack	1 lb gas/24-hours (A)
Tunnel	1/4 lb gas/24-hours (A)
• Airlock Operation	
Habitat	12/7 days MM 6-7 (G) 12/10 days MM 8-15
Shack	12/10 days (A)
Airlock Loss	10% volume/operation (A)

JSC/LaRC LUNAR BASE STUDY
ECLS SYSTEM SUMMARY

Mission	Weight (pounds) New	Weight (pounds) Resupply	Weight (pounds) Total	Volume (ft.^{3}) New	Volume (ft.^{3}) Resupply	Volume (ft.^{3}) Total	Power (kw)
MM-1	6,178	---	6,178	513.6	---	513.6	4.21
MM-2	2,428	65	2,493	77.6	3.3	80.9	4.21
MM-3	2,428	65	2,493	77.6	3.3	80.9	4.21
MM-4	4,238	158	4,396	226.8	7.9	234.7	4.21
MM-5 (Safe Haven)*	5,227 (579)	215	5,742	250.8 (52.3)	10.7	261.5	4.21
MM-6 (Safe Haven)	32,212	---	32,212	1,558.0	---	1,558.0	8.15 (.30)
MM-7 (Safe Haven)	4,917	363	5,280	182.2	26.0	208.2	8.46 (.30)
MM-8 (Safe Haven)	5,731	587	6,318	419.5	36.9	380.2	8.46 (.30)
MM-9 (Safe Haven)	8,258	645	8,903	611.2	41.1	652.3	8.46 (.30)
MM-10 (Safe Haven)	22,881	1,151	24,032	1,251.6	48.6	1,300.2	9.93 (.30)
MM-11 - 15 (Safe Haven)	8,683	797	9,480	721.2	49.1	770.3	9.93 (.30)

*(Safe Haven) Weight, Volume, and Power values must be added to the basic totals to keep the Shack as a Safe Haven

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ID	Element Id	Element Type	Ref Design Id	# 41
	ECLSS-Lshack 4/10	env		

Environmental control and Life Support System for Lunar Shack with 4 crew for 10 days

Dimension	Mass-kg	Height-m	Width	Length	Volume-m3	Stowed Vol
	2800.0				14.54	

Operation	Avg Power kW	Peak Power	Min Standby	EVA Deploy mh	EVA Maintain	IVA Maintain
	4.2					

Production	Tech Level	# Uses	Op Life
	2-3		

Reference Hypes[9/88]

Notes Simple, reliable, with little crew involvement. Minimum impact at revisit. Adaptable as safe haven. Current or SS technology. Resupply summarized below for same mission model as in inflatable: MM1-3 4 /10, MM4 4/30, MM5 4/42.

ECLS Function	MM-1 Crew 4	MM-2 Crew 4	MM-3 Crew 4	MM-4 Crew 4	MM-5 Crew 4
Air revitalization					
Technique for:					
• CO_2 removal	Molecular sieve →				
• CO_2 reduction	(N/A)				
• O_2 supply	Cryogenic	Same	Same	Same	Same
• N_2 supply	Cryogenic	Same	Same	Same	Same
• O_2/N_2 pres. cont.					
• Vent & relief	Space station →				
• Fire Det. & supp.	Inheritance				
• Contaminant mon.					
• Contaminant cont.	Catalytic oxidizer →				
• Emergency press.	Cryogenics →				
Water Management					
Source of:					
• Drink & food prep.	Stored potable	Same	Same	Same	Same
• Hygiene	MF-RO/cond. →				
Waste Management					
Approach to:					
• Fecal & wipes	Shredder-dryer & storage →				
• Trash	Compaction & Storage →				
Habitability					
Technique for:					
• Body cleaning	Shower →				
• Area cleaning	Vacuum & wipes →				
• Towels, linens, etc.	Disposable	Same	Same	Same	Same
Food & Feeding					
Provisions for:					
• Food	Freeze dried &Frozen	Same	Same	Same	Same
• Food lockers	Space station inheritance →				
• Storage & prep.	Refrig - freezer & galley →				
• Utensils	Disposable	Same	Same	Same	Same
Clothing:	Disposable	Same	Same	Same	Same
Provisions for:					
EVA					
Provisions for:					
• Suits					
• Env. control	PLSS units (4) —	+ units 5 —	+ units 6 →		
Microbiological cont.					
Technique for:					
• Decontamination	Steam sterilizer →				

PHOBOS EXPLORATION ASSESSMENT

This assessment began in June, 1988. Its objectives were to bring a team together to effect the assessment, define the trade space to be used in succeeding study tasks, begin the definition of various study requirements to be used as input to the studies, to define some preliminary studies that could be started in FY88 and to define future studies for FY89. A team was assembled and began to fulfill the objectives. The team has provided insight into the numerous areas needing study. Figure 1.3.2-1 describes the trade space.

Two study tasks were defined for FY88 which would provide a preliminary look at the problem of 1) anchoring to the surface and 2) an assessment of flight over the surface. Study requirements are still being gathered. FY89 tasks will be continuations of those previously defined. Additional studies will need to be incorporated to adequately cover the Case 2 and the Case 4 activities. Work is continuing.

FY88 Task A: Preliminary Study of Surface Anchoring Methods.

Objective

Investigate and determine candidate methods of surface anchoring.

Methodology

Background. The gravity force is 2,000 to 3,000 times less than on the Earth. Inadvertant flight above the surface for extended periods of time is highly likely. Anchoring methods to maintain surface contact and to create a stable work platform are needed.

Key Assumptions. The study should consider anchoring an EMU clad crew member and a modified EMU/MMU vehicle/crew combination. Surface characteristics are not well known. Anchoring entails physical contact and attachment to the surface soil or rock. Although tethers with a single-end attachment point will not be considered, movement along tension lines or wires with end-point attachments to stable platforms/pins will be considered.

Approach. Investigate existing Earth-based anchoring systems. Develop requirements for anchoring hardware while using inputs of the Phobian surface characteristics, its gravity force, sampling view/reach requirements, safety considerations, etc. This information will be obtained from existing reports, the Science community and other applicable experts. Figure 1.3.2-2 is an example of an anchoring device.

Activities & Results

Due to lack of resources, activity has been limited to work on a task plan. Some initial contacts were made with Oil & Gas industry representatives with a limited response. Figure 1.3.2-3 shows a pipeline stabilizing anchor and Table 1.3.2-1 shows some typical uses. As indicated, the anchor appears applicable to large pipes (32 inches diameter) or small pipes (4 inches diameter). It is also applicable to a) severe wave action and current in a surf zone, b) 8 fps current, c) gravel bottom, d) granite bottom and e) soft soil.

FY88 Task B: Preliminary Study of Flight over Phobian Surface.

Objective

Examine trajectory plans for Phobos operations and determine delta velocity requirements for flight activities on or near the surface.

Methodology

Background. Scientific and exploration objectives will require widespread movement about the surface. Flight over the surface gains the ability to alit 'anywhere', to avoid obstacles and to do it quickly. This will be countered by the need to carry propellant and GN&C flight systems.

Key Assumptions. Both the gravity potential models of Mars and Phobos must be included. This study will deal only with delta velocity usage (and indirectly, propellant usage) and trajectory motions over the surface.

Approach. Develop a software targeting simulation. Begin with a simple system evolving to a man-in-the-loop simulator with surface characteristics. Investigate various flight trajectories over the surface. These will include both short roundtrip traverses of less than 2 kilometers and long roundtrip traverses of up to 10 kilometers.

Figure 1.3.2-4 Flying Over Phobos

Activities & Results

See summary of activities and user requirements.

FIGURE 1.3.2-1

TRADE SPACE DEFINITION

PHOBOS/DEIMOS SURFACE EXPLORATION

What jobs?	Transit method?	Vehicles & Equipment?	Comm? GN&C?	Habitat?	Logistics & resupply?
Sample collection & packaging	Orbiter flyaround	Mars orbiter	Local/global comm	Whichever large vehicle is at Phobos	Self-contained but can be replenished from large vehicle at Phobos
Still photography	Phobos Sortie vehicle flyaround	Phobos Sortie vehicle	Local/global data transfer		
Movie film and Video	EMU/MMU flyaround	Enhanced MMU	Relative & global navigation	Surface anchored and stationkeeping flight	
Visit Multiple sites	*Surface* anchoring	EVA retriever	Guidance & targeting algorithms		
Package emplacement	Rovers (Teleop/auto)	Rovers (Mars/Phobos)			
Visual observations	Mountain climber EVA	Enhanced EMU			
Sample inspection		Anchors & penetrators			
Onboard sample analysis		Emplacement packages			
Map preparation		Photographic devices			
Mars observation		Sampling Tools			
Mars' rover teleoperation		Insitu analysis tools			
		Dust protection			

Items considered in FY88 tasks.

NOTE: This table presents elements to be considered as a part of each of the heading questions. No attempt has been made to identify horizontal correlations and none are implied.

FIGURE 1.3.2-2
SAMPLE ANCHORING DEVICE

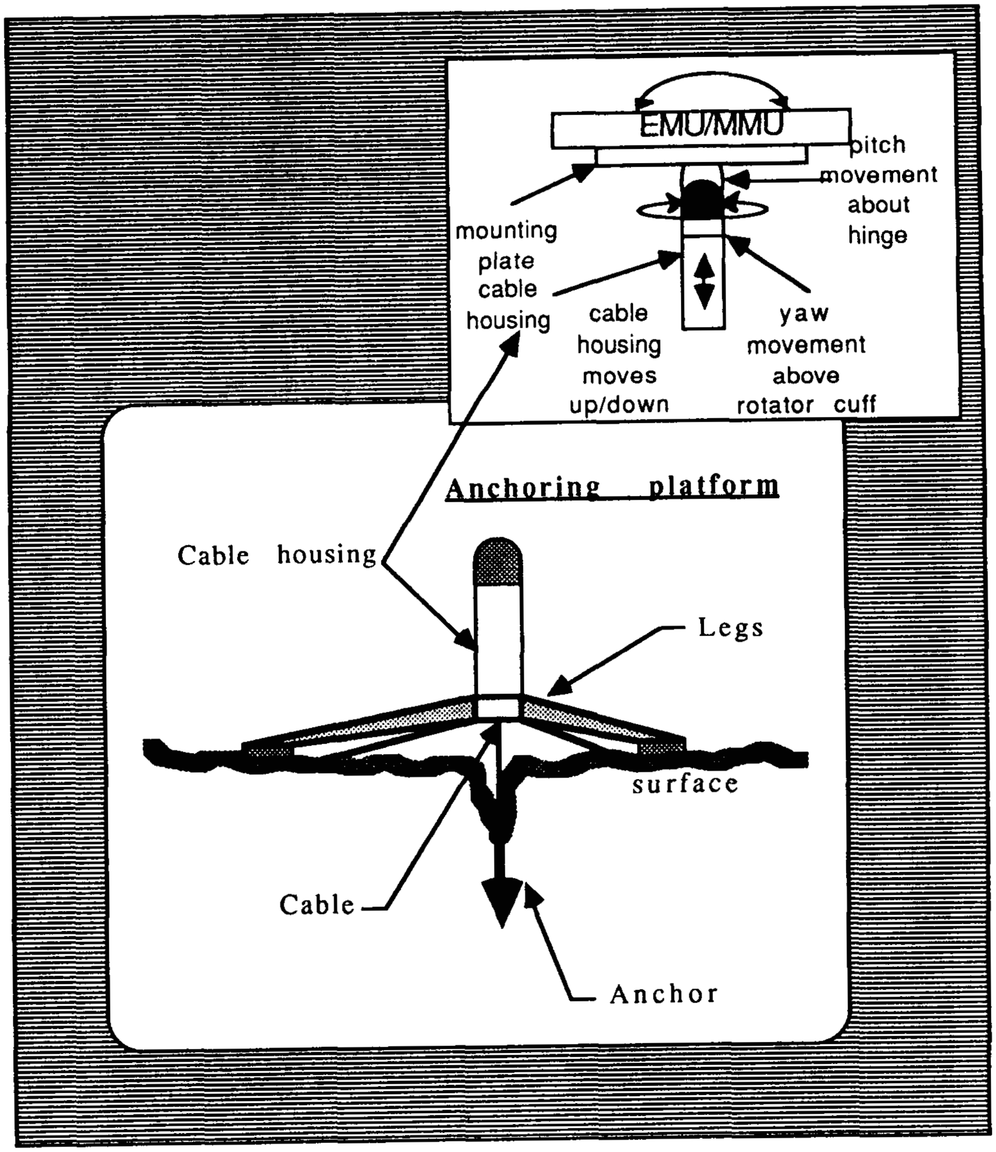


FIGURE 1.3.2-3
SAMPLE EARTH-BASED ANCHOR

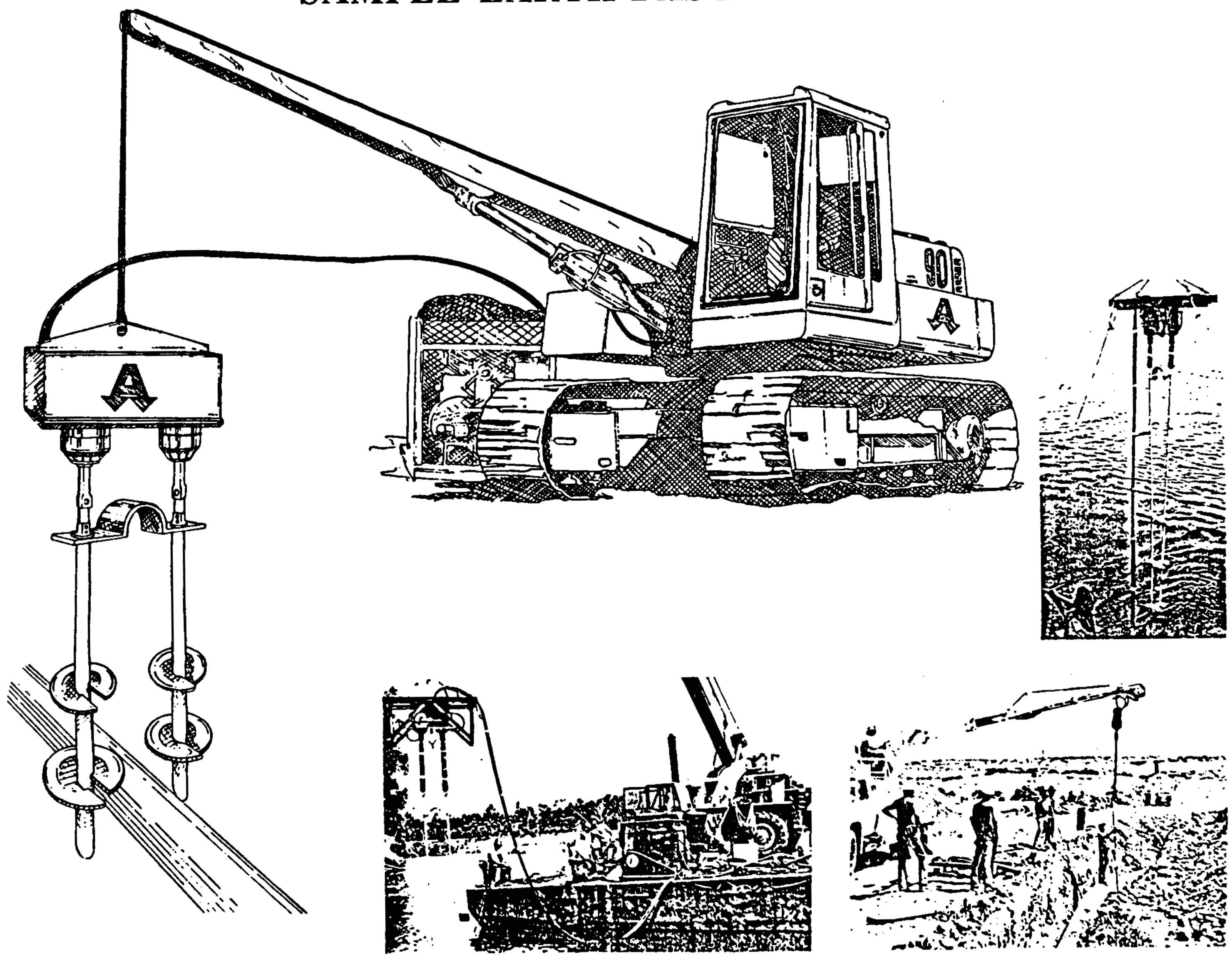


TABLE 1.3.2-1

SAMPLE EARTH-BASED ANCHOR SITES

Recent Case Histories

OFFSHORE

Company	Location	Pipe OD	Depth	Comments
Shell	Brent Field, North Sea	24"	548 ft.	Deepest anchoring project.
Zaire Gulf Oil	Coast of Zaire	8"	15 ft.	' Severe wave action and current in a surf zone.
Gulf Oil Nigeria	Coast of Nigeria	4"-6"	15 ft.	Stabilizer 360 brackets to be installed late 1980.
Arco (Brown & Root)	Java Sea	32"	12 ft.	' Line covered by 15 ft. of silt.
Esso Australia Ltd.	Sale, Australia	24"	Surf zone-80'	Expanding anchor installed in coral and screw anchor installed in sand.
Shell/Esso	Auk Field, North Sea	N/A	280 ft.	Anchored an SBM Manifold.
Corpoven	Lake Maracaibo, Venezuela	18"	45 ft.	Stabilized concrete coated flow lines.

RIVER/CREEK CROSSING

Company	Location	Pipe OD	Depth	Comments
Gulf Refining	Trinity River, Texas	6"-8"	30 ft.	**8fps current.
The Superior Oil Co.	San Juan River, Utah	4"-6"-8"	N/A	**Gravel river bottom
Tennessee Gas	Wax Lake Outlet Atchafalaya River, Louisiana	24"	60 ft.	**5fps current.
Dixie Pipeline Co. (Exxon)	Tallapoosa River, Alabama	12"	15 ft.	**Expanding anchor installed in granite bottom. Stabilizer 360 bracket.
Transco	Matagorda Bay, Texas	30"	5 ft.	Stabilized concrete coated line.
Gulf Oil Chemical Co.	Burnett Bay, Texas	8"	3 ft.	Subsidence caused line to float.
Lone Star Gas	Grapeland, Texas	10"	N/A	Line to be inundated.
Lone Star Gas	Coleta Creek, Nursery, Texas	8"	N/A	Stabilizer 360 used on line in washed out creek.

**Suspended pipelines

ONSHORE

Company	Location	Pipe OD	Comments
Esso Libya	Hateiba Project	30"	' Drill and set expanding anchors in epoxy in sandstone.
Florida Power & Light (Brown & Root)	West Palm Beach to Indiantown, Florida	18" (Insulated hot oil)	' High water table required anchors on 23 miles.
Oklahoma Natural Gas	El Reno, Oklahoma	24"	' Near South Canadian River.
Natural Gas Pipeline of America	Iowa, Kansas, Nebraska, Illinois	24"	' Anchored majority of overbend locations on Amarillo line.
United Texas Transmission	Katy, Texas	30"	' Weight coating not used due to difficulties caused by rice field terrain.
Valero Energy Transmission	Hallettsville, Texas	10"	' Dry ditch and creek crossing.
Valero Energy Transmission	Giddings, Texas	20"	' Anchors required in flood prone area.
Colorado Interstate	Rock Springs, Wyoming	24"	' Soft soil at creek crossings.

' New installation

SUMMARY OF ACTIVITIES

The activities that Barrios Technology, Inc. (BTI) has been involved in supporting the Mars/Phobos task fall in to three categories: Management, Technical and Documentation. The activities in each of these categories is summarized below:

1.0 MANAGEMENT:

a. **Task Orders**

We have written several draft versions of the Mars/Phobos Task Order. The first draft focused on the IBM PC-based CWPROP program and converting it for use in Mars/Phobos proximity operations studies. Further, we were to make recommendations on how to handle the 3-body problem. This evolved into a second draft Task Order which provides direction for BTI to build a baseline 3-body Mars/Phobos/Spacecraft Proximity Operations package. A third task order still being drafted, provides for enhancements to the 3-body program.

b. **Schedule/Resources**

We are using a PC-based project management tool to assess the activities involved in developing the Mars/Phobos software and in estimating the cost/schedule. We used this tool in assessing activities for the first PHOPOP release, and we plan to refine our schedule and resource estimates through further use.

c. **Project Reference Document**

BTI has collected information needed for the development of PHOPOP in a Project Reference Document that will be kept under tight configuration management. The document provides a single source for such data as astrodynamical constants, Mars and Phobos orbital elements, coordinate systems and reference material.

2.0 TECHNICAL

a. **Draft Requirements**

To guide the development of the Mars/Phobos software, we developed a draft set of user requirements. The requirements document focuses on both near-term and enhanced user needs. Therefore, a phased implementation approach is taken. In line with this is an emphasis on prototyping so that the system can evolve to meet undocumented user requirements and changes in requirements. Also included in the requirements document are guidelines/standards for software development and targeted hardware requirements.

b. Evaluation of the Colhessy-Wilshire (CW) Methodology as Applied to the Mars - Phobos case.

At the onset of this task it was suggested that the CW Targeting Methodology might be acceptable for use in the vicinity of PHOBOS, the investigations discussed below were undertaken in an effort to understand the applicability of this methodology to the Mars/Phobos system.

We are investigating the possibility of "factoring-in" the 3-body problem into the CW equations via additional terms in the equations. As a first step in this investigation we revisited the derivation of the CW linearized second order differential equation.

We have derived a system of three differential equations using the same methodology as the derivation of the CW equations except that the accelerations due to Phobos are included. The result is a set of three coupled non-linear differential equations. Hand calculation has shown that the Phobian components are at least as large as some of the components that arise in the standard CW approach. We intend to explore this further; however, our conjecture at this moment is that the CW equations will not provide the accuracy needed for targeting in the presence of Phobos.

Two other approaches have been suggested as methods of utilizing the CW equations. They are:

(1) An interactive, error correction approach and

(2) An approximation where the acceleration due to Phobos is taken as a constant for very short propagations.

Both of these approaches will be investigated in the near future.

c. CWPROP

One of the tasks in the initial draft Task Order was to convert the IBM PC-based CWPROP to perform Mars/Phobos Proximity Operations Studies. CWPROP was designed to do desk top studies of earth spacecraft relative motion. This is an acceptable tool when considering that two spacecraft orbiting the earth have insignificant mass and potential relative to the Earth's. However, when converting to the Mars/Phobos/Spacecraft System where Phobos and the Spacecraft are the two satellites of interest, we have reservations concerning the applicability (See 2.a above) due to the gravitational influence of Phobos. Therefore, we recommended the problem be further investigated and that a 3-body analysis tool be developed. However, we made the conversions to CWPROP so that point source satellite analysis could be accomplished either with Mars or Phobos as the center of the system. With Mars as the center of the system, the converted program, CWMARS can be a useful tool for assessing proximity operations of two spacecraft (not in proximity to Phobos) orbiting Mars. Once 3-body Mars/Phobos simulators are developed the influence of Mars can be better assessed and the validity of CWPHOBS as a legitimate analysis tool can be determined.

d. **Hardware Trade Study**

We studied three basic options for proposed PHOPOP hardware. These were an 80386-based system, the Commodore Amiga, and the Sun-3 systems. We selected the Sun-3 for the following reasons:

(1) The Sun work station provides a Unix programming environment that aids software development.

(2) It has all the graphic capabilities that will be required for the proposed processor.

(3) It provides a state of the art windowing user interface.

(4) Perhaps the best reason for selecting Sun computers was that we already have them.

e. **BTI Sun-3 Work to Date**

(1) We have identified and defined "objects" for use in an object-oriented design.

(2) We have also run test cases to verify that the Sun-3 graphics and windowing capability can satisfy user requirements.

(3) The MODULA-2 Prototype discussed below (g.) was translated to Ada on the PC.

(4) The Ada version of the prototype integrator was ported to the Sun and output data has been compared with the original version and no errors have been found.

f. **IBM PC Interface Option**

Work done thus far on the IBM PC consisted of developing a user interface in the Basic programming language. Input screens and graphic output screens (plots of Mars, Phobos, and a spacecraft) were developed. This interface will be the basis for the PHOPOP user interface for Version 1.0.

g. MODULA-2 Prototype

BTI personnel have engineered, designed, coded, and tested a prototype three body integrator (ORBIT) for an AT type microcomputer. ORBIT was written in MODULA-2 and supports the use of an 80287 math co-processor, if available. ORBIT uses central body gravitational forces for Mars and Phobos, and assumes that the spacecraft mass affects neither Mars nor Phobos and that the mass of Phobos does not affect Mars. The required inputs to ORBIT include: integration stepsize (sec), data output period (sec), length of run (sec), spacecraft position with respect to Phobos (km), and spacecraft velocity with respect to Phobos (km/sec). The outputs provided by ORBIT are time and both spacecraft and Phobos position and velocity, with a user option to output the spacecraft position and velocity in either UVW or Inertial coordinates.

This integrator has been used as an engineering tool to help verify not only the integration scheme to be used in PHOPOP but also as a tool to analyze the stability of Libration Points and Phobian orbits.

h. Potential Model Design

BTI personnel have designed and coded a generic gravitational potential model capable of use for up to 18x18 models. This computational method is heavily based on JSC Internal Note NO. 75-FM-42, by Alan C. Mueller, dated June 9, 1975, titled "A FAST RECURSIVE ALGORITHM FOR CALCULATING THE FORCES DUE TO THE GEOPOTENTIAL (PROGRAM: GEOPOT)". The BTI model is a single block of code capable of calculating a potential acceleration at a point in space for any body for which a Body Structure is provided. The required Body Structure is a data structure which contains and organizes all the body unique data required by the model, thus allowing a single model and several data structures to provide potential accelerations for many bodies. This potential model has been implemented in the MODULA-2 programming language and will be incorporated into a later version of ORBIT to be verified.

i. American Astronautical Society (AAS) Abstract and Letter

In April of 1989, the AAS and Goddard Space Flight Center will be co-hosting an International Symposium on Orbital Mechanics and Mission Design. Three BTI employees are writing a paper and have submitted a letter and abstract on the topic, "Investigations Into the Mars-Phobos-Spacecraft Three Body Problem". The information presented in the letter/abstract, and to be expanded in the paper, relates directly to our development of PHOPOP and its potential use in designing Mars/Phobos missions.

j. Calculation of the Libration Points of the Mars-Phobos System.

We have computed the five Libration Points (See Figure 1) Phobos System based on the following assumptions: 1) that all forces are central body, 2) Phobian gravity has no effect on Mars, and 3) any third body (a spacecraft) will have negligible mass with respect to both Mars and Phobos. We have also calculated the L2 point without assumption number 2 and found negligible difference (1 mm in position). We have made several computer runs using the PHOPOP prototype that indicate the libration points closest to Phobos (L2 and L3) are unstable.

Although the L1 Point has yet to be analyzed with the PHOPOP Prototype, analysis of the equations which give rise to these points indicate that L1 is unstable. We are just beginning our analysis of the L4 and L5 points. These are the stable L-points and could be useful as locations of COMM/NAV Satellites for example.

3.0 DOCUMENTATION

a. Library

As a result of our relatively new involvement in Lunar and Mars Exploration activities we are taking positive steps to improve our knowledge base through literature research and via our establishment of a Mars Exploration Library. The library currently consists of several published documents from NASA/JSC and JPL in addition to a variety of papers from professional societies.

b. Mars-Phobos Literature Review.

BTI has collected several books, papers and associated documents as part of a "Project Library". The specific documents can be found as part of the Project Reference Document. We have also conducted a preliminary literature search of articles that address topics of interest to a Mars/Phobos mission. References that were searched include: International Aerospace Abstracts, Physics Abstracts, and Scientific and Technical Aerospace Reports, and below are some of the more interesting abstracts found.

"Periodic Orbits Around a Satellite Modeled as a Triaxial Ellipsoid", a Masters Thesis by Rodney Werner, December 1987, Air Force Institute of Technology.

"The Case for Mars, II"
Science and Technology Series, Vol.62, 1985

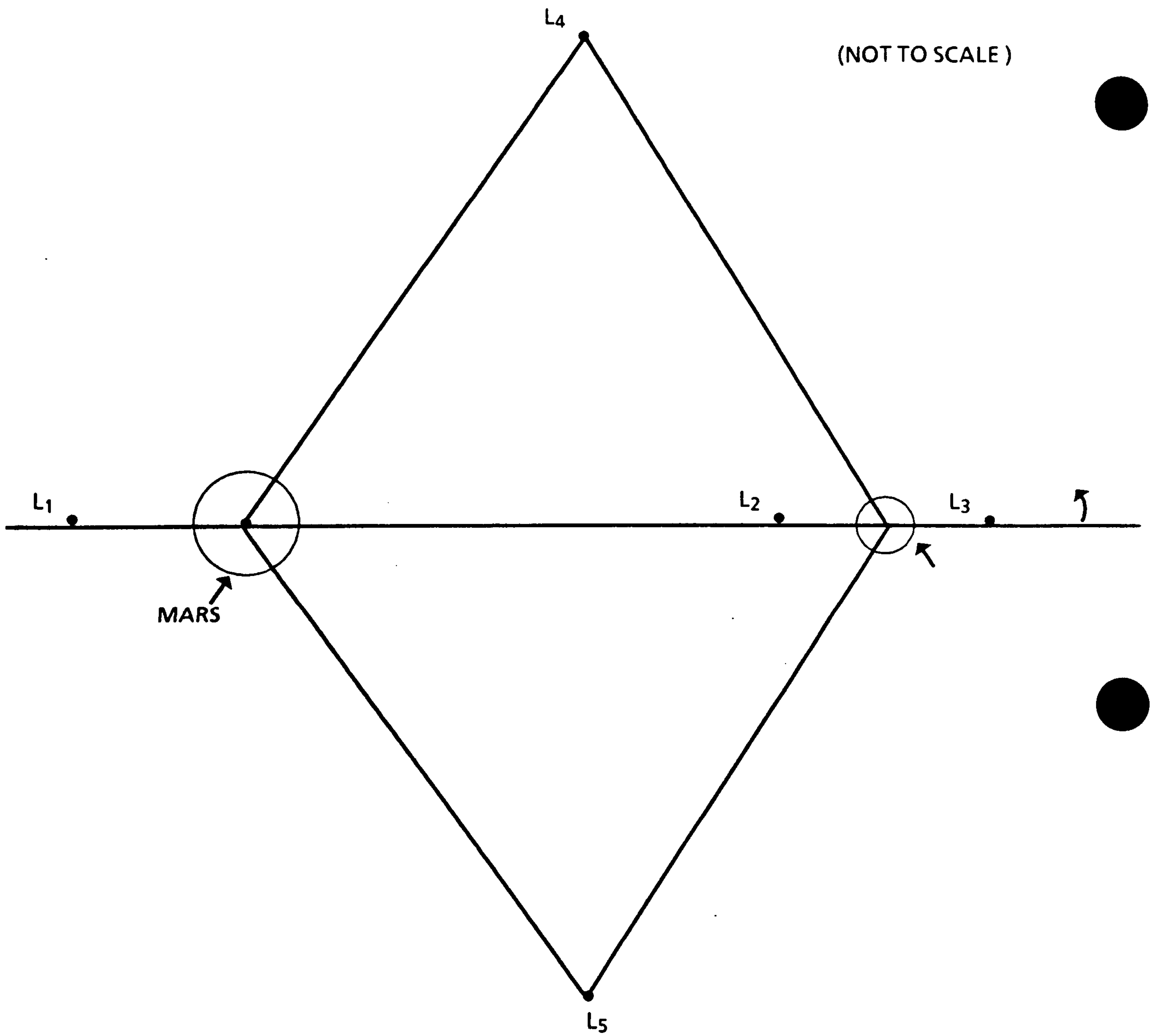
"Control of the Motion of a Spacecraft Near a Collinear Libration Center in the Restricted Elliplical Three Body Problem"
Kosmicheskic Issledovaniia, Vol. 24 July/Aug.'86

PRODUCTS

The following products have been provided to Mr. Al DuPont/(IZ):

- Updated CWPROP Programs (CWMARS and CWPHOBS) on floppy disk. BTI TM#2227

- Draft User Requirements Document. BTI TM#2215
- Hardware Trade Study (Meeting handout)
- Draft Task Orders
- AAS Abstract and Letter
- Numeric Output of Prototype Simulator
- Draft Listing of Object-Oriented Objects



LIBRATION POINTS FOR THE
MARS/PHOBOS SYSTEM

FIGURE 1

DRAFT

PRELIMINARY USER REQUIREMENTS
FOR THE
PHOBOS PROXIMITY OPERATIONS PROGRAM
(PHOPOP)

CONTRACT NO. TBS

7 OCTOBER 1988

PREPARED FOR: EXPLORATION STUDIES OFFICE
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MARS/PHOBOS USER REQUIREMENTS

GENERAL: The following requirements define the need for a software capability to model the Mars/Phobos system, and simulate multi-body inertial and relative motion involving Mars, its moons (Phobos or Deimos), and one or two artificial satellites.

The below stated requirements are divided into delivery phases. Each phase is further divided into General Requirements, Outputs, Inputs/Processing, Design/Development Standards, and Hardware. Only "additional" requirements are addressed after Phase 1. NOTE: An interim release, Version 1.5 may be required to meet minimum user needs (Specifics are TBD).

PHASE 1: (Prototyping Phase)

General Requirements: Convert CWPROP program to run with Mars/Phobos astrodynamic constants on an IBM PC or compatible system (done). Develop software to model the Mars/Phobos (or Deimos) 3-body problem to execute on a Sun 3 computer system. BTI will closely interface with the NASA task monitor via demonstrations and prototyping to further refine user requirements and to take advantage of Sun 3 capabilities in designing the user interface. As a general requirement, the Phase 1 software will functionally provide much of the same user interface capability as CWPROP (e.g., case-by-case options, manual or automatic, ability to change conditions, etc.); however, the interface will be enhanced via Sun 3 capabilities such as windowing and high resolution graphics.

Outputs: All outputs are required to be displayed at the user terminal with hardcopy options which include post-processing after each case and/or a stop screen/print screen option during execution. Phase 1 user terminal outputs will be monochrome and include both graphics and text. Printed output will be text only except in the print screen mode. Within these guidelines, the following will be provided in TBD format.

- Inertial and relative state (Local Vertical Local Horizontal (LVLH)) position, velocity, and time (see Figure 1). Time will be provided in hours, minutes and seconds for both total mission elapsed time and mission segment (case) elapsed time.

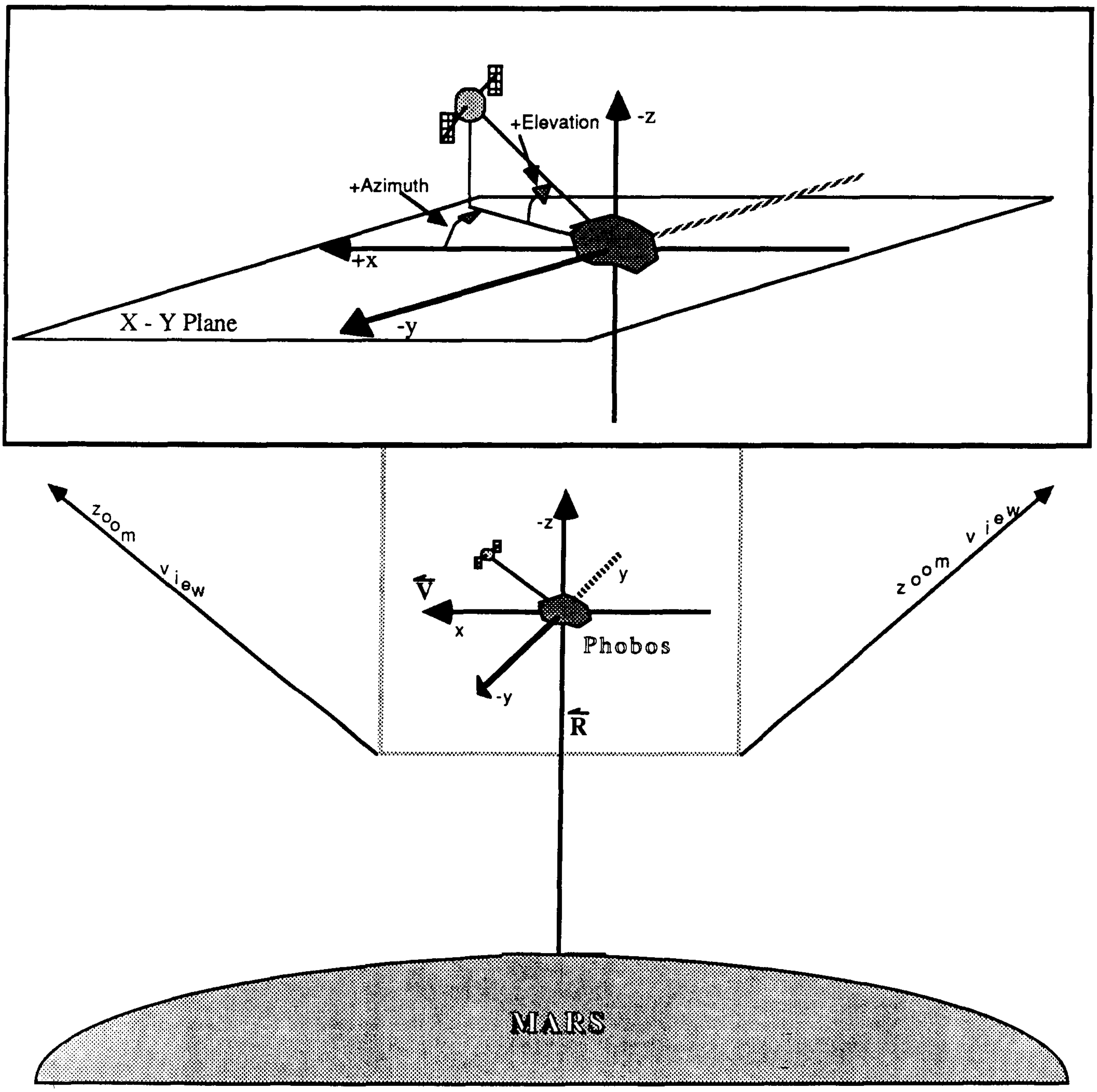
- Relative motion plot in Phobos-centered LVLH system. This is a plot of the position points of spacecraft motion relative to the center (Phobos) of the LVLH system. The points are plotted at user-selected time intervals.

- Targeted or external delta-v. Targeting allows the user to specify a final position where the chaser spacecraft is to end up and the time in which the trajectory should be completed. The program will calculate the velocity vector required to have the chaser at the specified position at the specified time and then integrates the chaser spacecraft to that position.

• Figure 1

PHOBOS

Local Vertical Local Horizontal System (LVLH)



- Delta-v angles measured in azimuth and elevation relative to the center of the LVLH system, i.e., Phobos (see Figure 1). Azimuth is measured in the XY plane, where the +X axis aligns with Phobos'es velocity vector, and the Y axis (out-of-plan axis) is perpendicular. The Z axis corresponds to the radius vector from the center of Mars to Phobos and is negative in the direction away from Mars. Positive azimuth is measured clockwise when looking down the Z axis toward Mars and rotating clockwise (up to 180 degrees) from the +X axis to to the projection of the spacecraft onto the XY plane. Negative azimuth is measured in the counterclockwise direction. Elevation angle is measured in the plus or minus Z directions from the azimuth point (spacecraft projection onto the XY plane) to the spacecraft's radius vector (from the center of the LVLH system to the spacecraft). Positive elevation is in increments from 0 to 90 degrees and measured in the -Z direction (away from Mars). Negative elevation is measured in the opposite direction.

- Summary of key input selections/conditions.

- Interim outputs will include menu-driven input selections and TBD error comments. At a minimum, Phase 1 will have the capability to give the user the option to further propagate (coast) or target using the existing relative state, or to run a new case with new conditions. Targeting has already been described above. Coasting allows the chaser craft to "drift" for a specified amount of time. Internally, the software, will actually be integrating the inertial state vectors to the specified time(s) and then converting these states to the LVLH system.

- Window outputs using inherent capabilities of SUN 3 systems. An option will be provided to display relative motion in one window with associated data in another window (See Figure 2, a sample from CWPROP). Another option will be to display differing degrees of detail of the 3-body (or 2-body) motion between Mars, Phobos, and a spacecraft orbiting Phobos. For example, the user will be able to change from a Mars/Phobos graphics view to a Phobos/spacecraft view or use the inherent capability on the Sun 3 operating system to zoom in/out on graphics.

- Data files will be saved for post-processing to be accomplished for each case and for parametric study and plot applications in future Phase releases. Included in these files will be initial and final state vector conditions, selected user conditions for each case, inertial and relative state vectors at user-selected time intervals, delta-v history, and targeting time in hours-minutes-seconds or orbit periods, mission and case time histories, and state vector information at the integration step size if selected.

Figure 2

CWPROP OUTPUT

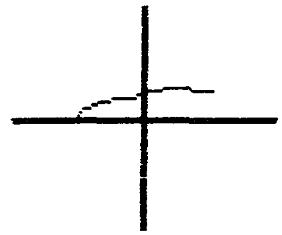
```
**** TARGET SEQUENCE ****

TIME DURATION IN MM.SS=  10.00

ALTITUDE=  100  NMI

UNITS IN kft  & FT/SEC

MET DURATION IN MM.SS=  45.07
```



```
  INITIAL POSITION                                FINAL POSITION

  X-AXIS=-4.5670E+00                              X-AXIS=-5.2293E+00
  Y-AXIS=+0.0000E+00                              Y-AXIS=+0.0000E+00
  Z-AXIS=-2.6435E+00                              Z-AXIS=-2.4803E+00

  INITIAL VELOCITY     DELTA VELOCITY             FINAL VELOCITY

  X-AXIS=-5.6601E+00  X-AXIS=-5.6530E-01  X-AXIS=-5.3531E+00
  Y-AXIS=+0.0000E+00  Y-AXIS=+0.0000E+00  Y-AXIS=+0.0000E+00
  Z-AXIS=+1.1497E+00  Z-AXIS=-2.2529E+00  Z-AXIS=+1.5675E+00
PRESS RETURN TO CONTINUE
```

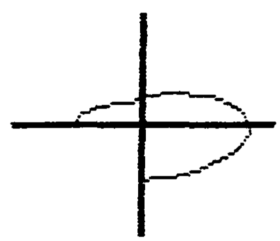
```
**** TARGET SEQUENCE ****

TIME DURATION IN MM.SS=  60.00

ALTITUDE=  100  NMI

UNITS IN kft  & FT/SEC

MET DURATION IN MM.SS=  87.07
```



```
  INITIAL POSITION                                FINAL POSITION

  X-AXIS=-1.0730E+00                              X-AXIS=-1.4610E-06
  Y-AXIS=+0.0000E+00                              Y-AXIS=+0.0000E+00
  Z-AXIS=+4.8419E+00                              Z-AXIS=+5.0000E+00

  INITIAL VELOCITY     DELTA VELOCITY             FINAL VELOCITY

  X-AXIS=+8.7812E+00  X-AXIS=-5.6530E-01  X-AXIS=+9.0864E+00
  Y-AXIS=+0.0000E+00  Y-AXIS=+0.0000E+00  Y-AXIS=+0.0000E+00
  Z-AXIS=+1.5259E+00  Z-AXIS=-2.2529E+00  Z-AXIS=+1.1063E+00
COAST,TARGET,BURN,EXDU,NEW CASE,NEW PLOT,DELETE,STOP (C,T,B,E,N,P,D,S)?
```

Processing:

- Process user inputs to establish initial conditions for the first case. These inputs and associated processing are:

-- Units to be used for plotting. Options are feet, kilofeet, nautical miles, or kilometers. Processing converts these inputs to internal units.

-- Boundaries for the LVLH axes for plotting in the units already selected, above. This establishes the maximum expected distance that the spacecraft's position will be plotted in the positive or negative directions for each of the LVLH system axes.

-- Plotting step size in the units already selected by the user. For example, if feet were selected and a step size of 100 is chosen, then a point will be plotted for every 100 feet of relative motion.

-- Print options. Based on user-selected input, conditions are set for printing text data (per output requirements stated above) to the printer.

-- State vector options. The user has several options for inputting initial state vectors:

1. Use default inertial state for Phobos and user input of spacecraft relative position and velocity. In this option, the program will use an internal state vector for Phobos and compute the inertial state for the spacecraft as initial conditions. User input units must be those already selected. If metric units were chosen, for example, the metric units will be used for position and velocity throughout the case. This applies to both inertial and relative states. In all cases the program will convert to standard internal units.

2. Use default inertial state for Phobos and user inputted inertial state for the spacecraft.

3. Use user-provided inertial states for both Phobos and the spacecraft.

-- Select Case option. Based on user input choice, set conditions for either coasting, targeting, or external delta-v logic to be used. If "coast" is chosen, then logic will be executed to coast (at the internal integration step size) to an input time or for an input duration. If "targeting" is selected, then user-input target point data in the Phobos-centered system, is used along with the user-input delta-time (to get from the initial position to the target point) to compute the required spacecraft inertial state delta velocity. This velocity is automatically applied and the state vector is integrated to the desired time. If external delta-v is chosen, then this delta-v provided by user input, is converted to inertial components, applied to the inertial state vector, and integrated to an input time or for an input duration. For all three user input choices, processing must convert inertial state information to relative system format for presentation to the user per output requirements.

-- Set conditions for scroll or manual output. Based on user input selection, set a flag for providing either a scrolled or manual control of screened graphics and textual output. Scrolled output will provide a relative motion plot update and state vector information for each relative data point to be plotted on the user terminal screen. This information scrolls up the screen at a rate reasonable for user scanning. A capability will be provided for the user to stop the scroll with one key stroke. The manual option allows for screen output of only one plot point, and associated data, at a time. When the user is ready to see the next point then a single key stroke will allow display of the next set of sequential data (see Figure 3 for an example of two screen data points and plots from CWPROP).

-- Set conditions for time or orbit period (rev) processing. If time is selected, then the user must also input the time duration and number of output data points required. The program will compute the delta time between each data point, and output to the screen at that step size. If revs is selected, then the user must also input the number of revs (fractions of revs are acceptable) and the required number of data points. The orbital period of Phobos is calculated and used in calculating the time duration for the output data. The remainder of the processing is the same as for the time option.

-- Execute targeting if applicable. Two-impulse targeting of a quality and precision comparable to that being used for Shuttle Rendezvous and Proximity Operations will be used. The spacecraft inertial state velocities are updated prior to integrating state vectors.

-- Integrate inertial state vectors. All inertial state vectors are integrated at the internal step size retrieved from a controlled file. State vector data is output to selected media at user-selected intervals. This includes both inertial and relative system data for the spacecraft. The precision and quality of the integrator will be comparable to that used in Shuttle Rendezvous and Proximity Operations, with software design providing the flexibility to easily add a more precise integrator at a later date.

- Process user inputs for running additional or new cases. After completing the first or previous case, a menu is presented to the user which provides an option to coast, target, apply an external delta-v, run a new case, run a new plot, delete previous case, or stop as defined below.

-- Options to coast, target, or to apply an external delta-v are the same as for an initial case. If a case has just been completed and one of these three options are selected, then the end conditions from the previous case are the starting conditions for the next case. The plot from the previous case is extended with the relative motion results of the next case.

-- If the option to run a "new" case is selected, then the plot is cleared and a completely new set of conditions, except plot boundaries and plot increments, must be provided by the user. Mission times and case times are set to zero.

-- If the option to run a new plot is selected, then the user must provide new plot boundary and plot increment conditions. Then the previous case's plot is cleared and replotted within the new plot conditions. This capability is useful, for example, when a relative trajectory is not within the stated boundaries, and the boundaries merely need changing to see the trajectory.

-- If the delete option is chosen, then all the information from the last case is erased from the screen (including its plot), and the end conditions from the previous case are displayed (including the plot to that point in time). This capability is useful for correcting for a bad set of data used in the last case. If the delete option is chosen twice, three, etc. consecutively, then the program will delete all data back to two, three, etc. cases respectively.

- Model full potentials, (18x18) for mars and (6x6) for Phobos as specified in Reference 1. Only verifiable potentials will be used for the Phase 1 delivery. The program will use the potential models and other astrodynamic data retrieved from internal files as part of the state vector integration process.

Design/Development Standards:

- Algorithms and Program Design Logic will be provided for new development.

- Configuration control of Mars/Phobos/Deimos constants used internally by the software will be maintained and coordinated with the user.

- Constants will be in a file rather than in the code, and controlled.

- Highly commented code will be delivered with program documentation.

- References will be documented.

- Integrated software design approaches will prevail with the goal of easily transitioning into future releases and new user requirements.

- Use of Object-oriented design will be considered as a design possibility. Correspondingly, use of Ada as a programming language will be considered.

- Optimize use of programming languages to provide fast math processing at least at double precision, while at the same time providing a potential for high resolution graphics, batch processing, and parametric/statistical studies.

Hardware: The SUN-3 computer at BTI will be the target machine for Phase 1 development. Printed output will be designed for use on an HP Laser Jet.

PHASE 2:

General Requirements: Phase 2 will functionally provide a user interface and display capability similar to that of CWPROP. However, results of prototying activities in Phase 1, combined with utilization of Sun 3 capabilities should yield an enhanced user interface over CWPROP and Phase 1. In addition, the following capabilities whose requirement details will be refined with the user should be provided in Phase 2:

- Simple engine model (TBD).

- Sequenced maneuver processing based on a user input table. The table sets up a batch process and will contain applicable coast, targeting, external delta-v, etc. information.

- Propellant and delta-v cumulation.

- First-order Mars and Phobos surface models.

- Introduction and Help software.

- Post processor plots based on user input selections.

- Additional case options:

  -- External Forces.

  -- Spacecraft Burn Sequence (dependent on fidelity of engine model - TBD).

- Plotter outputs.

- Option to select potential sizes.

PHASE 3:

General Requirements: Phase 3 requirements are:

- Initial G&C capability.

- Six degrees of freedom attitude.

- Jet modeling.

- Three-body problem with 4 bodies (i.e. - capability to include a second satellite into Phobos studies.

- Vehicle models including an excursion vehicle (1/2 Shuttle) on Phobos surface or in orbit.

PHASE 4:

General Requirements: The magnitude of the requirements is such that the Phase 3 package would be a module to be integrated with the other significant "modular" capabilities planned for Phase 4. Anticipated Phase 4 capabilities are:

- Statistical analysis module such as Monte Carlo.

- Navigation sensor models.

- Deimos model.

- Three-body equivalent to CW equations (to be studied).

- Object graphics - "Video-game" representation of relative motion of high resolution graphics figures from point-to-point on surface of Phobos (or Deimos) and between the surface and a satellite in motion.

- Atmosphere models.

- Digital Auto Pilot.

- Manned Maneuver Units (MMU).

- Window views from satellite vehicle or MMU.

- Man-in-the-loop simulation.

- State-of-art high resolution graphics.

REFERNCES

1. Yen, C.L. and Blume, W.H., "Mars Observer Planetary Constants and Models, Preliminary", JPL D-3444, July 1, 1986.

2. Duxbury, T.C., "The Figure of Phobos", #183-501, JPL and California Justitute of Technology, June, 1988.

POWER AND PROPULSION PARAMETERS FOR NUCLEAR ELECTRIC VEHICLES

COMPILED BY

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VERSION 1 • RELEASE 1

JULY, 1988

PREFACE

This document presents current and projected performance parameters for nuclear powered electric propulsion systems which could have application for a round-trip Mars mission. These are consistent in that they represent a consensus of several sources. They are realistic in that they are either actual test data, design data, or based upon expert projections of performance from existing systems. The authors are attempting to foresee future developments in space nuclear engineering, safety, and electric propulsion systems; a risky business to be sure. We recognize that serious and profound differences abound regarding design considerations and assumptions of these systems. We have assumed steady, but conservative growth based upon a modest increase in funding of existing development programs. As we acquire more information of the mission requirements upon the systems, we shall incorporate these. We will maintain such data for use by our colleagues who are analyzing future missions for the Office of Expoloration. All data will bear the imprimatur of the Office of Aeronautics and Space Technology as well as that of Lewis Research Center. Futhermore, we do not intend this document to be an advocacy manual for nuclear electric propulsion, but merely a source book of data for the mission analyst.

The reader is cautioned not to assume that these systems are currently available as flight type hardware. Much earth based testing remains before a flight demonstration of these propulsion and power systems can occur.

The document contains data in tabular and graphical form for current and projected propulsion and power systems with appendices which contain any supporting analyses. References also appear at the end of the document.

ELECTRIC PROPULSION PERFORMANCE

Jim Gilland

The accompanying tables and curves represent current and anticipated thruster efficiencies (η), specific impulse (Isp), and specific masses (α) of Electrostatic (ion) and Magnetoplasmadynamic (MPD) thrusters for exploration class missions to the Moon and Mars. Manned exploration of these planets using electric propulsion (EP) will require high ($\sim$ megawatts) power and Isp levels (3000-10000 seconds); the enclosed projections are therefore intended to model EP systems operating in the 100 kWe - 1MWe power regime. Specific issues and background for the two types of EP considered here are discussed below, as well as the bases for obtaining their respective performance models.

Ion Propulsion

Table 1 from reference 3 presents current and anticipated performance for ion thrusters which use xenon and argon as propellants.

Propellant Choice

Ion propulsion has a background extending over a range of propellants and power levels. Past propellant choices of cesium (Cs) and mercury (Hg) have been replaced by noble gas propellants in order to avoid the toxicity and contamination issues inherent in the two metallic propellants. Primary propellants are xenon (Xe), krypton (Kr), and argon (Ar).

Propellant choice governs the Isp range of efficient operation: xenon is effective in the lower (3000-5000 s) Isp range, and argon attains almost 10,000 s Isp. Although Xe is presently used almost exclusively in engine tests, its practicality for high power missions requiring large amounts of Xe is dubious because of its cost (see Appendix C). Thus argon or krypton are favored for high power applications where thrust and Isp are both relatively high.

Performance

The ion engine performance data used in this assessment are based on the experimental behavior of xenon thrusters at low powers. At present, Xe research engines have been operated at power levels ranging from 3 - 30 kWe per engine; it is anticipated that such engine designs can be scaled to higher powers by increasing thruster area while maintaining the Isp and efficiency behavior seen at low powers. Megawatt argon ion thrusters would require thrust areas of $\sim$2.5 m^2, implying a circular grid diameter of $\sim$1.8 m. Assumptions of realistic values for losses due to beam divergence, multiply ionized species, and propellant utilization are incorporated into the data[1]. The calculated performance data can be described by an efficiency - Isp equation of the form

$$\eta_T = \frac{bc^2}{c^2 + d^2}$$

where c = exhaust velocity = Isp* one Earth g, and b and d are coefficients obtained by a non-linear least squares approximation. This form seems to have been originated by Ernst Stuhlinger with b=1 for an "ideal" thruster. The equations, curves, and data are shown in Figure 1, Ion Thruster Projected

Performance. The values of the coefficients obtained from these data are:

Propellant	b(n.d.)	d(km/s)
Xe:	.856	11.869
Kr:	.855	15.00
Ar:	.835	22.51

This approximation has a continuous first derivative and appropriate behavior at the limits. Ion engine systems also require Power Processing Units (PPU) to configure the power into the high voltage (~2000 V), low current form used by these devices. The PPU introduces additional losses which must be included in thrust systems performance calculations; typical PPU power efficiencies (η_{PPU}) are 90 - 92%. Overall system efficiency if therefore the product of η_T and η_{PPU}.

Engine Life

Engine life at low powers has been seen to be greater than 10,000 hours. Projected life for high power, high Isp argon ion engines is greater that 5000 hours[3].

Specific Mass

Specific mass data were also calculated for ion thruster systems over a range of Isp. In the case of the ion thruster, specific mass is more a function of Isp than power. The thruster system considered includes a single thruster, mounting structure, gimballing, PPU, and thermal management systems. Propellant, tankage, and feed systems are not included. Propulsion system component masses were obtained using empirical relationships derived by Dave Byers of LeRC[4] and described by Galecki and Patterson[5]. These data were also fit with a polynomial curve which is shown with the data in Figure 2.

$$\alpha(\mathrm{kg/kWe}) = 2.66 + 1.42 \times 10^{-4} * \mathrm{Isp} + (4.26 \times 10^{7})/\mathrm{Isp}^2$$

Table 1. Current and Projected Ion Thruster Performance

	Xenon		Argon	
	Current	Projected	Current	Projected
Specific Impulse (K seconds)	3.3 - 5.0	2.5 - 5.5	5.7 - 7.7	4.4 - 9.4
Efficiency (%)	66 - 75	69 - 78	61 - 64	67 - 75
Thrust (N)	0.29 - 0.67	16 - 34	0.29 - 0.68	16 - 34
Power/unit (kWe)	7 - 22	290 - 1160	13 - 40	525 - 2105
Operating Life (hr.)	<5000	>5000	<5000	>5000
Effective diameter (cm)(equivalent area)	30	160	30	160

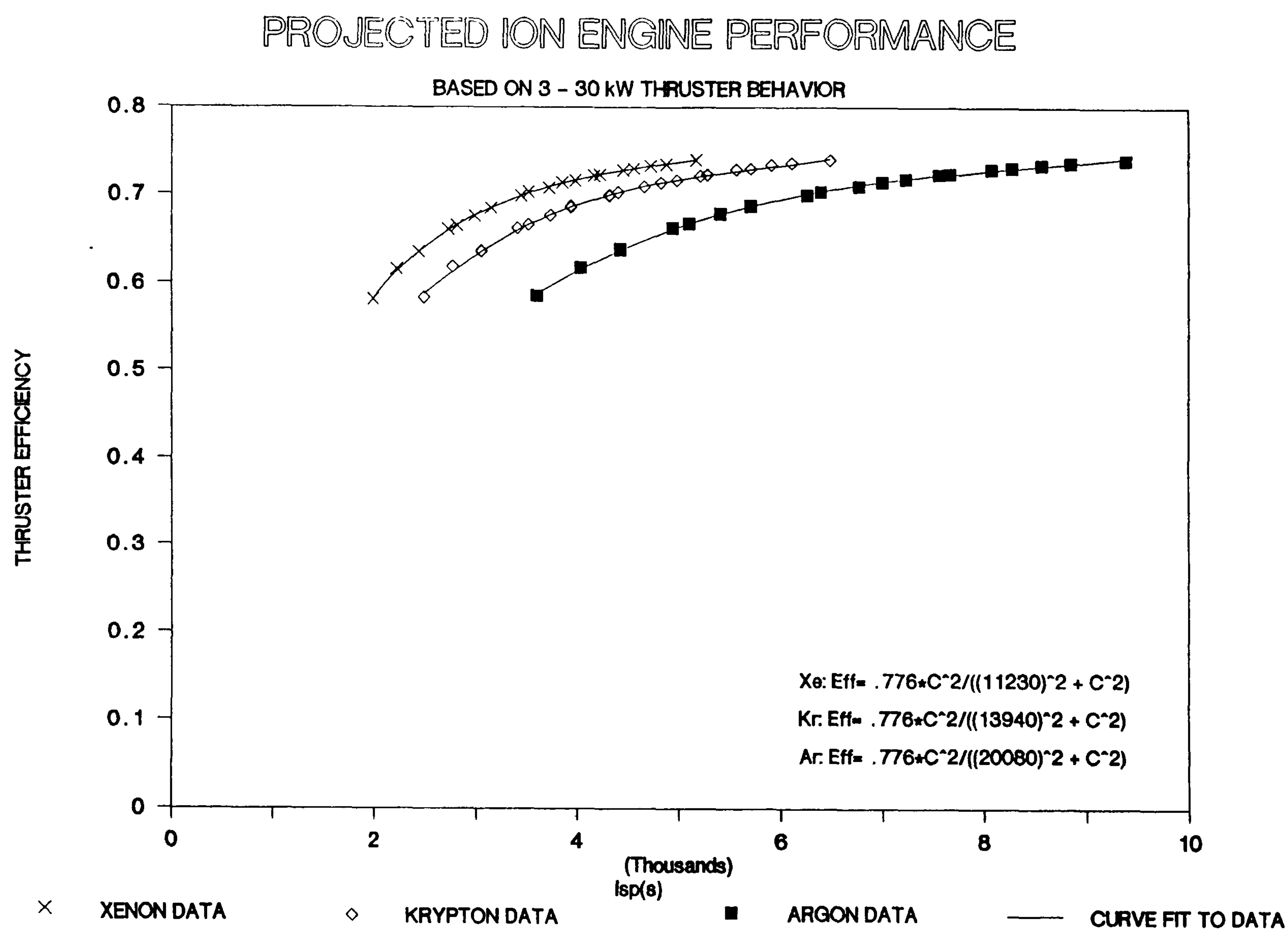

PROJECTED ION ENGINE PERFORMANCE
BASED ON 3 - 30 kW THRUSTER BEHAVIOR
THRUSTER EFFICIENCY
0.8
0.7
0.6
0.5
0.4
0.3
0.2
0.1
0
0
2
4
6
8
10
(Thousands)
Isp(s)
Xe: Eff= .776*C^2/((11230)^2 + C^2)
Kr: Eff= .776*C^2/((13940)^2 + C^2)
Ar: Eff= .776*C^2/((20080)^2 + C^2)
XENON DATA
KRYPTON DATA
ARGON DATA
CURVE FIT TO DATA

FIGURE 1

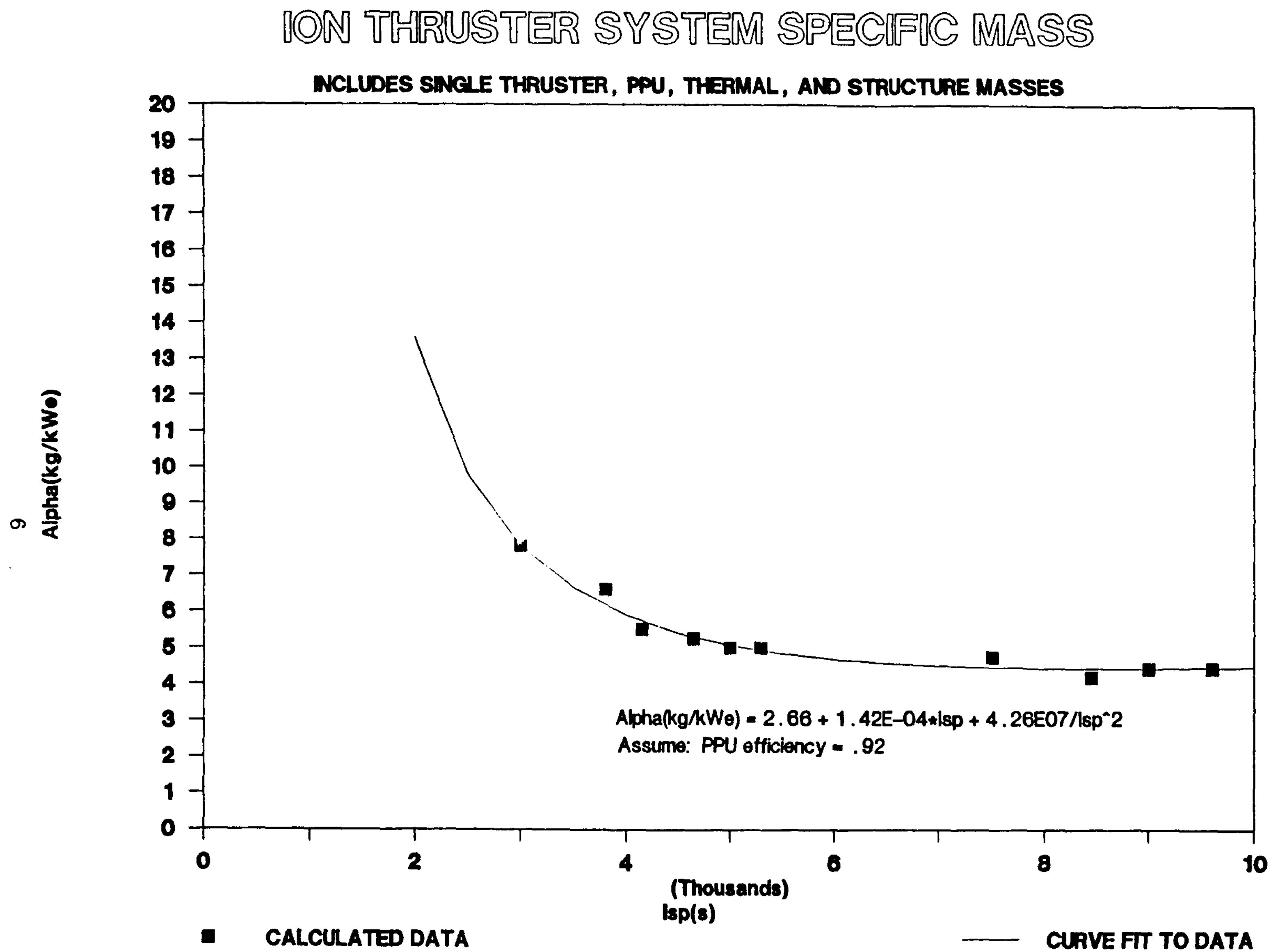

ION THRUSTER SYSTEM SPECIFIC MASS
INCLUDES SINGLE THRUSTER, PPU, THERMAL, AND STRUCTURE MASSES
Alpha(kg/kWe)
20
19
18
17
16
15
14
13
12
11
10
9
8
7
6
5
4
3
2
1
0
0
2
4
6
8
10
(Thousands)
Isp(s)
Alpha(kg/kWe) = 2.66 + 1.42E-04*Isp + 4.26E07/Isp^2
Assume: PPU efficiency = .92
CALCULATED DATA
CURVE FIT TO DATA

FIGURE 2

MPD Propulsion

MPD thrusters utilize electromagnetic forces to accelerate a plasma to high exhaust velocities (c). The magnetic forces used can be either self induced by current flow through the device (the "self-field" MPD thruster), or applied as a diverging "magnetic nozzle" to accelerate the plasma via expansion. Either form of device has shown the potential for high Isp, high thrust density operation at powers ranging form 100's kWe to multimegawatt levels. Table 2 presents current and projected MPD thruster performance.

Propellant Choice

The performance projections made herein assume H_2 for MPD thruster propellant because of its potential for achieving high levels of efficiency and Isp. Isp levels up to 6000 s have been demonstrated experimentally in pulsed MPD devices using H_2. Pulsed applied field devices have attained efficiencies up to 45% at Isp of 3000 s using H_2 propellant in the megawatt power range[6,7]. The possibility for Isp up to 10,000 s using H_2 may exist depending upon improvements in thruster design[8,9].

Performance

Because of the limited data base for MPD performance using H_2 propellant, particularly at high power and Isp, a model was derived to project experimental performance to higher Isp and to anticipate some improvements with efficiency with futher development of the thrusters. The model is empirical, based upon results reported by Tahara, et. al. using quasisteady applied field and self field devices at megawatt power levels and Isp levels up to 3000 seconds. The effects of applied field and self field MPD behavior are included in the approximation. The resulting equations calculate thrust efficiency (η) as a function of specific impulse and power input to the thruster (Figure 3). The actual derivations and equation results are explained in Appendix A. The projected efficiencies are for the thruster only, and do not include the PPU efficiency (typically 92%).

Engine Life

Actual MPD thruster systems will be operated in a steady state mode rather than a pulsed configuration. Present experiments are restricted to sub megawatt steady state or multimegawatt pulsed operation by the limited pumping and power capacity of the experimental facilities. At present such devices have been operated for 100's of hours at less that 100 kWe and for millisecond pulses in the megawatt range. A key development for actual space systems is therefore thruster lifetime. Both electrodes, and particularly the cathode, suffer from significant material erosion which presently puts an unacceptable limit on MPD thruster lifetime of a few hours[10]. Electrode erosion, particularly that of the cathode, is still not well understood. Projections of MPD thruster lifetime are presently 5000 hours or more.[8]

Specific Mass

The specific mass (α) of these systems for a range of powers was calculated using empirical relations also derived by Byers for arcjet/MPD-type thrusters. These data are shown in Figure 4, calculated for a single thruster assembly, as was done for the ion thrusters. Thruster system mass includes thruster, structure, and thermal control systems, but not propellant, tankage, or feed systems. Thermal mass calculations used assumptions of 92% PPU efficiency and 60% thruster efficiency to determine the mass of the system required for dissipation of the energy. The data yield the equation

$$\alpha(\mathrm{kg/kWe}) = 1800/\mathrm{P(kWe)} + 2.25$$

Table 2. Current and Projected MPD Thruster Performance

	CURRENT[1]	CURRENT[2]	PROJECTED[3]
Propellant	Hydrogen	Argon	Hydrogen
Specific Impulse,s	4900	1100	5000
Efficiency, %	43	17	60 - 70
Thrust, N	27	8.6	100
Power/Unit, kWe	1500	273	1500
Operating life, hr	1	1	5000
Operating mode	Pulsed	cw	cw

[1] Highest Observed Performance at Conditions below "onset" of high erosion. Ref. IEPC Paper 84-11, 1984. (ISAS, Japan)

[2] Highest Steady-state power data. Ref. AIAA Paper 87-1019, 1987. (Stuttgart, Germany).

[3] Significant Uncertainties exist in high pwer MPD thrust (efficiency) and life.

PROJECTED MPD THRUSTER PERFORMANCE

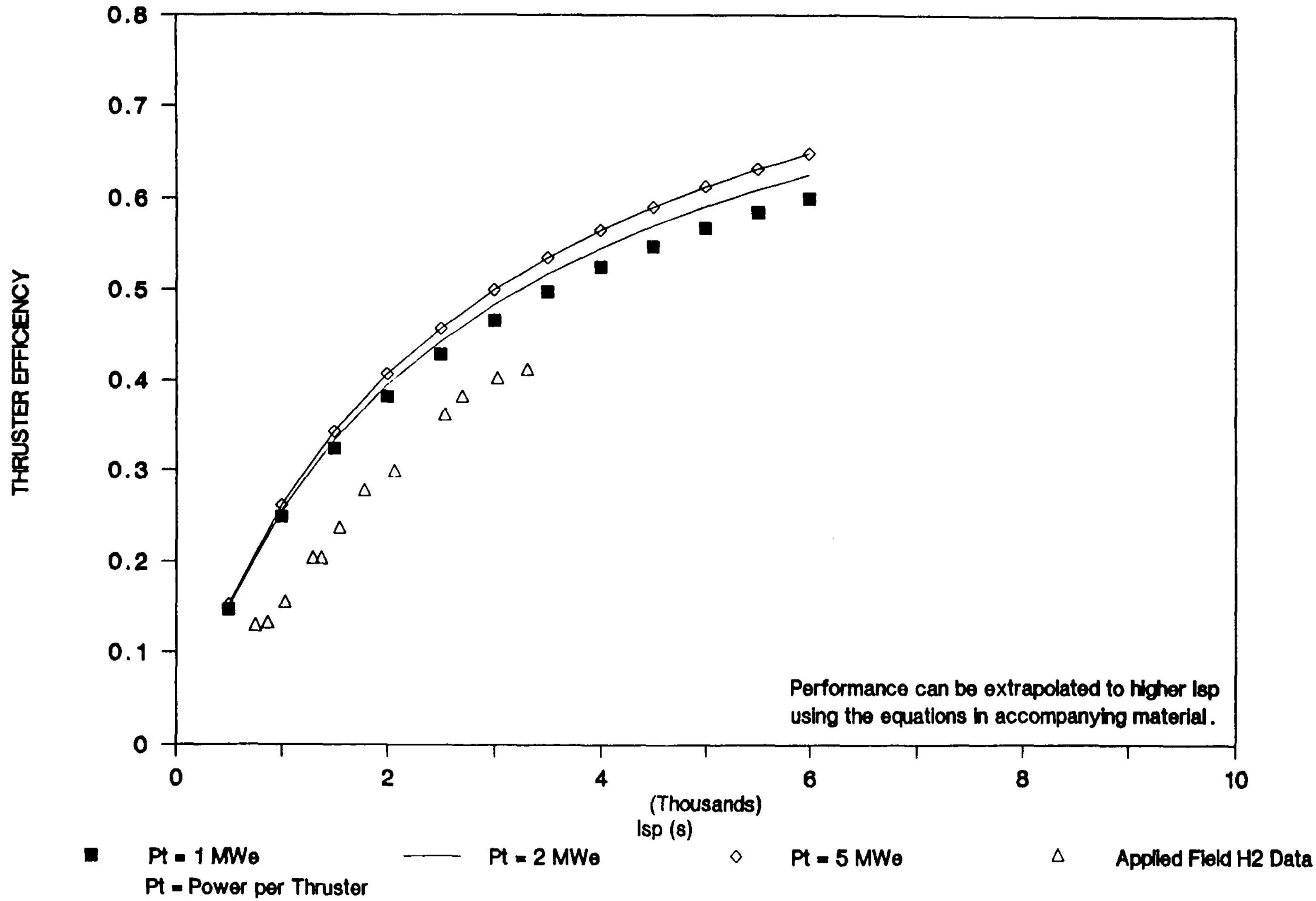


FIGURE 3

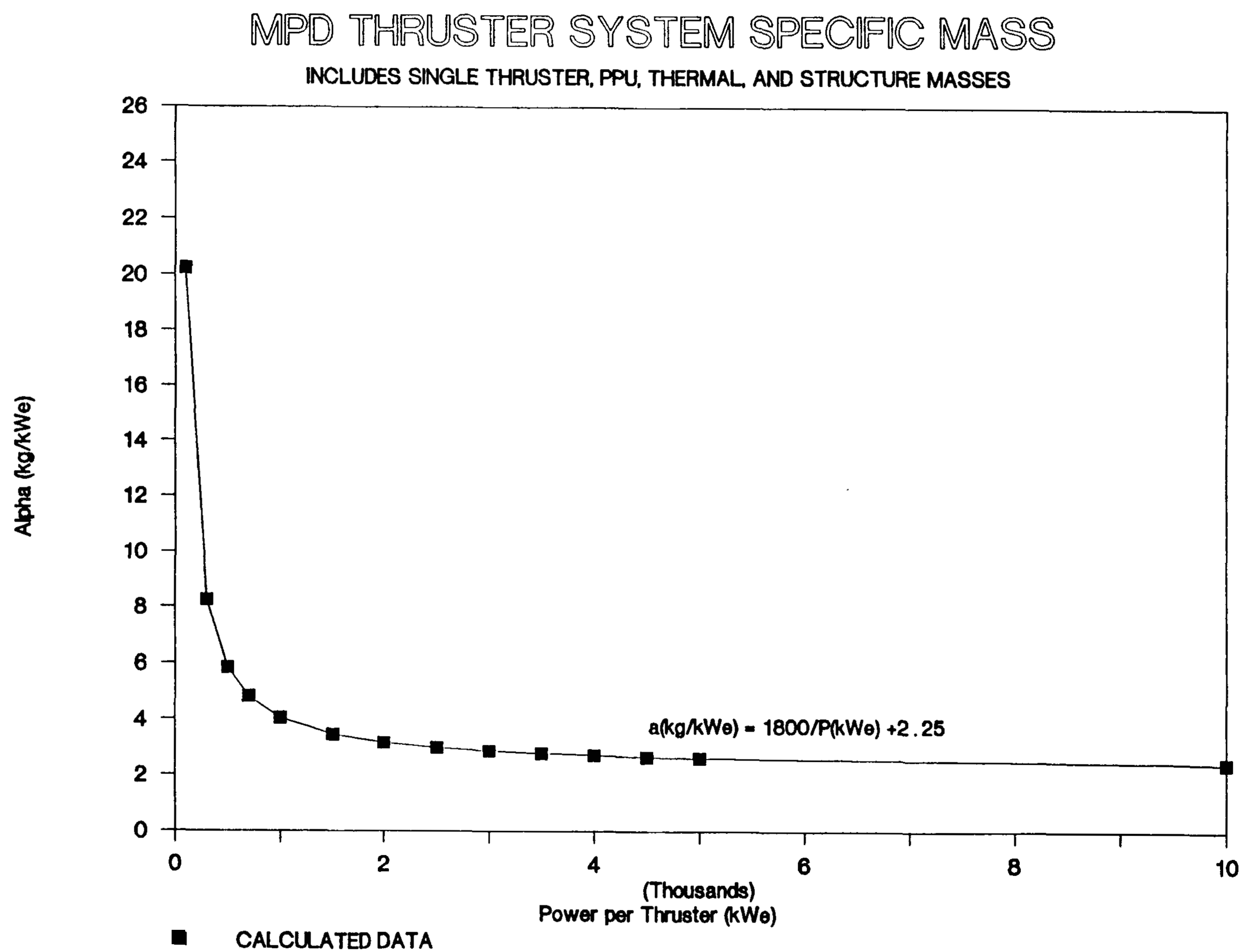

MPD THRUSTER SYSTEM SPECIFIC MASS
INCLUDES SINGLE THRUSTER, PPU, THERMAL, AND STRUCTURE MASSES
Alpha (kg/kWe)
26
24
22
20
18
16
14
12
10
8
6
4
2
0
a(kg/kWe) = 1800/P(kWe) +2.25
0
2
4
6
8
10
(Thousands)
Power per Thruster (kWe)
CALCULATED DATA

FIGURE 4

NUCLEAR REACTOR POWER SYSTEMS

Lee Mason

The system designs presented are based upon the growth of the SP-100 reactor and the use of an advanced Stirling cycle power conversion system. There are two designs, one for a manned vehicle, the other for an unmanned cargo vehicle which will have some limited human interaction. Accordingly, we have used conservative but plausible 4π shielding designs for the reactor.

The SP-100 is the only known reactor program in development which has civilian space application. It seems likely that if the SP-100 is built, it will be the only space based nuclear reactor for some time. The SP-100 is designed for surface power and lower power applications not as a power source for electric propulsion. (The authors understand that very high power reactors are under study by the Department of Defense. However these concepts will not be considered in our studies.) Further analysis will be performed and incorporated as better shielding performance and design data become available.

Table 3 presents the design assumptions made in order to generate[11,12] the data in Table 4. Figure 5 shows a conceptual design of the reactor, payload, and shielding which has the separation characteristics of Table 3. Of particular importance in Table 4 is the overall specific mass of the manned and unmanned designs. These designs reflect a trade-off between radiator size and the operating temperature ratio to achieve the desired electrical output. The reader is cautioned not to extrapolate these data to other power levels.

SP-100 Power for Nuclear Electric Propulsion

- SP-100 type Liquid Metal-Cooled Reactor
- Advanced Technology Stirling Cycle Conversion

Stirling Cycle Parameters:

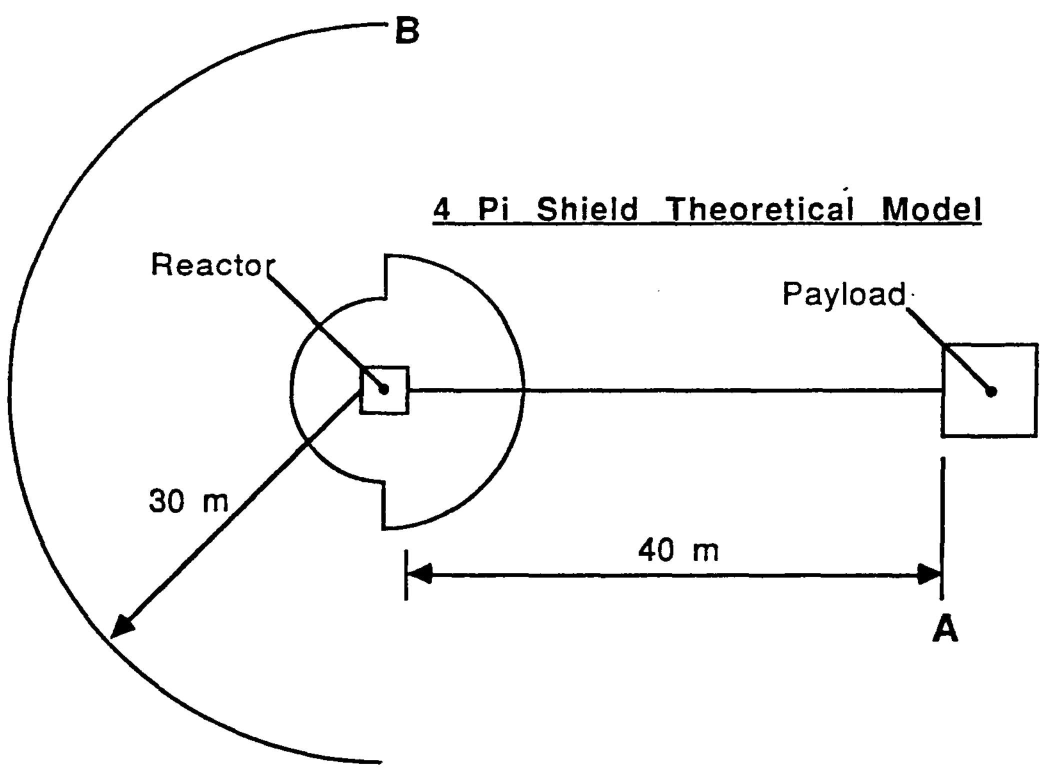
Stirling Heater Temperature (K)	1300
Sink Temperature (K)	250
No. of Engines/2500 kWth	8
No. of Engines/No. of Operating Engines	1.14

Shielding Parameters:

4-Pi Shielding	
Seperation Distance @A (meters)	40
Seperation Distance @B (meters)	30

	DR (mrem/hr) (@A)	DR (mrem/hr) (@B)	time (days)	Total Dose (rem) (@A)
Manned Vehicle	7	200	365	60
Unmanned Vehicle	42	200	5 *	5

* Manned Proximity Operations of no more than 5 days



				MANNED		UNMANNED	
Reactor Power (kWth)	Temperature Ratio	Alternator Power (kWe)	Pwr System Mass (kg)	Shield Mass (kg)	Sys Sp Mass (kg/kWe)	Shield Mass (kg)	Sys Sp Mass (kg/kWe)
2500	2.0	593.8	9574	9137	31.5	3428	21.9
2500	2.1	632.6	10444	9137	31.0	3428	21.9
2500	2.2	667.9	11390	9137	30.7	3428	22.2
2500	2.3	700.2	12425	9137	30.8	3428	22.6
2500	2.4	729.7	13566	9137	31.1	3428	23.3
2500	2.5	756.9	14832	9137	31.7	3428	24.1
5000	2.0	1187.5	18944	10515	24.8	4141	19.4
5000	2.1	1265.2	20683	10515	24.7	4141	19.6
5000	2.2	1335.8	22576	10515	24.8	4141	20.0
5000	2.3	1400.3	24647	10515	25.1	4141	20.6
5000	2.4	1459.4	26928	10515	25.7	4141	21.3
5000	2.5	1513.8	29460	10515	26.4	4141	22.2
7500	2.0	1781.3	28352	11385	22.3	4601	18.5
7500	2.1	1897.8	30962	11385	22.3	4601	18.7
7500	2.2	2003.8	33801	11385	22.5	4601	19.2
7500	2.3	2100.5	36906	11385	23.0	4601	19.8
7500	2.4	2189.1	40328	11385	23.6	4601	20.5
7500	2.5	2270.7	44127	11385	24.4	4601	21.5
10000	2.0	2375.0	37748	12031	21.0	4949	18.0
10000	2.1	2530.4	41228	12031	21.0	4949	18.2
10000	2.2	2671.7	45014	12031	21.4	4949	18.7
10000	2.3	2800.6	49155	12031	21.8	4949	19.3
10000	2.4	2918.9	53717	12031	22.5	4949	20.1
10000	2.5	3027.6	58782	12031	23.4	4949	21.1

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13.) E.Y. Choueiri, A.J. Kelly, R.G. Jahn, "MPD Thruster Plasma Instablity Studies", AIAA-85-1067, May 1987.

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APPENDIX A

MPD THRUSTER PERFORMANCE MODEL

This is an empirical model based on current (J), voltage (V), thrust (T), and mass flow rate ($\dot{m}$) behavior reported by Tahara, et. al.[8] for an applied field, quasisteady MPD thruster using H_2 propellant. Theoretical formulations originally derived for self-field MPD devices[16] were adapted to approximate both applied field and self field performance to project thrust efficiencies to Isp levels not experimentally measured.

The model is derived on the basis of the existence of an Alfven limiting velocity in partially ionized gases flowing across a magnetic field. MPD thruster behavior is therefore separated into two regimes: "partially ionized", and "fully ionized". In the "partially ionized" regime, thrust and voltage are seen to be linear with respect to J. In the "fully ionized" regime, thrust is quadratic with J ($T=bJ^2$,where b = thrust coefficient), and voltage increases as the cube of J. This behavior has been seen empirically in quasisteady self-field devices, and the data reported by Tahara for both self and applied field thrusters shows the same qualitative characteristics. This approach was therefore chosen to serve as a model for both devices.

In examining the self- and applied field characteristics (Figures A1-A3), several observations can be made. First, for a given propellant (such as H_2), an applied field has relatively little effect upon the voltage - current behavior of the device. Second, application of an axial field increases the thrust magnitude over the self-field case but maintains the same behavior with respect to current. Third, in the case of H_2, thrust and voltage are primarily linear with current over the range of currents tested; this phenomenon will be discussed in the theoretical section. Thrust and voltage data taken using ammonia propellant, however, show the expected self-field behavior at high currents. This fact indicates the effectiveness of considering self field effects for both types of thruster, particularly at high currents.

It is important to note that although the basic behavior of this model is based on a physical theory, the final equations and coefficients are empirical and are chosen to fit the data.

THEORY OF MPD PERFORMANCE

MPD thruster operation can be divided into "partially ionized" and "fully ionized" regimes on the basis of the Alfven critical velocity[10]:

$$U_A = \sqrt{\frac{2eV_i}{M}\left(2 + \frac{eV_d}{eV_i}\right)} \tag{1}$$

where eV_i = ionization energy of H and eV_d = dissociation energy of H_2

Hydrogen is the propellant of interest due to its high Isp potential. For H_2, $U_A \sim 7.7 \times 10^4$ m/s, corresponding to an Isp of 7850 seconds. The reference data used in this model extends only to ~3000 s, still well in the "partially ionized" regime. The quadratic behavior of thrust as well as the associated change in voltage at higher currents and Isp may be inferred from similar data taken with ammonia propellant which shows such a transition at lower currents. For H_2, the flow is thus thought to be "partially ionized" at Isp below 7850 s, and "fully ionized" at higher Isp.

Approach:

Equations describing thrust and voltage behavior with current have been combined to yield thruster efficiency in terms of specific impulse and power input to the thruster (Pt). These were deemed to be the parameters of interest to the mission analysts who are expected to use these projections. Prior to developing these equations, some terms will be defined that will be used extensively in the derivation.

J_{fi} = "Fully Ionized Current" = Current level where plasma reaches fully ionized regime, defined as

$$J_{fi} = \sqrt{\frac{\dot{m}U_a}{b_T}}, \tag{2}$$

b_T = Quadratic thrust coefficient obtained from experimental data.

c = Exhaust velocity = g*Isp, g = 9.81 m/s^2

Pt = Electric power input to the thruster

η = Thrust efficiency = $\frac{\dot{m}c^2}{2Pt}$

Taking each region of operation separately:

I.) "Partially Ionized" - c < U_A:

$$T = b_T J_{fi} J \tag{3}$$

$$V = \frac{b_T U_A J}{2} + \frac{b U_A J_{fi}}{2} + V_t \tag{4}$$

where b = self field thrust coefficient obtained from data

V_t= Thermal or fall voltage, assumed constant

In the case of the applied field thruster, $b_T > b$. For a self field device, $b_T = b$. In general, the effect of the applied field is to increase the thrust levels over the self field case (increase b_T) without changing the voltage appreciably. Therefore, the thrust energy term in the voltage, $\frac{b_T U_A J}{2}$, depends on the applied field thrust (and so b_T), while the ionization power term $\frac{b U_A J_{fi}}{2}$ is taken to depend on the self field b, which is more closely related to thruster dimensions. This assumption was seen to be necessary in order to obtain calculated behavior which agreed favorably with the data.

From equations (2), (3), (4):

$$\dot{m} = \frac{2\eta Pt}{c^2} \qquad J = \frac{T}{b_t J_{fi}} = \frac{2\eta Pt}{b_T U_A} \qquad J_{fi} = \frac{2\eta Pt U_A}{b_T c^2} \tag{5}$$

$$Pt = JV = \frac{b_T U_A J^2}{2} + \frac{b U_A J_{fi} J}{2} + V_t J \tag{6}$$

$$= \eta Pt(1 + Z\Big(\frac{U_A}{C}\Big)) + V_T\sqrt{\frac{2\eta Pt}{b_T U_A}} \tag{7}$$

where $Z = \frac{b}{bT}$. Let $X = 1 + Z(\frac{U_A}{c})$, and $Y = \frac{V_T^2}{(b_T U_A)}$; rearranging in terms of η yields a quadratic equation:

$$\eta^2(PtX^2) - 2\eta(PtX + Y) + Pt = 0$$

solving for η:

$$\eta = \frac{(Pt(1+Z(\frac{U_A}{c})) + \frac{V_t^2}{b_T U_A}) - \sqrt{(Pt(1+Z(\frac{U_A}{c})) + \frac{V_t^2}{b_T U_A})^2 - Pt^2(1+Z(\frac{U_A}{c}))^2}}{Pt(1+Z(\frac{U_A}{c}))^2} \tag{8}$$

The negative root is the realistic ($\eta < 1$) solution.

II.) "Fully Ionized" - $c > U_A$:

In the absence of data in this regime, a fully ionized, self field acceleration process will be assumed by way of inference from the quadratic behavior seen in NH_3 MPD thruster data[8]. Therefore a fully ionized, self field acceleration process will be assumed. For this case, the pertinent equations are

$$T = b_T J^2 \tag{9}$$

$$V = \frac{b_T^2 J^3}{2\dot{m}} + \frac{\dot{m} U_A^2}{2J} + V_T \tag{10}$$

in this regime, $J = \sqrt{\frac{2\eta Pt}{b_T c}}$.

Power input to the thruster becomes

$$Pt = J \cdot V = \eta P(1 + \left(\frac{U_A}{c}\right)^2) + V_T\sqrt{\frac{2\eta Pt}{b_T c}} \tag{11}$$

Now, for some continuity between the two regimes, the power will be required to be continuous at the transition point, $J = J_{fi}$, $U_A = c$, using both formulations. This requires the (arbitrary) introduction of a factor Z into the fully ionized equation:

$$Pt = \eta Pt\left(1 + Z\left(\frac{U_A}{c}\right)^2\right) + V_T\sqrt{\frac{2\eta Pt}{b_T c}} \tag{12}$$

Now for $X = 1 + Z(\frac{U_A}{c})^2$, $Y = \frac{V_t^2}{b_T c}$, we have the same form for the efficiency quadratic:

$$\eta^2(PtX^2) - 2\eta(PtX + Y) + Pt = 0$$

and thus the efficiency becomes:

$$\eta = \frac{\left(Pt\left(1+Z\left(\frac{U_A}{c}\right)^2\right)+\frac{V_T^2}{b_T c}\right)-\sqrt{(Pt(1+Z(\frac{U_A}{c})^2)+\frac{V_t^2}{b_T c})^2-Pt^2(1+Z(\frac{U_A}{c})^2)^2}}{Pt(1+Z(\frac{U_A}{c})^2)^2} \quad (13)$$

The Z coefficient in the second equation is artificially introduced, but it allows a continuous transition between the two regimes at the expense of a change in slope of the efficiency-Isp function at the interface. This is only true when basing the performance on applied field data; in a self field device where Z = 1, the function is smooth.

DATA INPUT TO THE EQUATION:

Performance data for Tahara's applied field MPD thruster yielded the following parameters using linear regression to thrust and voltage characteristics:

b	= .068 N/kA^2 (from self field data)
b_T	= .18 N/kA^2 (from best applied field performance)
V_t	= 30 V (from corresponding applied field voltage data)

For projected MPD thruster performance, the parameters used are:

b	= .068 N/kA^2
b_T	= .2 N/kA^2 (Princeton quasisteady Benchmark value)
V_t	= 15 V (Qualitative assumption of engineering improvements)

Substituting these projected parameters into the η-Isp equations yields the functions:

$c < U_A$:

$$\eta = \frac{Pt(1+\frac{2.62\times10^4}{c})+1.46\times10^4-\sqrt{(Pt(1+\frac{2.62\times10^4}{c})+1.46\times10^4)^2-Pt^2(1+\frac{2.62\times10^4}{c})^2}}{Pt(1+\frac{2.62\times10^4}{c})^2}$$

$c > U_A$:

$$\eta = \frac{Pt(1+\frac{2.0\times10^9}{c^2})+\frac{1.13\times10^9}{c}-\sqrt{(Pt(1+\frac{2.0\times10^9}{c^2})+\frac{1.13\times10^9}{c})^2-Pt^2(1+\frac{2.0\times10^9}{c^2})^2}}{Pt(1+\frac{2.0\times10^9}{c^2})^2}$$

The results using these parameters are shown in Figure 3 compared to the present applied field H_2 experimental data. The figure extends to 6000 s Isp, which is below the transition point to a self-field form of operation because of the lack of experimental data beyond this point; for the purpose of mission analysis, the above equations can be used to extrapolate MPD thruster performance to values up to 10000 seconds. The relation for $c < U_A$ results in behavior for the MPD thruster that is reasonably close to that seen in experiments; the equations used to extrapolate to $c > U_A$ have been applied to quasisteady self-field devices with marked success[16]. Further investigation of the acceleration processes and loss mechanisms of MPD thrusters is required to develop a definitive model; for the present, the above equations may serve in mission analyses pending resolution of the unknowns in MPD thruster design and operation.

APPENDIX B

ION ENGINE PERFORMANCE CALCULATIONS

Calculated ion engine performance is based on operating conditions demonstrated in the laboratory at power levels up to 20 kWe. To first order, the performance is independent of input power and engine size. That is, Isp and η are primarily functions of the accelerating potential, which does not depend on power or size. Thrust and power levels are functions of the area of the electrodes due to space charge effects in these thrusters[14].

Experimental data used in the performance calculations are summarized in Table B-1. Calculations and assumptions were provided by Michael Patterson of the Low Thrust Propulsion Branch of the Space Propulsion Technology Division at NASA Lewis Research Center.

Table B-1. Assupmtions for Ion Engine Performance Calculations

	Propellant	
Parameter	Krypton	Argon
Perveance Amperes	$Jb = \frac{(Area)\times 5.3\times 10^{-5}\times (Vt)^{2.2}}{\sqrt{M}}$	
Total Accelerating Voltage, volts	2000, 2500, 3000*	
R-ratio of optics	0.55 - 0.85 inclusive (2 grid)	
Discharge Chamber Propellant Efficiency	0.950	0.920
Total Propellant Efficiency	0.922	0.894
Estimated doubly-charged Ion Current Ratio	0.068	0.053
Thrust Loss Factor due to Beam Divergence	0.980	0.980
Thrust Loss Factor due to Doubly-charged Ions	0.981	0.985
Fixed Power Loss (kWe)	0.050	0.050
Discharge Losses (W/A)	115	135
Neutralizer Mass Flow Rate, equivalent Amperes	$0.0316 \times Jb$	

* grid-gap 0.60 mm at 2000-2500 v; 0.866 mm at 3000 v with perveance expression modified for larger gap.

APPENDIX C

Xe is currently priced at approximately \$550/kg[15], which is about $\frac{1}{4}$ the optimistic Shuttle launch cost estimate of \$2,000/kg. Argon is more abundant and operates at higher Isp levels at a cost of only \$30/kg, ~1% of its launch cost. Furthermore, a study by Graeme Aston of JPL for a solar powered, ion engine propelled lunar ferry calculated that 4 300 kWe ferries would require 20 metric tons (mT) of Xe annually, which is greater that the total annual production of the Western World[16]. This quantity of Xe would also cost \$16.5 million at today's prices.

Code Z Vol. III

"SP-100 Power System Conceptual Design for Lunar Base Applications"

L. S. Mason, Advanced Space Analysis Office
H. S. Bloomfield, Power Technology Division
NASA Lewis Research Center

The following is a presentation package describing the results of the above study. This package was presented at the Office of Exploration July program review and subsequently to the Propulsion, Power, and Energy Division in the Office of Aeronautics and Space Technology on July 21, 1988. Included in this package is detailed information of the component configurations that were considered as well as the rationale for the final choices. Also included is an evaluation of nuclear power system impacts on a mature lunar base.

The goal of this study was to develop a conceptual design of a nuclear power plant for use on the lunar surface. To more fully understand the interactions between the lunar base and the reactor power system, a possible lunar base infrastructure was defined by the Office of Exploration Surface Systems Integration Agent.

The nuclear power plant concept consists of a 2.5 MW_{th} SP-100 reactor coupled to eight free-piston Stirling engines. The design power level of the system is 825 kWe and the system mass is 20 metric tonnes.

SP-100 Power System Conceptual Design for Lunar Base Applications

Lee S. Mason / Advanced Space Analysis Office
Harvey S. Bloomfield / Power Technology Division

NASA Lewis Research Center
Cleveland, Ohio

Presentation to Code RP
July 21, 1988

Study Goals

- Objective
  - Develop a conceptual design of a nuclear power system utilizing an SP-100 reactor and Stirling engine conversion for operation on the lunar surface.
  - Select system configurations which enable and/or enhance the lunar base mission.
  - Compare system components and coupling options for safety implications, high performance, low mass, and ease of assembly.

- Products
  - Power system performance and sizing data including recommended power system and radiator configurations and shielding options.
  - Lunar base layout rationale addressing the power system interface with the base's installations and activities.
  - Identification of nuclear power system impacts on the Code Z "Case Studies".
  - NASA Tehnical Memorandum in Oct/Nov.

Methodology

- **Background**
  - 3 month study performed in-house.
  - Joint effort of Advanced Space Analysis Office (ASAO) and Power Technology Division (PTD).
  - Code RP request for performance and sizing data to support artist's rendering of SP-100 nuclear power system on lunar surface.
  - Related to Code Z "Case Studies" through evaluation of nuclear power system impacts on advanced lunar base.

- **Approach**
  - Utilize PTD experience with SP-100 reactor design and Stirling engine development to determine power system performance, sizing, and layout.
  - Coordinate power system design with a typical mature lunar base as proposed by the Code Z Surface Systems Integration Agent and ASAO.
  - Establish common assumptions and groundrules with the Code Z Integration Agent studies.

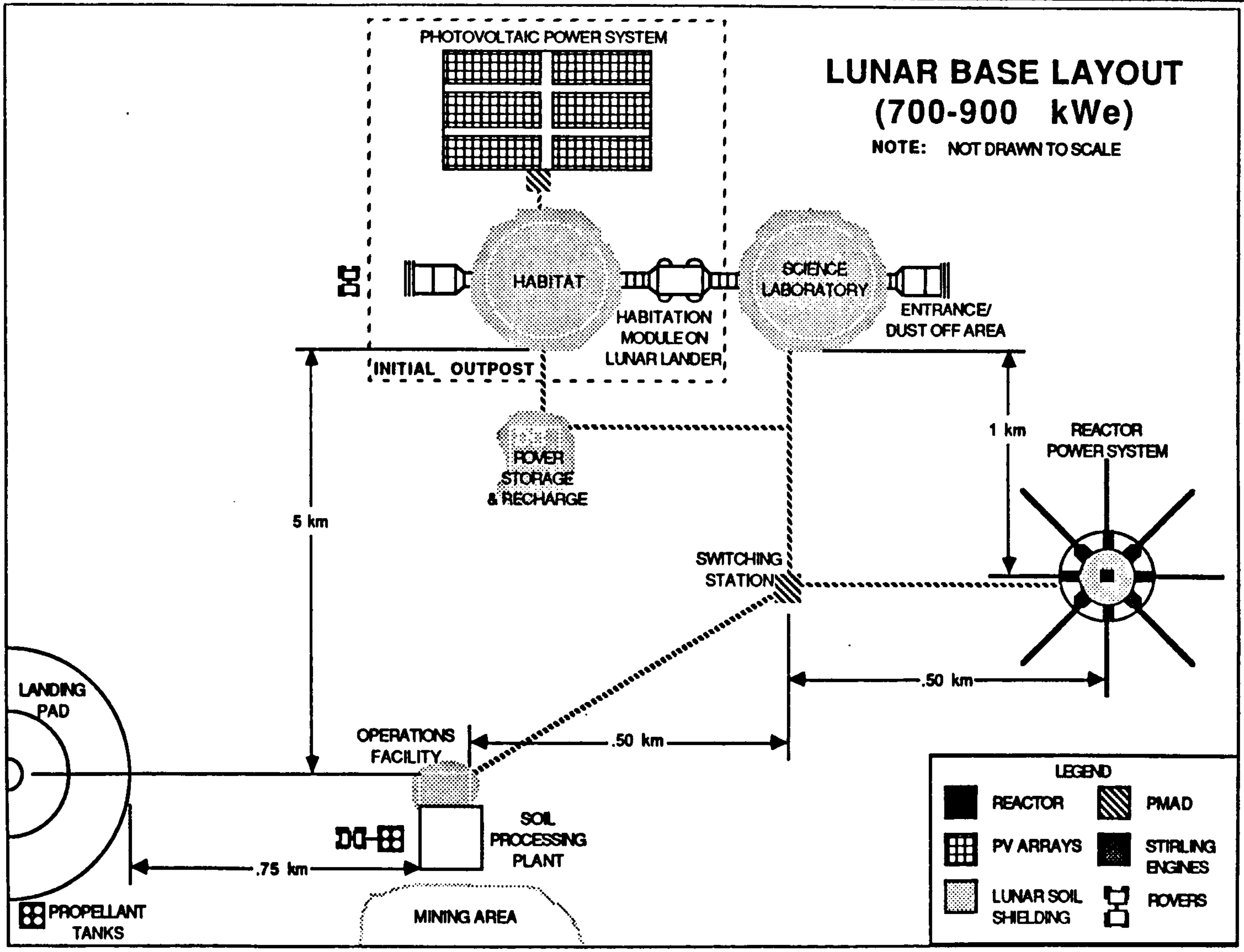
Methodology (cont.)

- Key Assumptions/Groundrules
  - Evolutionary (i.e. advanced) lunar base with power requirements in the 700-900 kWe range.
  - Sufficient rover arsenal for construction and maintenance.
  - Availibility of advanced technologies including free-piston Stirling engines, SP-100 type reactor, and mercury heat pipes.
  - Nuclear power system supplies electrical power only; direct thermal power and/or waste heat usage could be examined in future studies.
  - Reactor shielding must be designed to meet safety requirements associated with manned presence.

NASA
Lewis Research Center
Space Flight Systems Directorate

Lunar Base Top View

ASAO



ADVANCED SPACE ANALYSIS OFFICE

NASA
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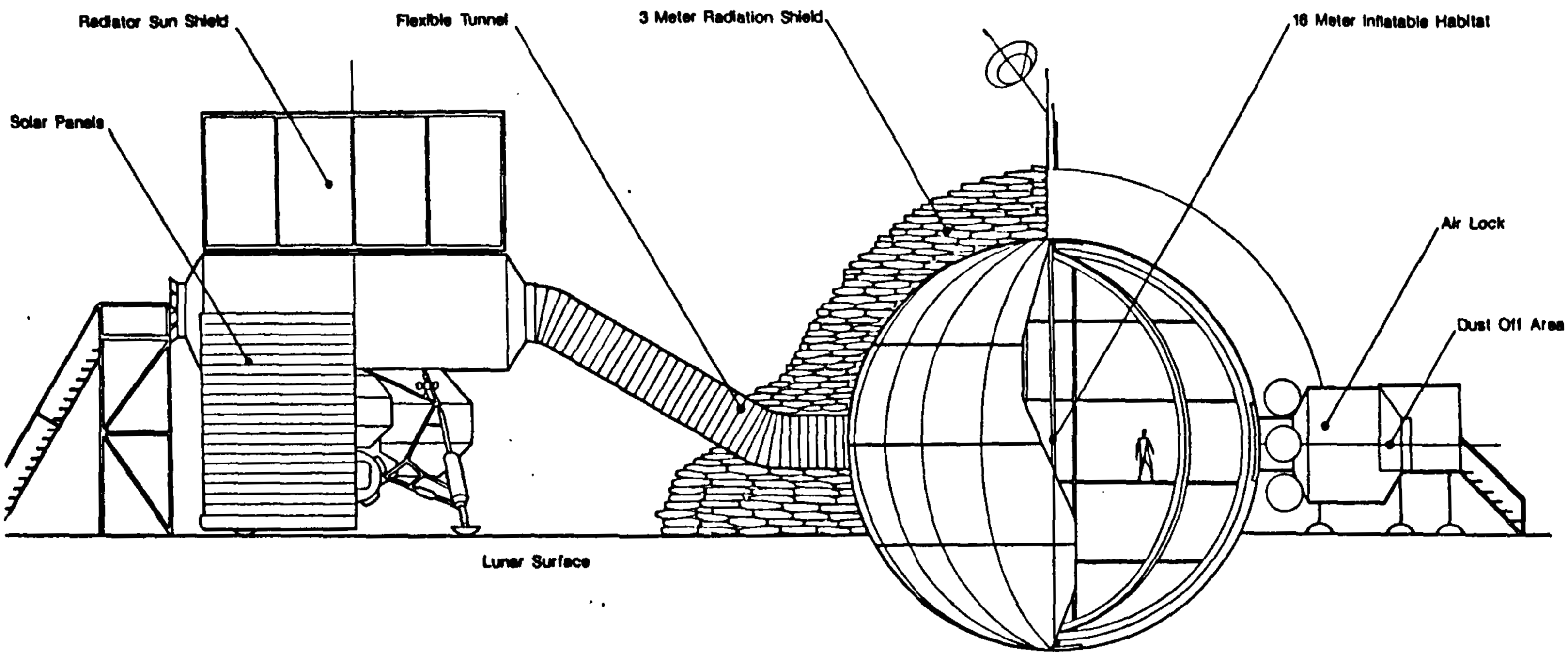
Inflatable Habitat/Initial Outpost

ASAO



Source: "Inflatable Habitation for the Lunar Base", M.L. Roberts (NASA-JSC)

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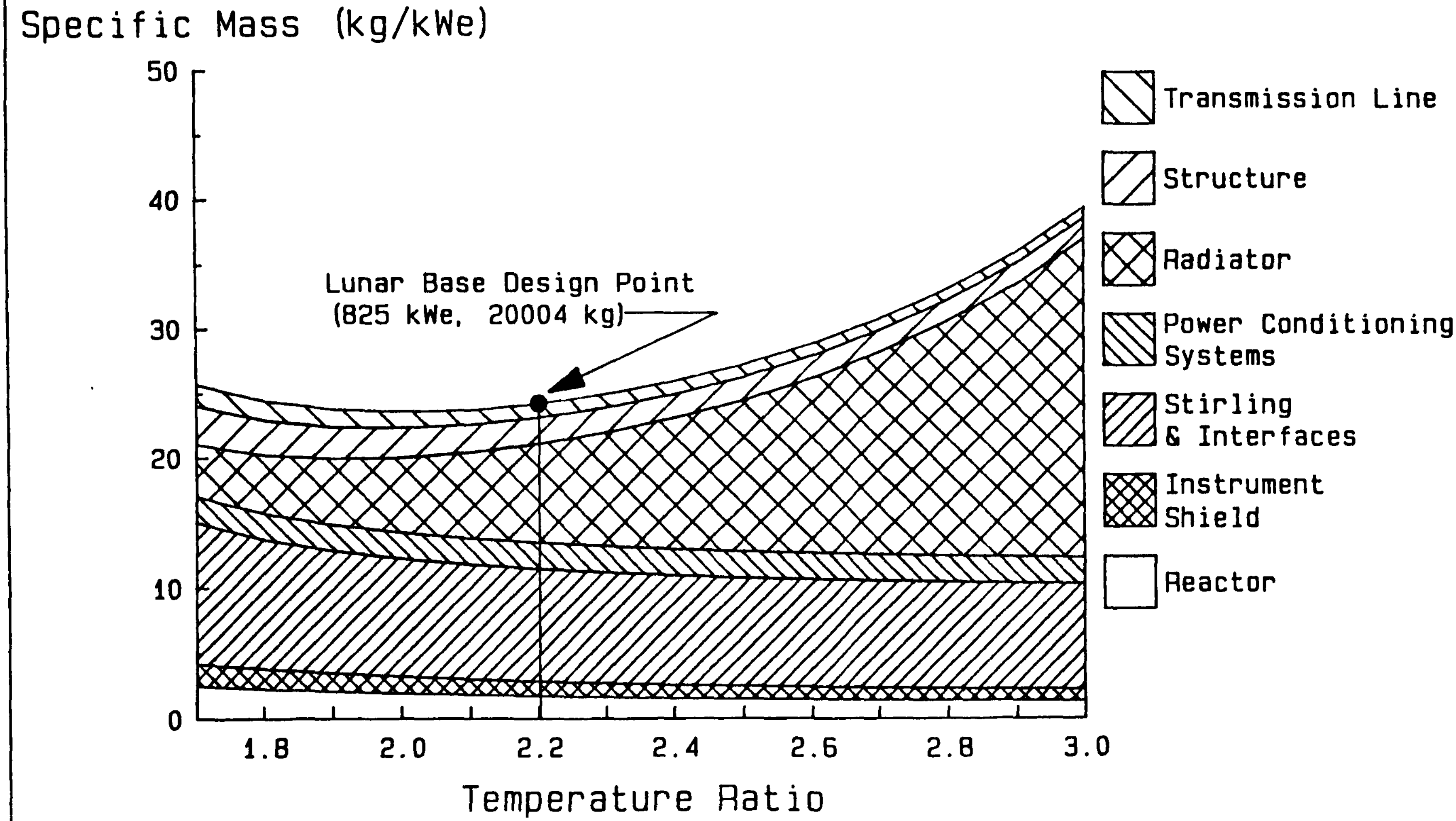
Initial Outpost Power

- Amorphous Silicon Arrays - Technology Goals
  - 15% cell efficiency
  - 90% packing factor
  - 300 W/kg specific mass (cells, kapton substrate, and interconnect wiring)
  - Roll-Up storage capability
- Regenerative Fuel Cells - Technology Goals
  - 60% round trip efficiency
  - 1000 W-hr/kg energy density
  - High pressure filament wound storage tanks
- Power Management and Distribution
  - 92% Power Conditioning/Distribution efficiency
  - 20 kg/kWe specific mass

Power (kWe)	Array Area (m2)	Array Mass (kg)	100% Night Power RFC Mass (kg)*	Total Mass (kg)
25	397.6	241.5	9130	9872
50	795.2	483.1	18261	19744
75	1192.8	724.6	27391	29616
100	1590.4	966.2	36522	39488

* 336 Hour Lunar Night

Stirling System Component Mass Sensitivity to Temperature Ratio Thot=1300 K, 2.5 MWth



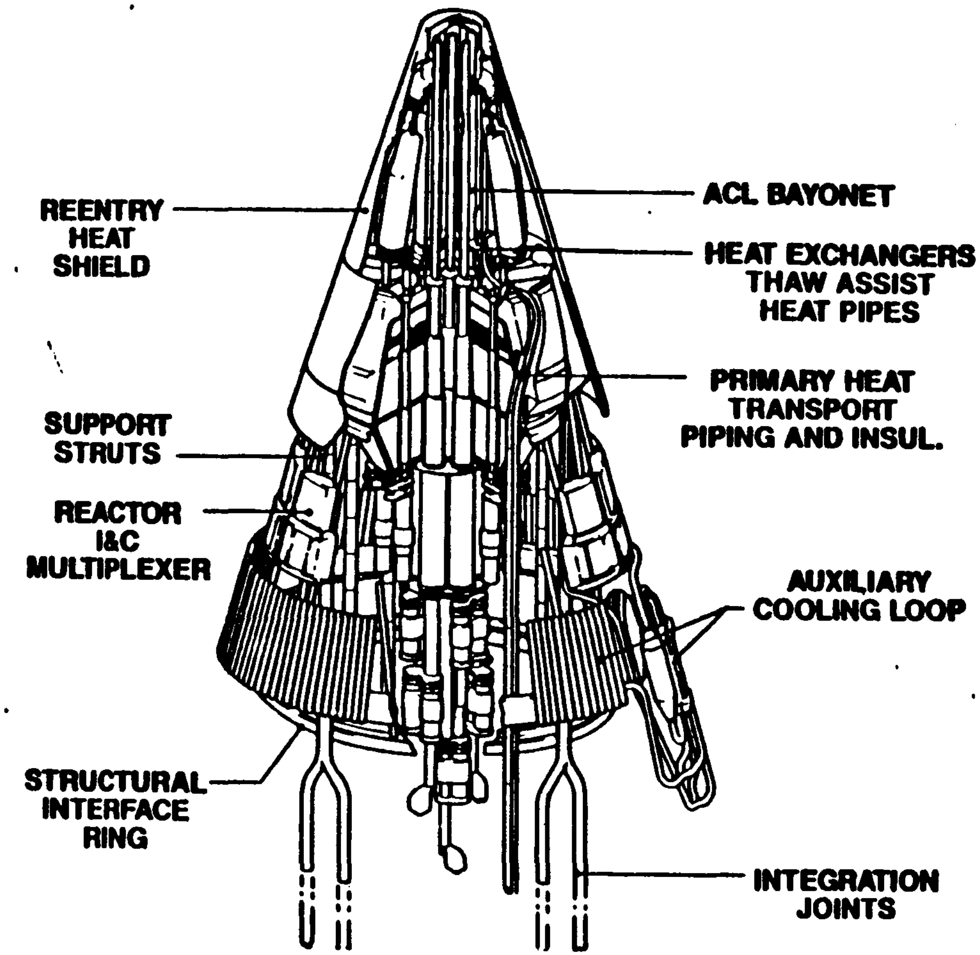
Nuclear Power System Performance Results

Reactor Thermal Power	2500	kWt
Reactor Design Lifetime (@ full power)	7	yrs
Electrical Output (6 of 8 Engines)	825	kWe
Electrical Output/Operating Engine	138	kWe
Rated Electrical Output/Engine	150	kWe
Percent of Operating Engine Capacity	91.7	%
Thermal-to-Electric Efficiency	33.0	%
Stirling Heater Temperature	1300	K
Stirling Temperature Ratio	2.2	
Stirling Cooler Temperature	591	K
Radiator Surface Temperature	525	K
Total Heat Rejected	1675	kWt
Lunar Surface Temp. (w/Apron)	222	K
Lunar Sky Temperature	267	K
Radiator Emisivity	0.85	
Radiator Area (Spoked Wheel)	780	m2

Nuclear Power System Mass

Subsystem	Mass (kg)	Material
Reactor	755	Refractory Metal
Primary Heat Transport	342	Refractory Metal
Instrumentation & Control	359	
Shadow Shield	931	LiH/W
Stirling Engines (8; incl HXs)	5871	Refractory Metal
Reactor/Stirling Manifold (Dual)	423	Refractory Metal
Heat Rejection Loops (8)	648	Stainless Steel
EM Pumps (8)	160	Hiperco-27
Accumulators (8)	24	Refractory Metal
Radiator (Spoked Wheel)	6240	Stainless Steel/ Mercury Heat Pipes
Structure		
Reactor Excavation Bulkhead	679	Aluminum
Engine Support Platforms (8)	1005	Carbon-Carbon
AC-DC Converter (& PLR)	1650	
Transmission Lines (5051 m)	917	
Total	20004	kg
Total System Specific Mass	24.2	kg/kWe

REACTOR POWER ASSEMBLY



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Lunar Base Reactor Cutaway

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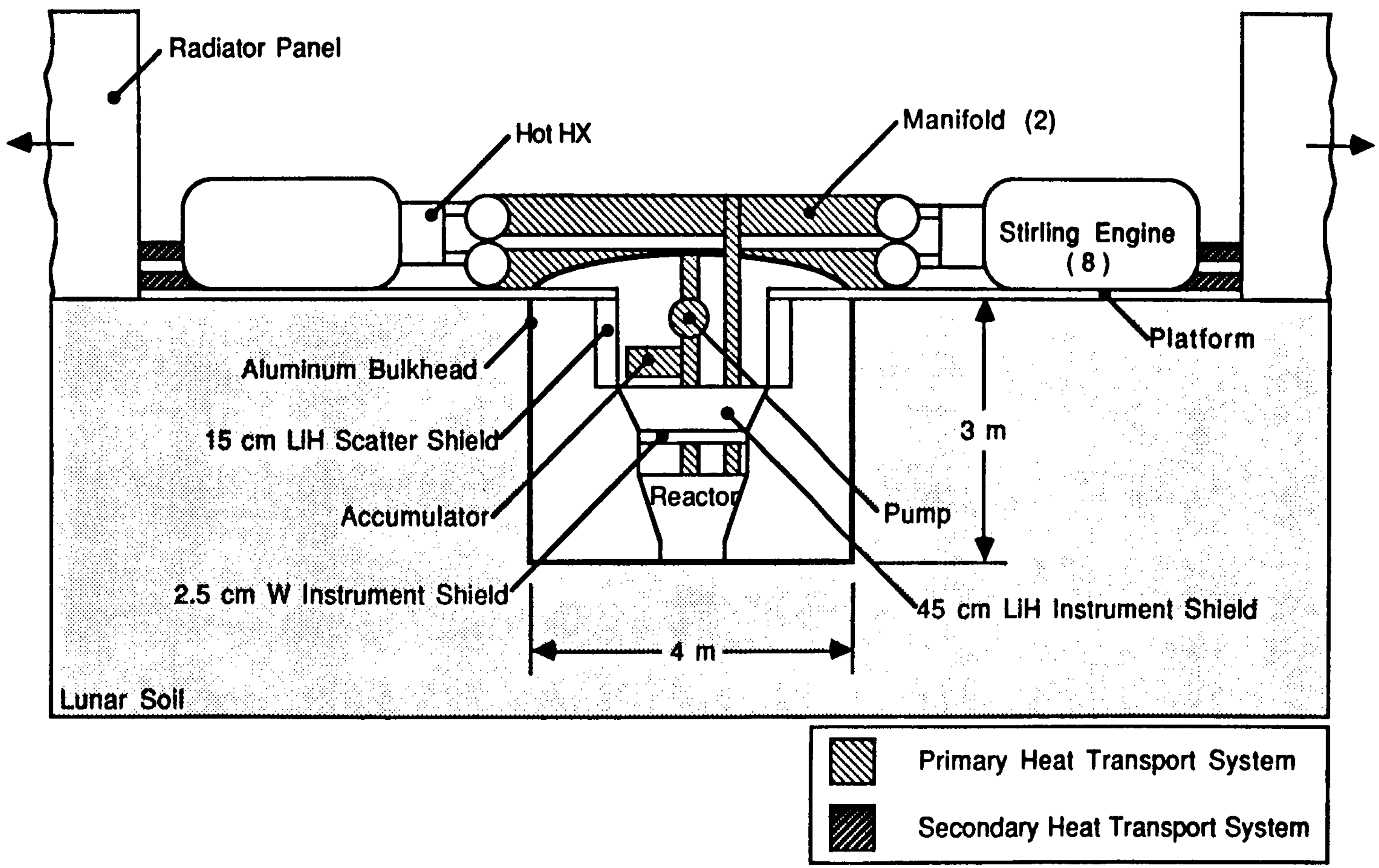
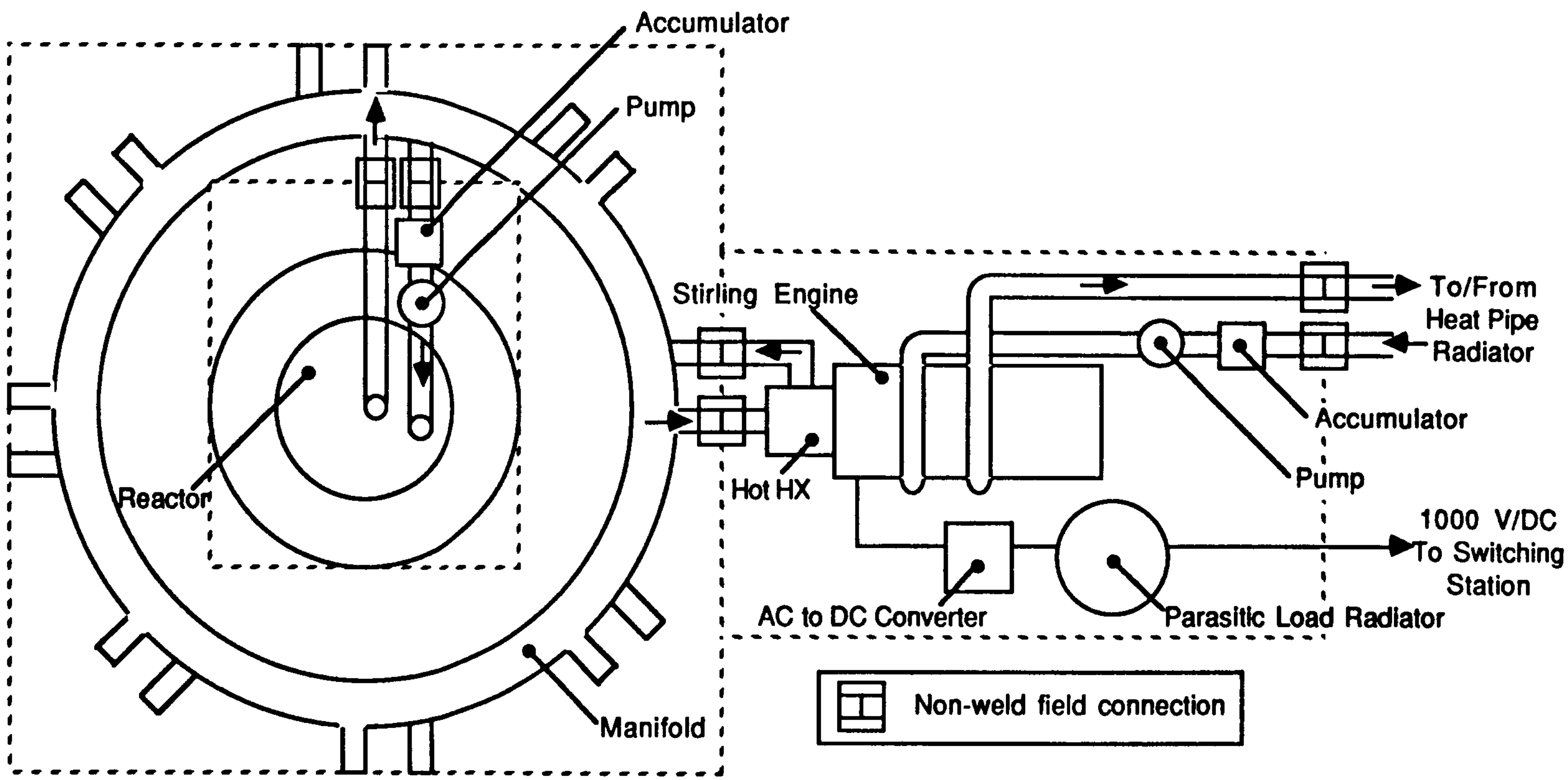


Reactor/Stirling Engine Configuration



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Reactor Shielding Option Comparison

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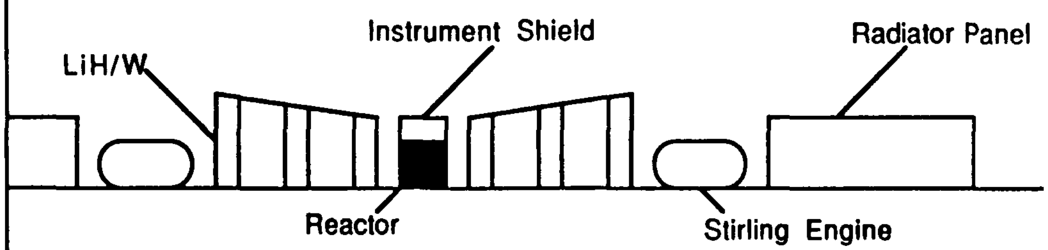
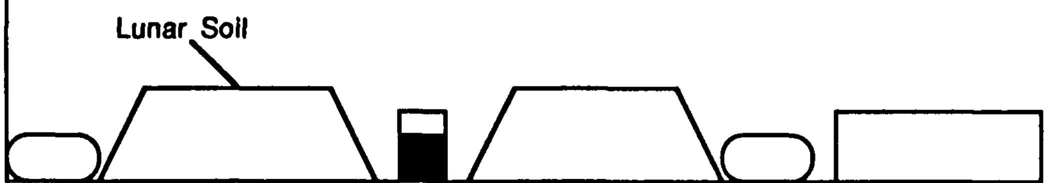


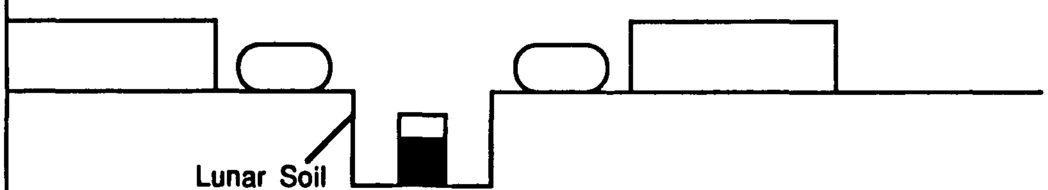
360° Circular LiH/W Shield (from earth)

LiH thickness (2 layers)	35.2 cm
W thickness (3 layers)	19.8 cm
Total thickness	55.0 cm
Total Shield Mass	218 tonnes



360° Circular Lunar Soil Shield

Shield thickness	7 m
Shield height	3 m
Volume of Soil Moved	730 m3
Mass of Soil Moved	870 tonnes



Excavated Cylinder Lunar Soil Shield

Cylinder diameter	4 m
Cylinder height	3 m
Volume of Soil Moved	38 m3
Mass of Soil Moved	45 tonnes

Assumptions: 2500 kWt Reactor
< 5 rem/month at Radiator Panels
Lunar Soil density of 1.2 g/cc

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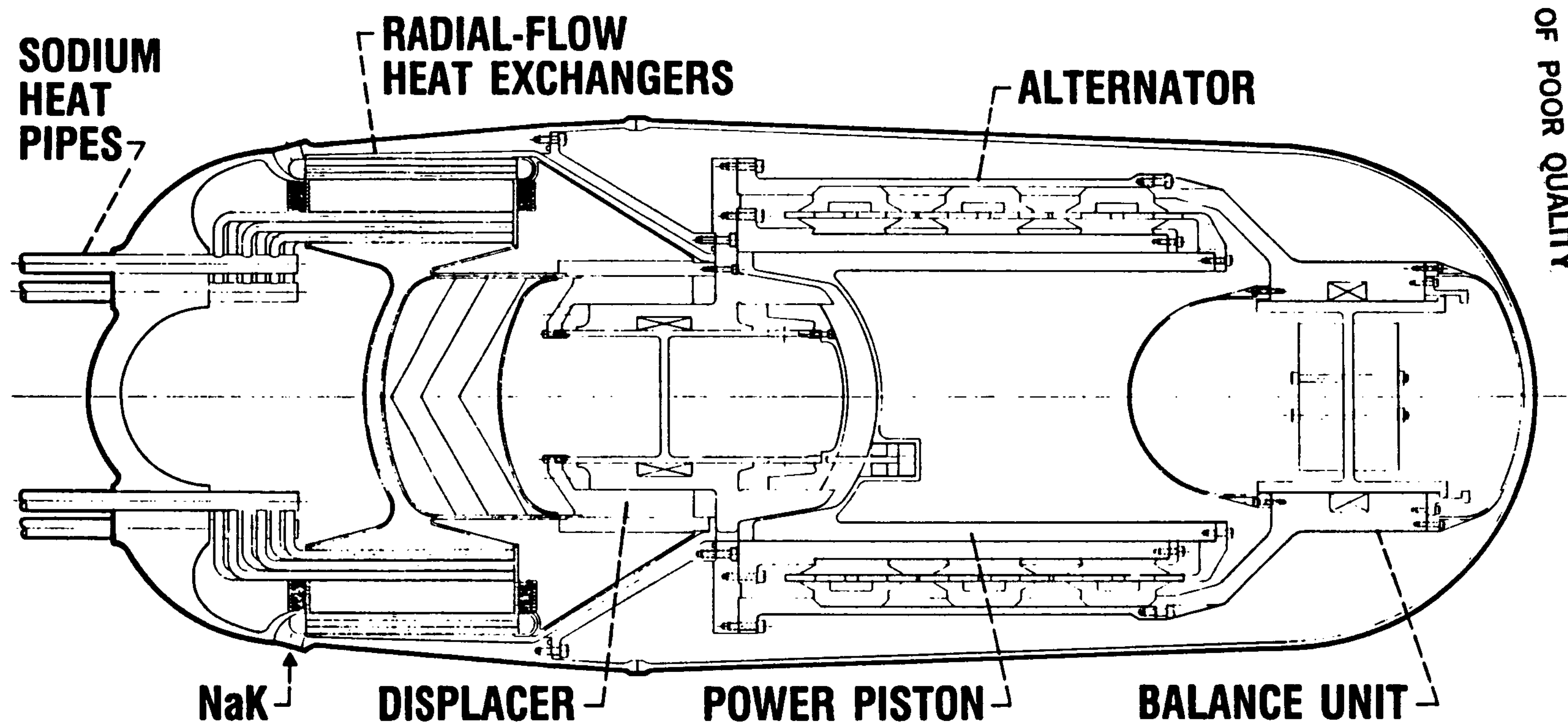
150 kWe FREE-PISTON STIRLING ENGINE/LINEAR ALTERNATOR



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OVERALL SIZE: 30 INCHES DIAMETER (76 cm)
78 INCHES LENGTH (198 cm)

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Radiator Configuration Options

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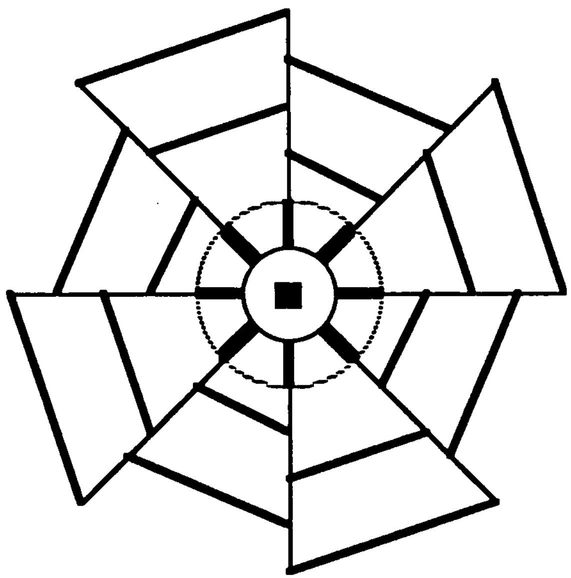

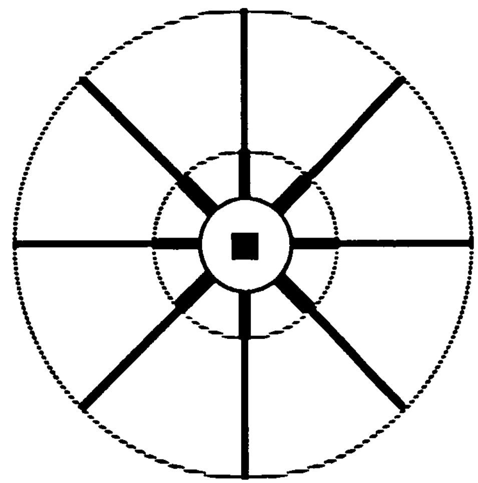

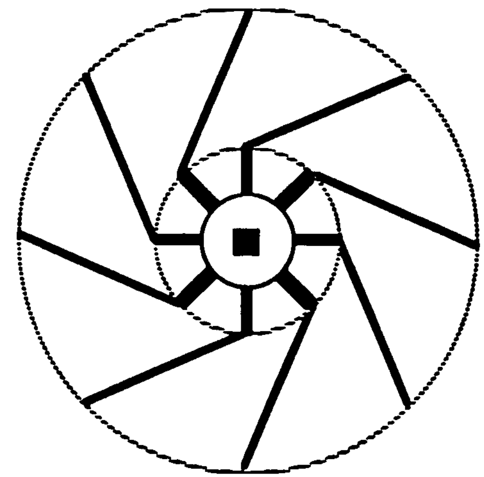

I. Vertical Staggered Circumferential
- 4 rings
- 9 m to 1st panel
- 7 m spacing
- 1.8 m height

TOTAL AREA = 840 m2

II. Vertical Radial Spoked-Wheel
- 5 m to panels
- 25 m long panels
- 1.95 m height

TOTAL AREA = 780 m2

III. Vertical Radial Pin-Wheel
- 5 m to panels
- 25 m long panels
- 1.97 m height

TOTAL AREA = 788 m2

Note: High α/ε thermal apron used to reduce sink temp.

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Heat Rejection Options

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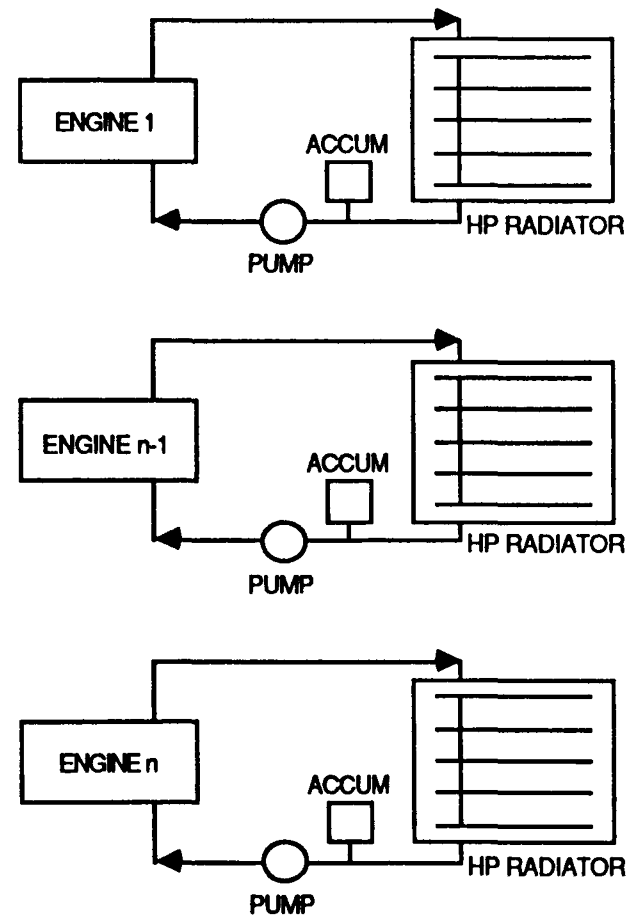

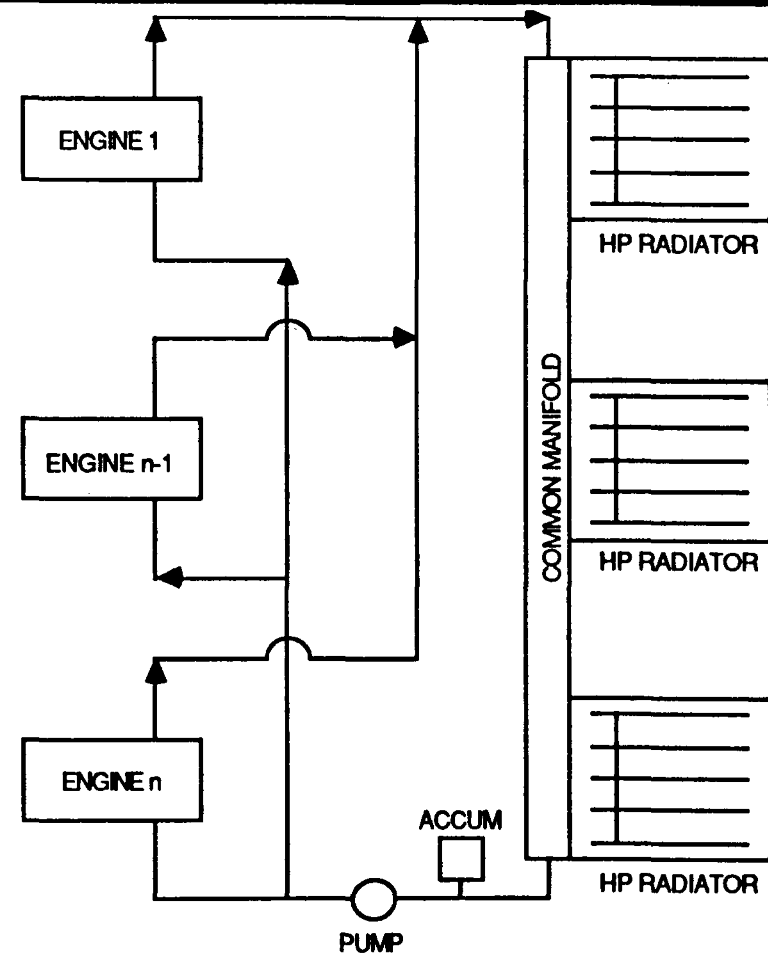

Advantages:	Simple construction	Shared heat rejection
Disadvantages:	1 Pump/Accum per engine	Complex plumbing, Heavier

Power System Design Summary

- <u>Reactor</u>
  - Dual manifold heat transport system designed for easy installation of Stirling engines at non-weld field connections.
  - Aluminum bulkhead provides reactor with a dust-free enviroment.
- <u>Shielding</u>
  - Excavated cylindrical shield enables position of reactor to be independent of habitat location.
  - Utilization of lunar soil for shielding substantially reduces mass of system.
  - Excavated shield allows astronauts to perform maintanence on radiator panels.
  - Instrument shields included for protection of drive motors

Power System Design Summary (cont.)

- Power Conversion
  - Stirling Cycle Conversion offers high efficiency for increased power (825 kWe) over thermoelectric conversion when coupled to the baseline 2500 kWth SP-100 reactor.
  - Six operating Stirling Engines run at 91.7% of rated capacity and two reserve engines ensure dependable power generation.
  - AC to DC converter allows each engine to be run independently and autonomously.
- Heat Rejection
  - Vertical spoked-wheel radiator panels offer minimum area over considered options.
  - High α/ε apron effectively reduces the lunar surface temperature from 375K to 222K.
  - Independent heat rejection loops for each Stirling engine chosen for simple assembly and easy packaging.
  - Heat pipe radiator offers built-in redundancy.

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Specific Impacts on Code Z "Case Studies"

Photovoltaic/regenerative fuel cell system recommended for lower power requirements of initial outpost

- Safe, reliable power generation
- Relatively quick and easy installation
- Remains available for emergency and/or life support requirements

Nuclear power recommended for higher power levels associated with advanced base

- Power rich for increased operations
  - Processing of lunar materials (propellant production)
  - Scientific experimentation
  - Completely closed-loop life support systems (food production)

Specific Impacts on Code Z "Case Studies" (cont.)

Nuclear power (cont.)

- Substantial mass savings compared to photovoltaic/regenerative fuel cell power systems at multi-hundred kilowatt power levels
  - Nuclear system offers 16:1 specific mass (kg/kWe) savings over PV/RFC system
- Day and night operations without the need for energy storage
  - 336 hour night requires heavy energy storage system with solar power
- Potential use of thermal energy from nuclear power system for Lunar material processing
- Hybrid power generation from nuclear and solar power systems offer increased reliability and safety

Nuclear power enables large scale operations on the Lunar surface.

Nuclear power reduces dependency on earth supplied materials.

Potential Future Studies

Several important studies related to this conceptual design have been identified for possible future investigation:

- <u>General Studies:</u>
  - A design for the nuclear power system to supply thermal power, either from the reactor or from waste heat rejection, for lunar material processing
  - A chronological description of Lunar base power system evolution based on power requirements and technology readiness
  - A design for a similar nuclear power plant on the surface of Mars

Potential Future Studies (cont.)

- <u>Component Trade-Offs:</u>
  - The trade-offs involved in replacing the mercury heat pipes with lower temperature water heat pipes for waste heat rejection
  - A comparison of AC vs. DC transmission for long transmission lines based on user requirements
  - Transmission line mass as a function of power system location for various base layouts and distribution forms
  - The safety and technological implications associated with transmission line deployment for buried and suspended lines

Key Sources of Information

	Title-Author	Organization
1.)	"Nuclear Reactor Systems for Lunar Base Surface Power" Harvey S. Bloomfield	NASA-LeRC
2.)	"Advanced Photovoltaic Power System Technology for Lunar Base Applications" David J. Brinker and Dennis J. Flood	NASA-LeRC
3.)	"Power Requirements for Lunar Base Scenerios" Alan Friedlander and Kevin Cole	SAIC
4.)	"Inflatable Habitation for the Lunar Base" Michael L. Roberts	NASA-JSC
5.)	"Conceptual Design of a Lunar Oxygen Pilot Plant" Eric Christiansen and Charles H. Simonds	Eagle Engineering
6.)	"Excavation and Construction Operations for a Lunar Base" John Graf	NASA-JSC
7.)	"Lunar Base Launch and Landing Facilities Conceptual Design" Paul G. Phillips, Charles H. Simonds, and William R. Stump	Eagle Engineering

Solar Photovoltaic Versus Nuclear Power for a Lunar Observatory

J. M. Hickman, Advanced Space Analysis Office
H. S. Bloomfield, Power Technology Division
NASA Lewis Research Center

OBJECTIVE

To compare solar photovoltaic (PV) and nuclear power systems for the construction and operations phases of a farside lunar observatory outpost and document significant issues and findings.

BACKGROUND

The objective of this Office of Exploration (OEXP) case study is to emplace and operate a moderately sophisticated complement of scientific observational instrumentation on the farside of the moon. The baselined ground rules for this case study are that the setup of the observatory will be accomplished over a two-year period beginning in the year 2000, with one cargo and one crew mission per year. Crew stay-times for construction are baselined at 14 days per trip or less. Also, the lunar observatory will be operating unattended for long periods of time. Therefore, the power system selected must show high reliability and autonomy. A comparison of total power system masses between solar photovoltaic and nuclear power systems was made for the observatory construction and operations phases.

CASES STUDIED

Three different cases were considered encompassing 6 different scenarios. A summary of the cases studied is presented in Table 1. Case 1 is the "photovoltaic only" case. In this case gallium arsenide sun-tracking arrays are used to provide daytime operations power. Amorphous silicon (a-Si) arrays are used initially to provide construction power, and then later are used to provide power to regenerative fuel cells (RFCs) for operational night-time storage.

In case 2, a-Si photovoltaic (PV) arrays provide construction power, after which an SP-100 nuclear reactor provides power for operations. Case 2 is divided into four sub-cases. The first two (a and b) require construction of the lunar observatory and its power system through a single lunar night period with assumed construction power requirements of 20 kWe and 40 kWe, respectively. The third and fourth sub-case (c and d) require construction of the observatory and power system over multiple lunar nights, again with construction power at 20 kWe and 40 kWe, respectively. If only one lunar night is required (subcases a and b), then non-regenerative primary fuel cells (PFCs) may be employed to support the crew and limited activity during this time. If multiple nights are required, hydrogen/oxygen regenerative fuel cells (RFCs) are assumed to support the construction phase.

For Case 3, an alternative nuclear power system was identified and first order calculations performed. This system consisted of an SP-100 reactor thermoelectric (TE) power system encased within a fully shielded lunar lander. The shielding is rated for man-tended operation. After landing at the surface site, radiators are deployed from the lander, a power cable is installed, and the power system is fully operational within 24 hours of cable connection to the load. Depending on the lander power capability, an amorphous silicon PV array could be initially deployed on the lunar surface to provide power if required. The TE power system could be used for both construction and operations phases, or could be activated just before crew departure for the operations phase only.

ASSUMPTIONS

The construction phase power requirement is assumed at 20 - 40 kWe; the operations phase power requirement is assumed at 50 - 100 kWe. These requirements were obtained from the Office of Exploration (OEXP) Surface Systems Integration Agent.

Two separate solar power systems were assumed for the photovoltaic only case (Case 1). Amorphous silicon (a-Si) solar photovoltaic arrays were assumed to supply the construction power requirements. During this time sun-tracking gallium arsenide (GaAs) fold-out arrays and regenerative fuel cell storage stacks, gaseous reactant tanks, radiators, and power management and distribution equipment are deployed/erected. The GaAs tracking arrays were selected to supply the daytime operational power of the observatory. The a-Si arrays would be used to charge the regenerative fuel cells (RFCs) for night-time power after the construction is completed. Ten 20-kWe roll-out non-tracking a-Si arrays at 2218 W/kg were assumed to insure a 20-kWe power output (or better) for any angle of incident solar insolation less than 84.3 degrees, as measured from the normal to the array surface. Because the arrays are laid out flat on the surface, solar insolation at angles greater than 84.3 degrees would not receive enough solar energy to produce the 20 kWe construction power requirement. The GaAs array was assumed to have a mass specific power of 185 W/kg. The RFC storage density was rated at 1000 W-hr/kg, and the storage power management and distribution (PMAD) efficiency was assumed to be 92 percent. Due to relatively simple construction equipment needs, the construction PMAD mass estimate was assumed to be 10 kg/kWe, while the operations PMAD was assumed to be 20 kg/kWe because of more complex power conditioning needs of the observatory scientific equipment. The specific mass for the RFC radiator includes the storage structure and is assumed to be 5 kg/kWe. The complete operational power system should be erectable within one fourteen-day stay-time. (Refer to Table 2 for a summary of the assumptions for each case.)

Amorphous silicon PV arrays were selected for case 2 (Solar PV for construction/Nuclear System for Ops) because it was assumed that these arrays could be easily rolled out on the lunar surface in a few days. The PV system consists of ten roll-out a-Si non-tracking arrays as in the solar only case described above. The nuclear reactor heat source is located in a surface excavation[1], thereby utilizing lunar soil for radiation shielding. Stirling cycle power conversion is assumed. Construction times for this nuclear power system are assumed to be greater than 14 days and thus precluded the use of such a system as a power source for the Lunar Observatory construction phase.

As in Case 1, the Case 2 study assumes the construction PMAD and radiator/storage structure are rated at 10 kg/kWe and 5 kg/kWe, respectively. If only a single lunar night is required for construction then cryogenic primary fuel cells (PFCs) are used. The cryo PFCs have an energy density of 1500 W-hr/kg. Should construction of the nuclear power system for the observatory take longer than one lunar night, then gaseous hydrogen/oxygen (GH_2/GO_2) RFCs are assumed. The GH_2/GO_2 RFCs have an energy density of 1000 W-hr/kg.

The third and final case is for the Nuclear Lander concept. This concept locates an SP-100 reactor within a lunar lander. The reactor has a 4-pi shield which provides for a radiation dose level of 5 rem per 30 days at a distance of 1-km. The reactor could be activated just before the construction crew leaves the lunar surface or, alternatively, the reactor could be brought on-line soon after landing to provide construction power. Optional roll-out a-Si non-tracking arrays could be employed for the construction phase.

RESULTS

The operations phase power requirements can be met by either solar photovoltaic or nuclear power systems, depending on the power level. For operational power levels below 25 kWe, a solar photovoltaic power system was found to be attractive from both a construction time and system mass viewpoint (see figure). Mass totals for Case 1 are given in Table 3.

Depending on the nuclear power system selected, either the nuclear system or the PV system fairs better in the range of 25 to 60 kWe. (A mass summary for Case 2 is given in Table 4.)

For operational power levels in excess of about 60 kWe, the nuclear reactor power system exhibits a mass advantage over the solar PV power system, and that advantage increases significantly with higher power requirements. This concept

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[1] SP-100 Power System Conceptual Design for Lunar Base Applications, Mason, Lee S., Bloomfield, Harvey S., NASA Lewis Research Center, 1988.

includes longer construction times than the solar PV power system.

Although, the nuclear lander concept (case 3) has not been analysed extensively at this point, there is good reason to be optimistic about its mass and performance. The mass for the nuclear lander is slightly less than that for the surface reactor/single lunar night 20 kWe construction case (Case 2a) and is less massive than the PV system for all power levels above 25 kWe (see figure). The system mass for the lander power system is under 14 metric tonnes at a power level of 100 kWe, which is provided continuously for both operations and construction (See Table 5). This concept is now under further investigation.

Table 1

Case	Construction Power, kWe	Construction Power System	Operations Power System
1	20	a-Si roll-out PV arrays (no storage)	GaAs Sun-tracking PV arrays for Daytime power, a-Si roll-out PV arrays to charge RFCs
2-a	20	a-Si roll-out PV arrays for daytime power, PFCs for night-time power (single lunar night)	SP-100 nuclear power system
2-b	40	[same as above]	[same as above]
2-c	20	a-Si roll-out PV array for daytime power, RFCs for night-time power (multiple lunar nights)	[same as above]
2-d	40	[same as above]	[same as above]
3	20	SP-100 Nulcear Power System in lunar lander, or a-Si roll-out PV arrays (with no storage)	SP-100 nuclear power system in lunar lander

Table 2 Assumptions

case	a-Si specific power (W/kg)	GaAs specific power (W/kg)	constr. PMAD specific mass (kg/kWe)	ops PMAD specific mass (kg/kWe)	PFC specific mass (kg/kWe)	PFC energy density (W-hr/kg)	RFC specific mass (kg/kWe)	RFC energy density (W-hr/kg)	construction power level (kWe)
1	2218	185	10	20	-	-	5	1000	20
2a	2218	-	10	20	5	1500	-	-	20
2b	2218	-	10	20	5	1500	-	-	40
2c	2218	-	10	20	-	-	5	1000	20
2d	2218	-	10	20	-	-	5	1000	40
3	2218	-	10	-	-	-	-	-	20

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Mass Summary for Lunar Observatory Power System - Case 1 [LUNOBS.WK1]

Table 3 Solar Photovoltaic Arrays for Construction and Operation
Masses in kg

Ops. Power (kWe)	Constr Power (kWe)	ten roll-out non-track a-Si mass 2218 W/kg	fold-out tracking GaAs mass 185 W/kg	constr. power management 10 kg/kWe	regen fuel cell mass 1 kWh/kg	Ops. PMAD mass 20 kg/kWe	RFC radiator w/storage structure 5 kg/kWe	TOTAL Constr. and Ops (kg)
20	20	90	108	200	7703	400	100	8602
	40	-		-				-
40	20	90	216	200	15407	800	200	16913
	40	180		400				17203
60	20	90	324	200	23110	1200	300	25224
	40	180		400				25514
80	20	90	432	200	30813	1600	400	33536
	40	180		400				33826
100	20	90	541	200	38516	2000	500	41847
	40	180		400				42137
500	20							
	40							
800	20							
	40							

Table 4 Solar Power for Construction -- Nuclear Power for Surface Operations Case 2

Masses in kg

Construction w/PFC for 1 night @100% duty cycle

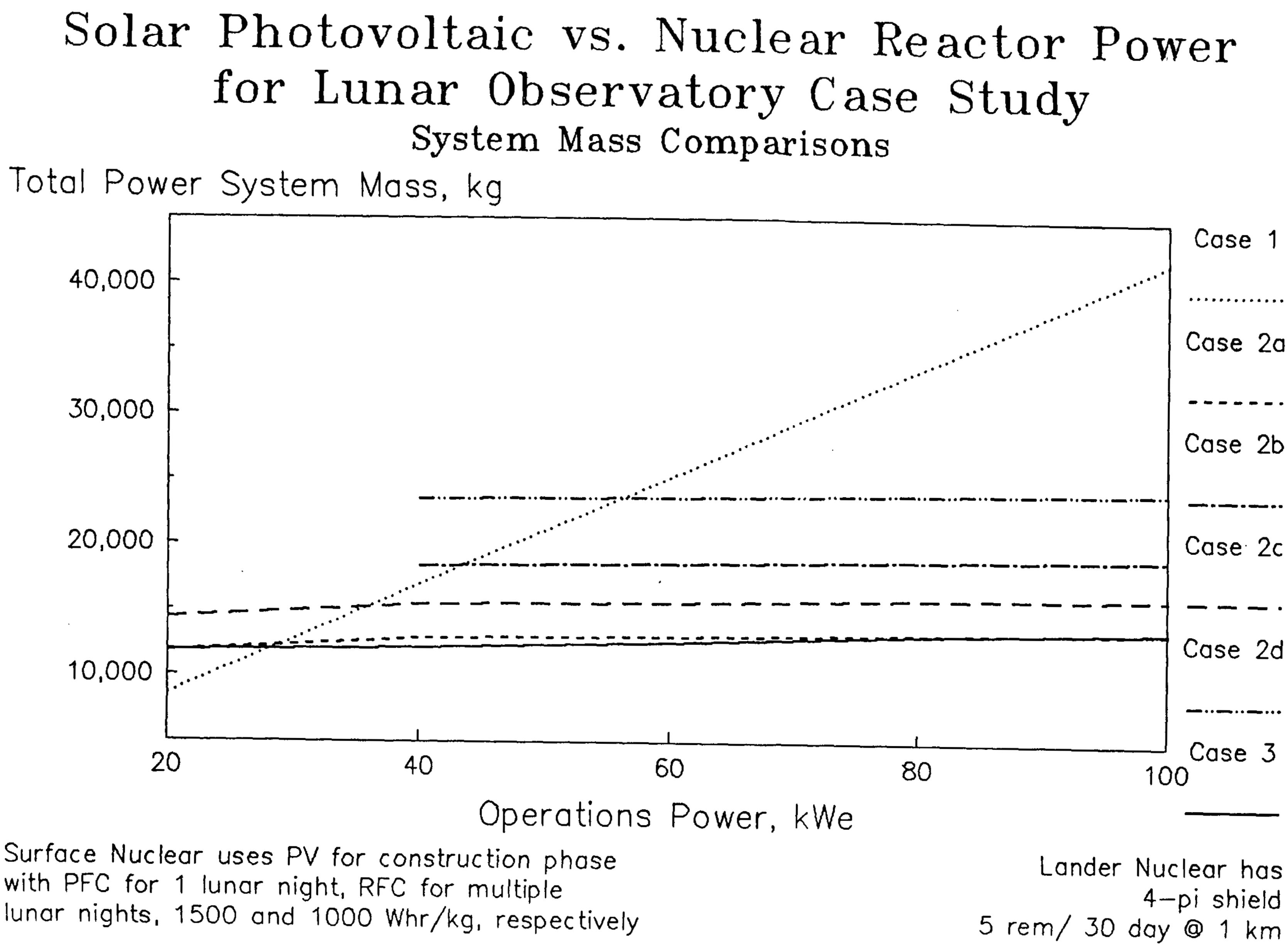
Ops. Power (kWe)	Constr Power (kWe)	ten roll-out non-track a-Si 2218 W/kg	Constr. power management 10 kg/kWe	PFC/RFC radiator w/storage structure 5 kg/kWe	Surface reactor power system mass for ops (kg)	Primary fuel cell w/cryo 1500 Wh/kg (kg)	TOTAL constr. and ops mass for 1-night constr.	constr. w/RFC for multiple nights 1 kWh/kg (GH2/GO2)	TOTAL constr. & ops mass - multinight constr. (kg)
20	20	90	200	100	6300	5136	11826	7703	14393
	40	-	-	-		-	-	-	-
40	20	90	200	100	7300	5136	12826	7703	15393
	40	180	400	200		10271	18351	15407	23487
60	20	90	200	100	7500	5136	13026	7703	15593
	40	180	400	200		10271	18551	15407	23687
80	20	90	200	100	7800	5136	13326	7703	15893
	40	180	400	200		10271	18851	15407	23987
100	20	90	200	100	8000	5136	13526	7703	16093
	40	180	400	200		10271	19051	15407	24187
500	20	90	200	100	15000	5136	20526	7703	23093
	40	180	400	200		10271	26051	15407	31187
800	20	90	200	100	20000	5136	25526	7703	28093
	40	180	400	200		10271	31051	15407	36187

Table 5

Nuclear Lander - Case 3

Masses in kg

Ops. Power (kWe)	Constr Power (kWe)	Optional ten roll-out non-track a-Si 2218 W/kg	Constr. power management 10 kg/kWe (kg)	Lander Reactor power system ops w/4 pi shield 5 rem/30 day at 1 km (kg)	TOTAL Lander Reactor mass w/optional solar PV
20	20	90	200	11600	11890
	40	-	-		-
40	20	90	200	11800	12090
	40	180	400		12380
60	20	90	200	12300	12590
	40	180	400		12880
80	20	90	200	12900	13190
	40	180	400		13480
100	20	90	200	13300	13590
	40	180	400		13880
500	20				
	40				
800	20				
	40				



Solar Photovoltaic vs. Nuclear Reactor Power
for Lunar Observatory Case Study
System Mass Comparisons
Total Power System Mass, kg
40,000
30,000
20,000
10,000
20
40
60
80
100
Operations Power, kWe
Case 1
Case 2a
Case 2b
Case 2c
Case 2d
Case 3
Surface Nuclear uses PV for construction phase
with PFC for 1 lunar night, RFC for multiple
lunar nights, 1500 and 1000 Whr/kg, respectively
Lander Nuclear has
4–pi shield
5 rem/ 30 day @ 1 km

Fig. 1

Power Technology Workshop

A mini, first-of-a-series, Power Technology Workshop was held at the Lewis Research Center on April 15, 1988. The workshop served as an open forum for Code Z's MASE and IA's and Lewis' power technologists to freely exchange information, concerns, ideas, and issues. Representatives from NASA Headquarters' Code Z and Code R also participated in the workshop.

The first topic of discussion was the origin and rationale of the power requirements chart that appears in the Code Z PRD. The ensuing dialog produced suggested amendments and augmentations to the chart that were carried out on-line during the workshop.

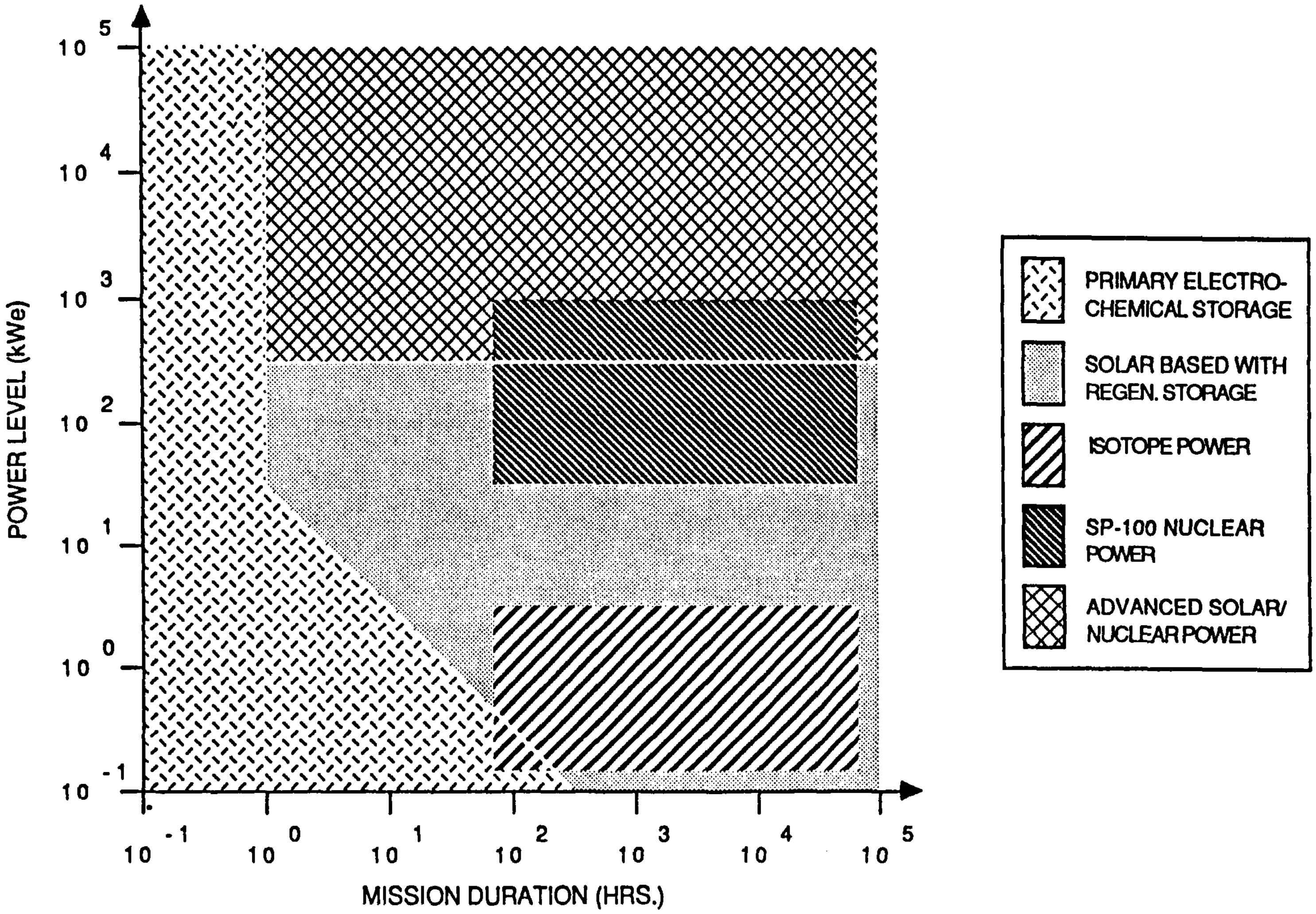
Each Integration Agent presented their perspectives on projected requirements and issues. Projected power needs for nodes, spacecraft, and planetary surfaces were discussed. Concerns were raised about some of the preconceived attributes of the various power systems selected to satisfy the power requirements. Several action items were, therefore, levied on Lewis to amend the power domain graph and to develop a preliminary figure of merit list for selected power systems. These documents are included in this section.

An additional IA perception led to a quick-cycle study. The study analyzes the mass associated with adding advanced storage for the lunar night during the construction phase of the lunar observatory.

Overviews of the technological status of space power systems were presented by Lewis technologists encompassing the topics of nuclear power, isotopic dynamic systems, photovoltaics, solar dynamics, batteries, and fuel cells.

Future power workshops will be held as deemed appropriate.

APPROXIMATE POWER SYSTEM DOMAINS



CAUTION: THIS CHART OVERSIMPLIFIES THE POWER SYSTEM ARENA. IT SHOULD BE USED AS ONLY A GUIDE AND NOT A DESIGN TOOL. THE ACTUAL POWER SYSTEM DESIGN IS HEAVILY DEPENDENT ON MISSION SPECIFIC CHARACTERISTICS.

EXPLORATION OF POWER SYSTEM DOMAINS

PRIMARY ELECTROCHEMICAL STORAGE:

- o Above about 1 hour and 50 kWe the energy storage requirements favor regenerative systems. With regenerative systems all the energy need not be carried into space.
- o At levels below 50 kWe, the energy requirements favor primary electrochemical systems as a low weight, less complex system of choice.

SOLAR BASED WITH REGENERATIVE STORAGE SYSTEMS (Power Generator Coupled to Energy Storage Subsystem):

- o Beyond 500 kWe, civilian applications of solar based regenerative power systems have not been currently identified.
- o Below 1 hour missions, primary electrochemical systems are lighter due to low energy requirements.
- o Above 10^5 hour missions, space environmental effects degrade performance of exposed components.

RADIOISOTOPES:

- o Beyond 5 kWe the availability of isotopes becomes an issue.
- o Below 100 Watts, primary electrochemical systems are lighter and less expensive.
- o Below 100 hours usually means near sun missions, (high insolation), where solar based regenerative systems are lighter.
- o Above 10^5 hours, the half-life of isotopes become an issue.

SP-100 (Single Unit):

- o Beyond 10^5 hours, fuel depletion for single SP-100 system limits applications.
- o Below tens of hours advanced solar based regenerative systems may become lighter.
- o 1000 kWe is the upper limit of power available from a SP-100 reactor. However, the SP-100 technology is not limited to this upper bound level.
- o Below 50 kWe, reactor criticality factors result in higher specific weights. Solar based regenerative systems are lighter.

ADVANCED SOLAR OR NUCLEAR POWER SYSTEMS:

- Beyond the 0.5 to 1 MWe power level, both solar and nuclear based power systems have application. The choice depends on mission constraints such as weight, volume, area, complexity, reliability, duration, duty cycle, etc.
- At mission durations less than 1 hour, the choice of low weight, low cost power systems tend to favor primary electrochemical systems.

POWER SYSTEM FIGURES OF MERIT

6/21/88
[ASA0Z4]

CAUTION: - THESE DATA SHOULD BE USED ONLY AS A GROSS GUIDE AND NOT A DESIGN TOOL
- THE POWER SYSTEM DESIGN IS HEAVILY DEPENDENT ON MISSION SPECIFIC CHARACTERISTICS SUCH AS DURATION, ENVIRONMENT, DUTY CYCLE, RELIABILITY, COST, SAFETY, ETC.

TECHNOLOGY	MISSION	FIGURE OF MERIT (FOM) UNITS	STATE-OF-THE-ART FOM	POTENTIAL FOM	COMMENTS
PRIMARY ELECTROCHEMICAL STORAGE (SINGLE DISCHARGE)					
Lithium Thermal Battery	Munitions	W/KG	2000	4500	
		W-HR/KG	12	120	
Lithium Reserve Battery	Various	W/KG	1000	4500	
		W-HR/KG	200-300	300-400	
Alkaline Primary Fuel Cell	Various	W/KG	100	8000	Power Gen. Portion
		W-HR/KG	750	1200	Energy Stor. Portion
SOLAR BASED SYSTEMS					
Solar Dynamic/TES	LEO	W/KG	4.5	12	
		W/M^2	190	400	
PV-Si/Ni-H2 Battery	LEO	W/KG	4.5	7	
		W/M^2	99	123	
PV-Si/Ni-H2 Battery	GEO	W/KG	12	15	
PV-GaAs/NaS or Lithium Battery	LEO	W/KG	NA	16	
		W/M^2	NA	272	
PV-GaAs/NaS or Lithium Battery	GEO	W/KG	NA	28	
PV/RFC	Lunar Surface Base	W/KG	0.15	3	
PV/RFC	Mars Surface Base	W/KG	1.5	8	
ISOTOPE POWER SYSTEMS					
GPHS/TE	LEO to Deep Space	W/KG	5	6-7	
		EFF. %	5-10	10-13	
GPHS/Brayton	LEO to Deep Space	W/KG	6-7	6-7	
		EFF. %	26	26	
GPHS/Stirling	LEO to Deep Space	W/KG	NA	10	
		EFF. %	NA	30	
NUCLEAR POWER SYSTEMS					
SP-100 with TE's (Baseline)	Space & Surface	KG/KW	NA	30	100-300 kWe
SP-100 with Adv. Dynamic Tech.	Space & Surface	KG/KW	NA	12	1000 kWe
MMWe Reactor with Rankine or TI's	NEP	KG/KW	NA	7	

The Reader Is Referenced To NASA Conference Publication 10018, Lunar Helium-3 and Fusion Power. This previously published document describes the proceedings from the NASA Lunar Helium-3/Fusion Power workshop held April 25-26, 1988 at the NASA Lewis Research Center. Within this volume are reports by two working groups on varios aspects of mining Helium-3 from the Moon for use in terrestrial fusion power plants. Also included in this report are summaries of group discussions, minority opinions, and papers submitted after the workshop that have bearing on the workshop topic.

EVALUATION OF ADVANCED PROPULSION/ POWER CONCEPTS

PRESENTED
TO

ADVANCED SPACE PROPULSION
WORKSHOP

BY
ADVANCED SPACE ANALYSIS OFFICE
SVERDRUP/NASA-LERC

APRIL 12-13, 1988

EVALUATION OF ADVANCED PROPULSION CONCEPTS

BACKGROUND:

- A REQUEST HAS BEEN MADE BY CODE Z TO ASAO/LeRC TO PROVIDE AN EVALUATION OF ADVANCED PROPULSION CONCEPTS AND THEIR SUITABILITY FOR FUTURE NASA INITIATIVES

- SPECIFICALLY - WHAT PROPULSION CONCEPTS ARE AVAILABLE?
  - WHAT CONCEPTS MAKE SENSE FOR FUTURE MISSIONS?

- REQUESTED INFORMATION TO INCLUDE AN ASSESSMENT OF THE CAPABILITIES OF VARIOUS PROPULSION/POWER COMBINATIONS AND THEIR ASSOCIATED PROS AND CONS

- THIS TASK EXAMINED OVER 700 POWER/PROPULSION COMBINATIONS OBTAINED FROM OVER 50 REFERENCES DURING A FOUR WEEK PERIOD

- THIS WORK IS IN RESPONSE TO THE DECEMBER 1987 REQUEST FOR RESULTS TO BE PRESENTED TO THE SSTAC WORKING GROUP FOR LUNAR AND PLANETARY MISSION PROPULSION ON MARCH 10-11, 1988

NASA
Lewis Research Center
Space Flight Systems Directorate

PRESENTATION OUTLINE

ASAO

- OBJECTIVE OF STUDY
- APPROACH (SUCCESSIVE FILTERING/SCREENING PROCESS)
  - DEFINE FILTERS (WHAT/WHY)
  - ILLUSTRATE FILTER IMPLEMENTATION
- IDENTIFICATION OF "CREDIBLE" PROPULSION/POWER CONCEPTS (SUMMARY MATRIX)
- PROS AND CONS OF CREDIBLE SYSTEMS
- SUMMARY
- SUPPORTING INFORMATION (APPENDIX)

STUDY PARTICIPANTS

S. K. BOROWSKI J. M. GILLAND	SVERDRUP TECHNOLOGY, INC.
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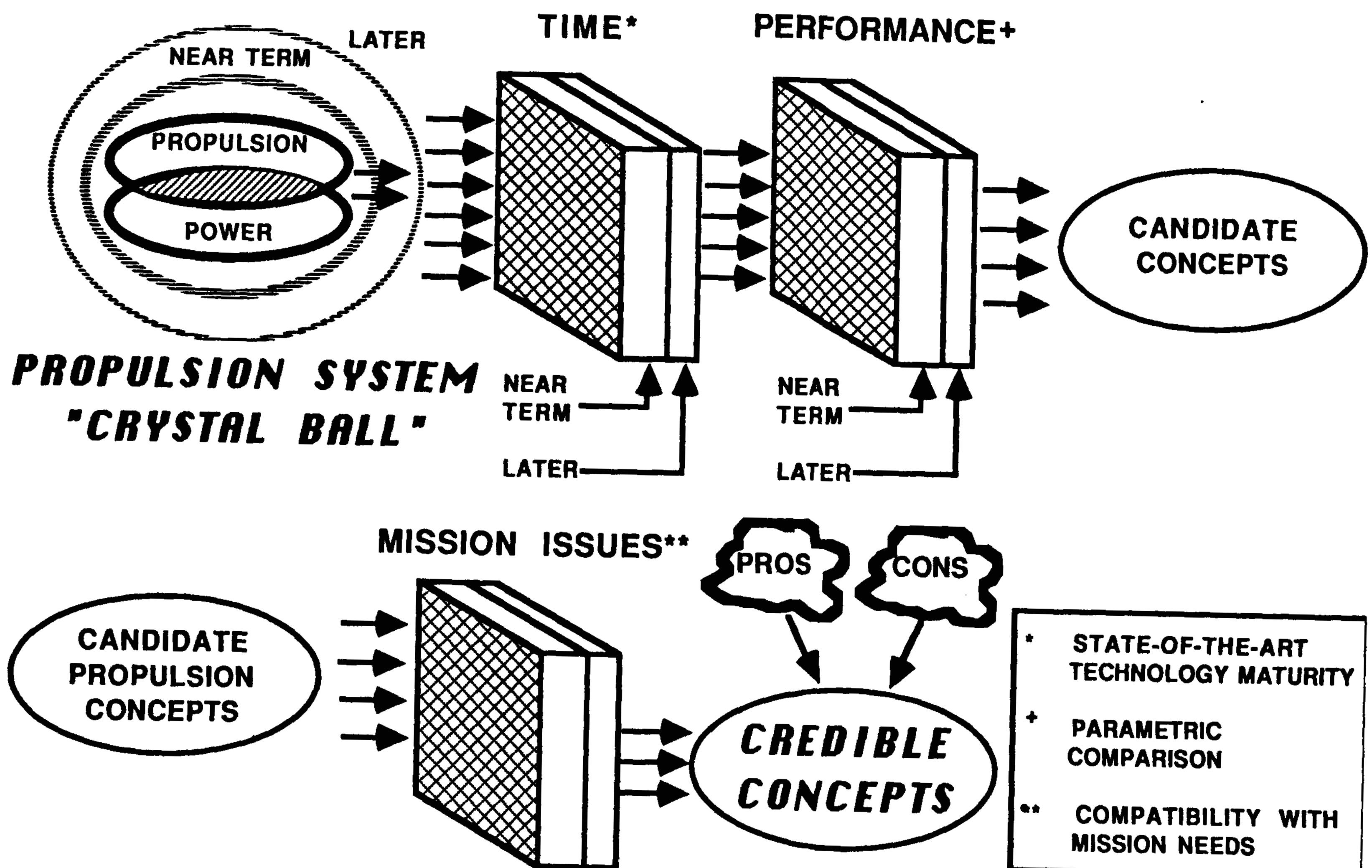
D. J. FLOOD J. M. BOZEK	NASA/ POWER TECHNOLOGY DIVISION

EVALUATION OF ADVANCED PROPULSION CONCEPTS

- OBJECTIVES:
  (1) ASSEMBLE THE NECESSARY INFORMATION TO ASSESS THE CAPABILITIES AND POTENTIAL OF VARIOUS PROPULSION AND POWER TECHNOLOGIES

  (2) CONSTRUCT A SUMMARY MATRIX OF "CREDIBLE" PROPULSION/POWER CONCEPTS FOR POSSIBLE CONSIDERATION IN FUTURE NASA MISSIONS

FILTERING APPROACH USED IN EVALUATION



* STATE-OF-THE-ART TECHNOLOGY MATURITY

\+ PARAMETRIC COMPARISON

** COMPATIBILITY WITH MISSION NEEDS

WHY THE NEED FOR A FILTERING/ SCREENING PROCESS

- A LOT OF PROPULSION CONCEPTS EXIST IN WHICH PRIMARY POWER IS USED EITHER:
  (1) DIRECTLY (CHEMICAL PROPULSION, SOLAR SAILS)
  (2) INDIRECTLY (CONVERSION TO ELECTRICITY FOR EP)
- A LOT OF POWER SOURCE/CONVERSION CONCEPTS EXIST

POWER SOURCES	POWER CONVERSION TECHNIQUES
SOLAR: PHOTON	PHOTOVOLTAIC (PV)
THERMAL	THERMOELECTRIC (TE)
NUCLEAR: RADIOISOTOPE	THERMIONIC (TI)
FISSION REACTOR:	BRAYTON
- HEAT-PIPE COOLED	RANKINE
- FLUID COOLED	STIRLING
- LIQUID	MHD:
- GAS:	- SEEDED GAS
- SOLID CORE	- LIQUID METAL
- PARTICLE BED	ELECTROSTATIC (ES)
- GAS CORE	MAGNETIC INDUCTION
FUSION REACTOR:	
- MAGNETIC	
- INERTIAL	

- DISCRIMINATOR FOR "CREDIBLE" VS. "NOT-SO-CREDIBLE" CONCEPTS REQUIRED

FILTER DEFINITION

- FILTERS USED:

  (1) TIME - OPERATIONAL DATE FOR SYSTEM AVAILABILITY, TECHNOLOGY MATURITY/READINESS

  LEVELS: (1) SIGNIFICANT GROUND/FLT. TEST/DEMO
  (2) LABORATORY DEVICE
  (3) DESIGN CONCEPT/IDEA

  NEAR TERM: (1), (2)
  LATER: (2), (3)

- STATE-OF-THE-ART DATA (AVAILABLE/PROJECTED) FOR CANDIDATE PROPULSION SYSTEMS AND KEY TECHNOLOGIES (EP AND POWER) HAS BEEN ASSEMBLED AND USED IN OUR EVALUATION

FILTER DEFINITION
(Continued)

(2) PERFORMANCE COMPARISON - PROPULSION/POWER SYSTEM PERFORMANCE PARAMETERS. THESE ARE CALCULATED FROM ASSEMBLED DATA.

(3) MISSION/PROPULSION SYSTEM COMPATIBILITY

- COMPARE "GENERIC" MISSION REQUIREMENTS AND PROPULSION SYSTEM CHARACTERISTICS
- SHORT TRIP TIMES AND HIGH PAYLOAD FRACTION ARE TWO FAVORABLE REGIONS OF PERFORMANCE
- THESE REGIONS ARE ALSO DEFINED USING DATA FROM PREVIOUS MISSION STUDIES USING HIGH PERFORMANCE SYSTEMS

POWER/PROPULSION COMPONENTS

POWER

SOLAR	
	PHOTOVOLTAIC (PV)
	DYNAMIC (SD)
NUCLEAR	
ISOTOPE	THERMOELECTRIC (TE)
	DYNAMIC
FISSION	THERMOELECTRIC (TE)
	THERMIONIC (TI)
	DYNAMIC
FUSION	DYNAMIC
	ELECTROSTATIC (ES)
	INDUCTION
MASS ANNIHILATION	DYNAMIC

PROPULSION

ELECTRIC PROPULSION (EP)
- RESISTOJET, ARCJET, MICROWAVE THRUSTER
- ION
- PULSED ELECTROTHERMAL THRUSTER (PET)
- MAGNETOPLASMADYNAMIC THRUSTER (MPD)
- PULSED INDUCTIVE THRUSTER (PIT)
- RAIL GUN
- MASS DRIVER

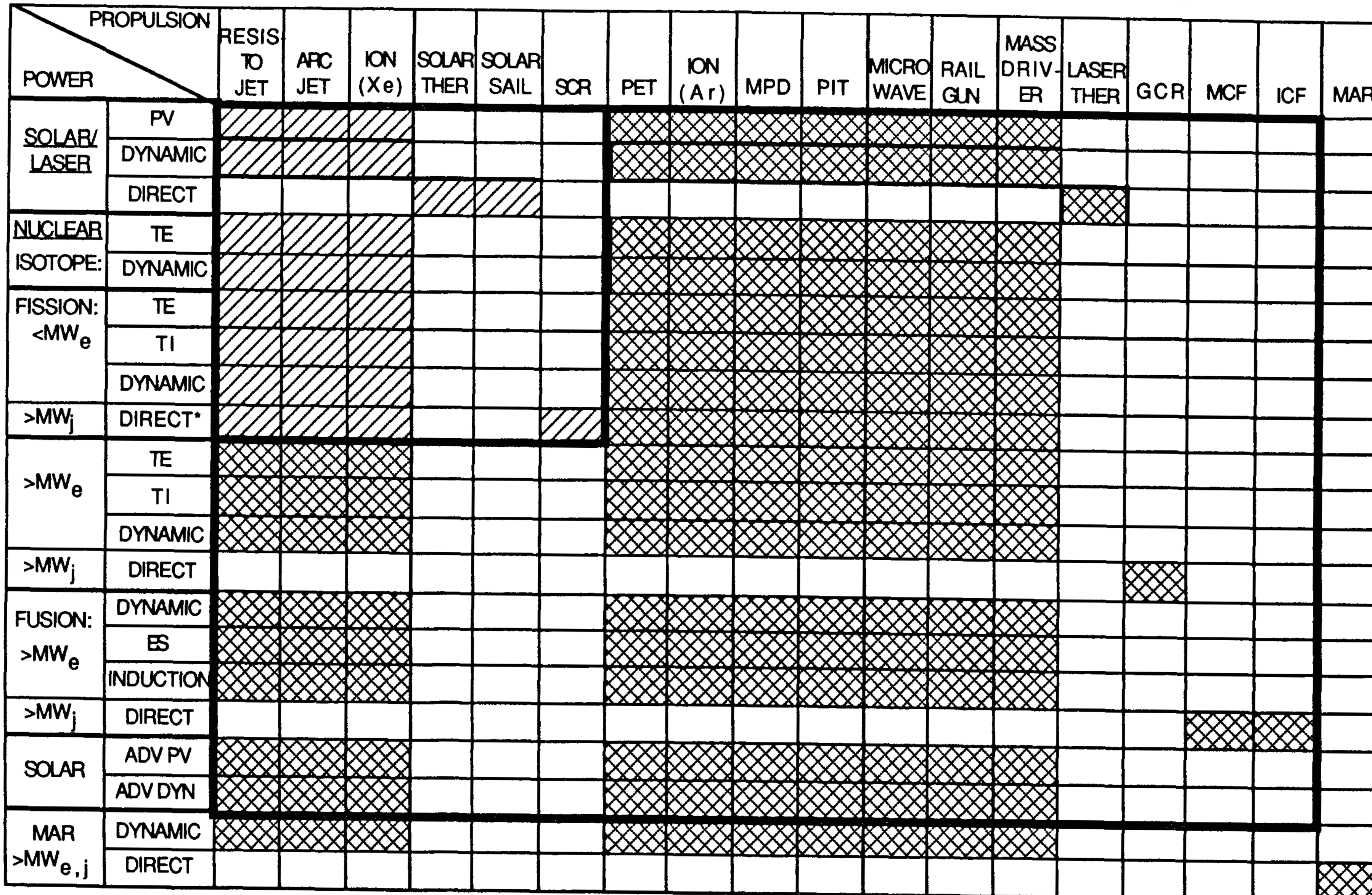
SOLAR SAIL
SOLAR THERMAL ROCKET (STR)
LASER THERMAL ROCKET (LTR)

NUCLEAR FISSION
- SOLID CORE ROCKET (SCR)
- GAS CORE ROCKET (GCR)

NUCLEAR FUSION
- MAGNETIC CONFINEMENT FUSION (MCF)
  - -TOKAMAK FUSION ROCKET (TFR)
- INERTIAL CONFINEMENT FUSION (ICF)
  - -INERTIAL FUSION ROCKET (IFR)
  - -LIVERMORE IFR CONCEPT (VISTA)

MASS ANNIHILATION ROCKET (MAR)

POSSIBLE PROPULSION/POWER CONCEPTS FILTERED ACCORDING TO TIME



*NOTE: DIRECT NUCLEAR ROCKETS CAN BE INTEGRATED WITH POWER CONVERSION AND EP TO OBTAIN A "HYBRID" SYSTEM WITH THE ADVANTAGES OF BOTH SYSTEMS.

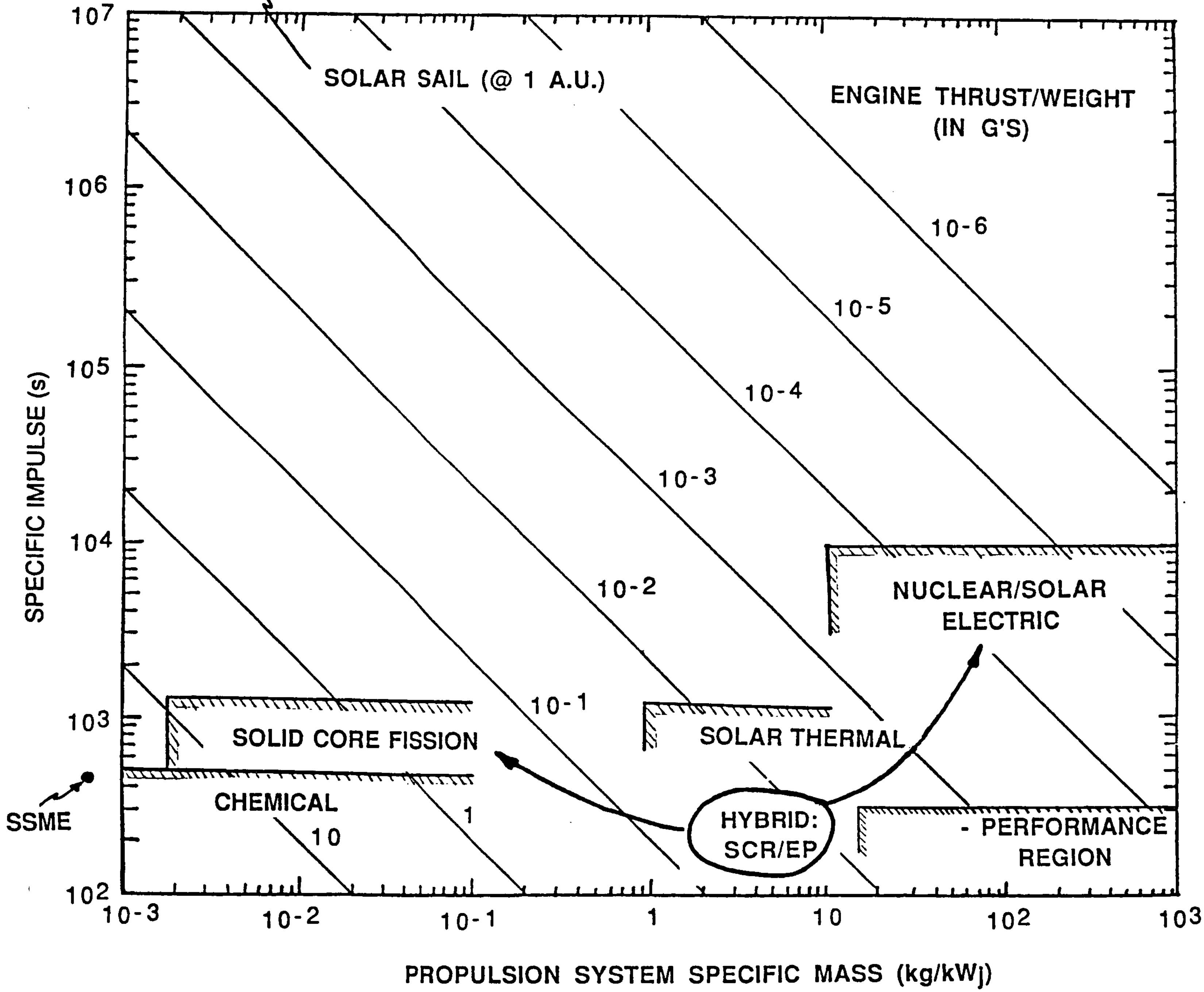
NEAR TERM (10 - 15 YEARS)

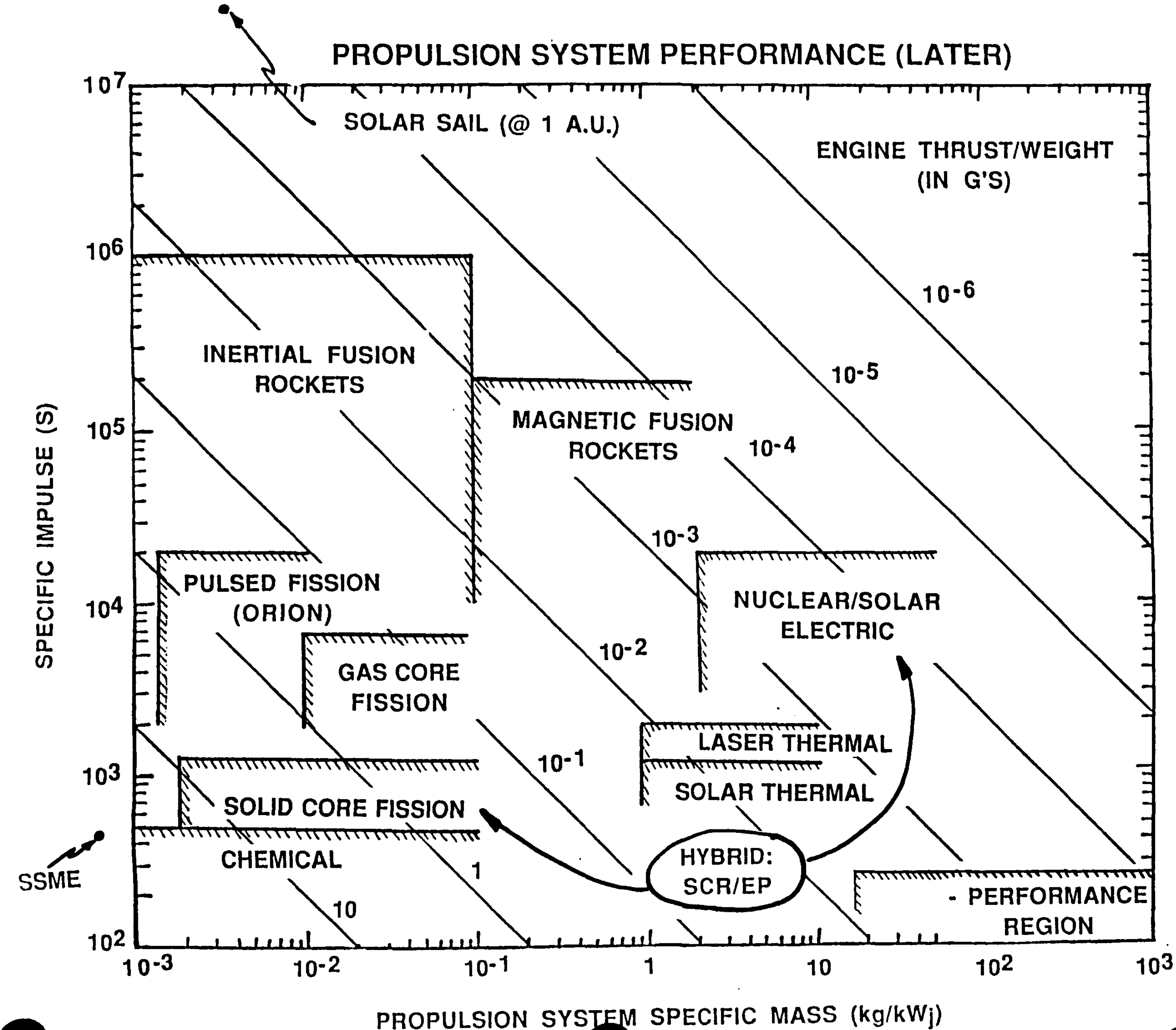
LATER TERM (> 15 YEARS)

KEY PERFORMANCE PARAMETERS

PARAMETER	GOODNESS	IMPORTANCE
SPECIFIC IMPULSE (SECONDS)	HIGH	FUEL EFFICIENCY
SPECIFIC MASS (RECIPROCAL OF SPECIFIC POWER, kg/kW_j)	$(HIGH)^{-1}$	ENGINE POWER PRODUCING CAPABILITY
ENGINE THRUST/WEIGHT	HIGH	PROPULSION SYSTEM ACCELERATION CAPABILITY

PROPULSION SYSTEM PERFORMANCE (NEAR TERM)





PROPULSION SYSTEM PERFORMANCE (LATER)
SOLAR SAIL (@ 1 A.U.)
ENGINE THRUST/WEIGHT
(IN G'S)
10^{-6}
10^{-5}
10^{-4}
10^{-3}
10^{-2}
10^{-1}
1
10
INERTIAL FUSION
ROCKETS
MAGNETIC FUSION
ROCKETS
PULSED FISSION
(ORION)
NUCLEAR/SOLAR
ELECTRIC
GAS CORE
FISSION
LASER THERMAL
SOLAR THERMAL
SOLID CORE FISSION
CHEMICAL
HYBRID:
SCR/EP
- PERFORMANCE
REGION
SSME
SPECIFIC IMPULSE (S)
10^{7}
10^{6}
10^{5}
10^{4}
10^{3}
10^{2}
10^{-3}
10^{-2}
10^{-1}
1
10
10^{2}
10^{3}
PROPULSION SYSTEM SPECIFIC MASS (kg/kWj)

BASIS FOR PERFORMANCE COMPARISON

COMPARISON WITH:

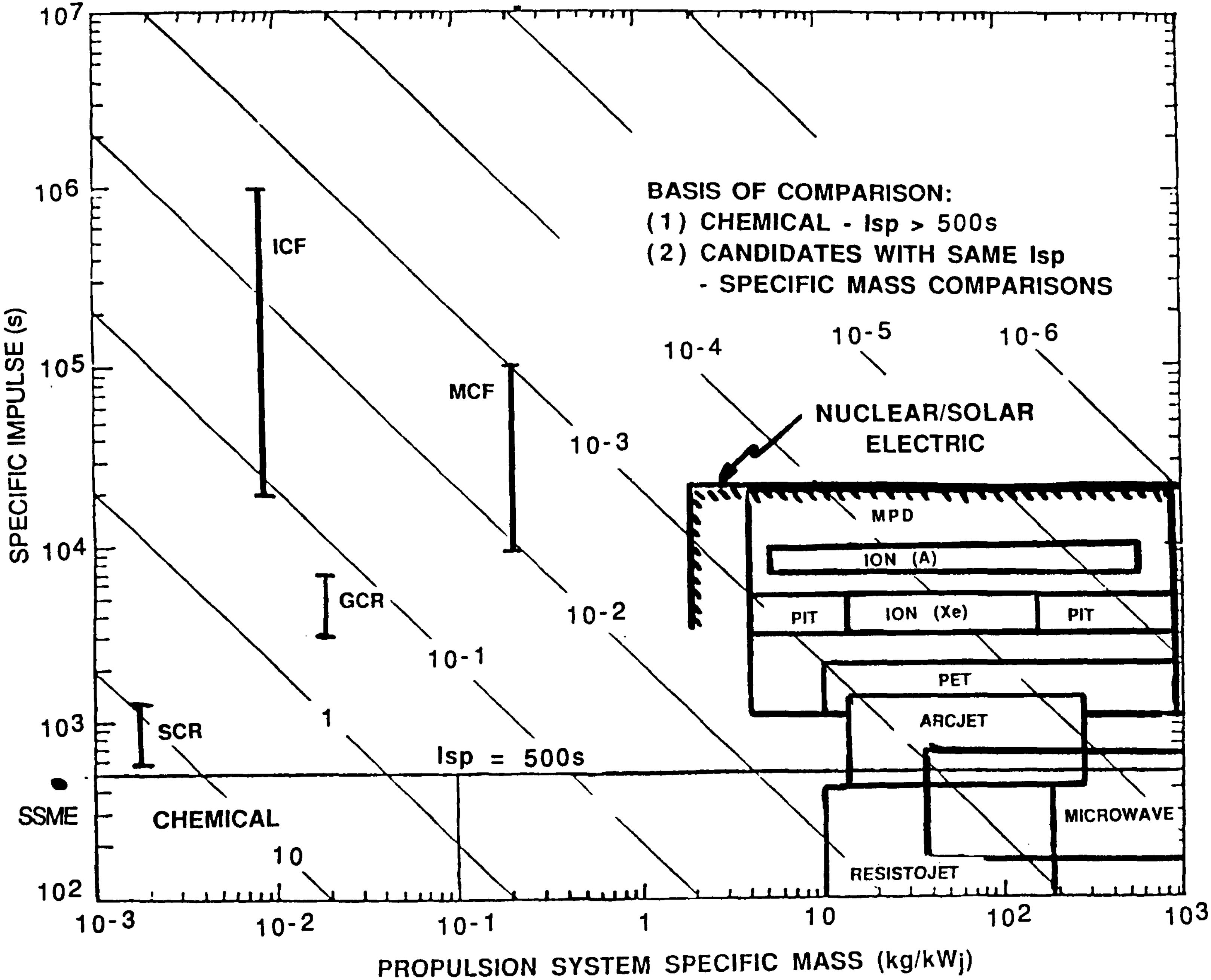
- CHEMICAL PROPULSION

  SYSTEMS WITH SPECIFIC IMPULSE LESS THAN OR EQUAL TO CHEMICAL STATE OF THE ART CAN BE ELIMINATED FOR PRIMARY PROPULSION APPLICATIONS.

- SIMILAR PROPULSION SYSTEMS

  FOR PROPULSION SYSTEMS WITH COMPARABLE SPECIFIC IMPULSE CAPABILITY, THOSE CONCEPTS WITH SIGNIFICANTLY HIGHER SPECIFIC MASS CAN ALSO BE ELIMINATED.

PROPULSION SYSTEM PERFORMANCE (LATER)



RESULTS OF PERFORMANCE FILTER

POWER	PROPULSION	RESISTO JET	ARC JET	ION (Xe)	SOLAR THER	SOLAR SAIL	SCR	PET	ION (Ar)	MPD	PIT	MICRO WAVE	RAIL GUN	MASS DRIVER	LASER THER	GCR	MCF	ICF	MAR
SOLAR/ LASER	PV		///	///				XXX	XXX	XXX	XXX		XXX	XXX					
	DYNAMIC																		
	DIRECT				///	///									XXX				
NUCLEAR ISOTOPE:	TE																		
	DYNAMIC																		
FISSION: $<MW_e$	TE		///	///				XXX	XXX	XXX	XXX								
	TI		///	///				XXX	XXX	XXX	XXX								
	DYNAMIC		///	///				XXX	XXX	XXX	XXX								
$>MW_j$	DIRECT*			///			///		XXX	XXX	XXX								
$>MW_e$	TE		XXX	XXX				XXX	XXX	XXX	XXX			XXX					
	TI		XXX	XXX				XXX	XXX	XXX	XXX			XXX					
	DYNAMIC		XXX	XXX				XXX	XXX	XXX	XXX			XXX					
$>MW_j$	DIRECT															XXX			
FUSION: $>MW_e$	DYNAMIC		XXX	XXX				XXX	XXX	XXX	XXX		XXX	XXX					
	ES		XXX	XXX				XXX	XXX	XXX	XXX		XXX	XXX					
	INDUCTION		XXX	XXX				XXX	XXX	XXX	XXX		XXX	XXX					
$>MW_j$	DIRECT																XXX	XXX	
SOLAR	ADV PV		XXX	XXX				XXX	XXX	XXX	XXX		XXX	XXX					
	ADV DYN		XXX	XXX				XXX	XXX	XXX	XXX			XXX					
MAR $>MW_{e,j}$	DYNAMIC																		
	DIRECT																		

*NOTE: DIRECT NUCLEAR ROCKETS CAN BE INTEGRATED WITH POWER CONVERSION AND EP TO OBTAIN A "HYBRID" SYSTEM WITH THE ADVANTAGES OF BOTH SYSTEMS.

/// NEAR TERM (10 - 15 YEARS)

XXX LATER TERM (> 15 YEARS)

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MISSION/PROPULSION SYSTEM COMPATIBILITY CONSIDERATIONS

ASAO

	MISSION CATEGORY	PROPULSION SYSTEM FEATURES	EXAMPLE SYSTEMS	MISSION IMPACT
NEAR	SPRINT	LOW I_{SP}/HIGH ENGINE THRUST/WEIGHT	CHEMICAL WITH AEROBRAKE	HIGH PROPELLANT FRACTION /MODERATE-SHORT TRIPTIME
NEAR	CARGO	HIGH I_{SP}/LOW ENGINE THRUST/WEIGHT	NUCLEAR ELECTRIC PROPULSION	HIGH PAYLOAD FRACTION /LONG TRIPTIME
LATER	SPRINT + CARGO	HIGH I_{SP}/HIGH ENGINE THRUST/WEIGHT	GAS CORE ROCKET /FUSION ROCKET	HIGH PAYLOAD FRACTION /SHORT TRIPTIME

MISSION ANALYSIS CONSIDERATIONS

- DETAILED MISSION ANALYSIS REQUIRES ADDITIONAL DATA (SUCH AS DESTINATION AND PAYLOAD) TO PROVIDE AN ACCURATE ASSESSMENT OF TRIP TIME AND PAYLOAD FRACTION.

- A COMPARISON OF SYSTEM MISSION PERFORMANCE REQUIRES CONSISTENT GROUND RULES AND CONDITIONS.

- SYSTEM MISSION PERFORMANCE IS LIMITED BY SPECIFIC IMPULSE AND SPECIFIC POWER CAPABILITIES. WITHIN THESE LIMITS, A RANGE OF PERFORMANCE (TRIP TIME, PAYLOAD FRACTION) CAN BE ATTAINED.

- PRIMARY PROPULSION CAN BE APPLIED TO A RANGE OF MISSIONS, FROM ORBITAL TRANSFER TO INTERPLANETARY EXPLORATION. A CHANGE IN MISSION DEMANDS MAY ALTER THE BOUNDARIES BETWEEN CARGO AND COURIER REGIONS.

- PROPULSION SYSTEM POWER LEVELS ARE NOT FIXED BY SPECIFIC IMPULSE AND SPECIFIC POWER. SYSTEM POWER REQUIREMENTS HAVE NOT BEEN FILTERED INTO THIS STUDY.

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REFERENCE ENGINE DESIGNS

ASAO

	SYSTEM	DESTINATION	TRIP TIME+	PAYLOAD MASS* / INITIAL MASS	APPENDIX REFERENCES
1)	NEP/ION (SP-100)	NEPTUNE	12 YEARS (PROBE-1 WAY)	0.10	8
2)	LTR (1 MW)	LEO-GEO	28 DAYS	0.44	4
3)	STR (2 MW)	LEO-GEO	30 DAYS	0.54	6
4)	SEP/ION (300 KW_e)	MOON	370 DAYS	0.57	7
5)	PEGASUS/MPD (8.5 MW_e)	MARS	1000 DAYS	0.45	13
6)	NEP/ION (300 kW_e)	NSO-MARS	770 DAYS (CARGO-1 WAY)	0.44	25
7)	NEP/ION (3 MW_e)	MOON-MARS	413 DAYS (CARGO- 1 WAY)	0.54	25

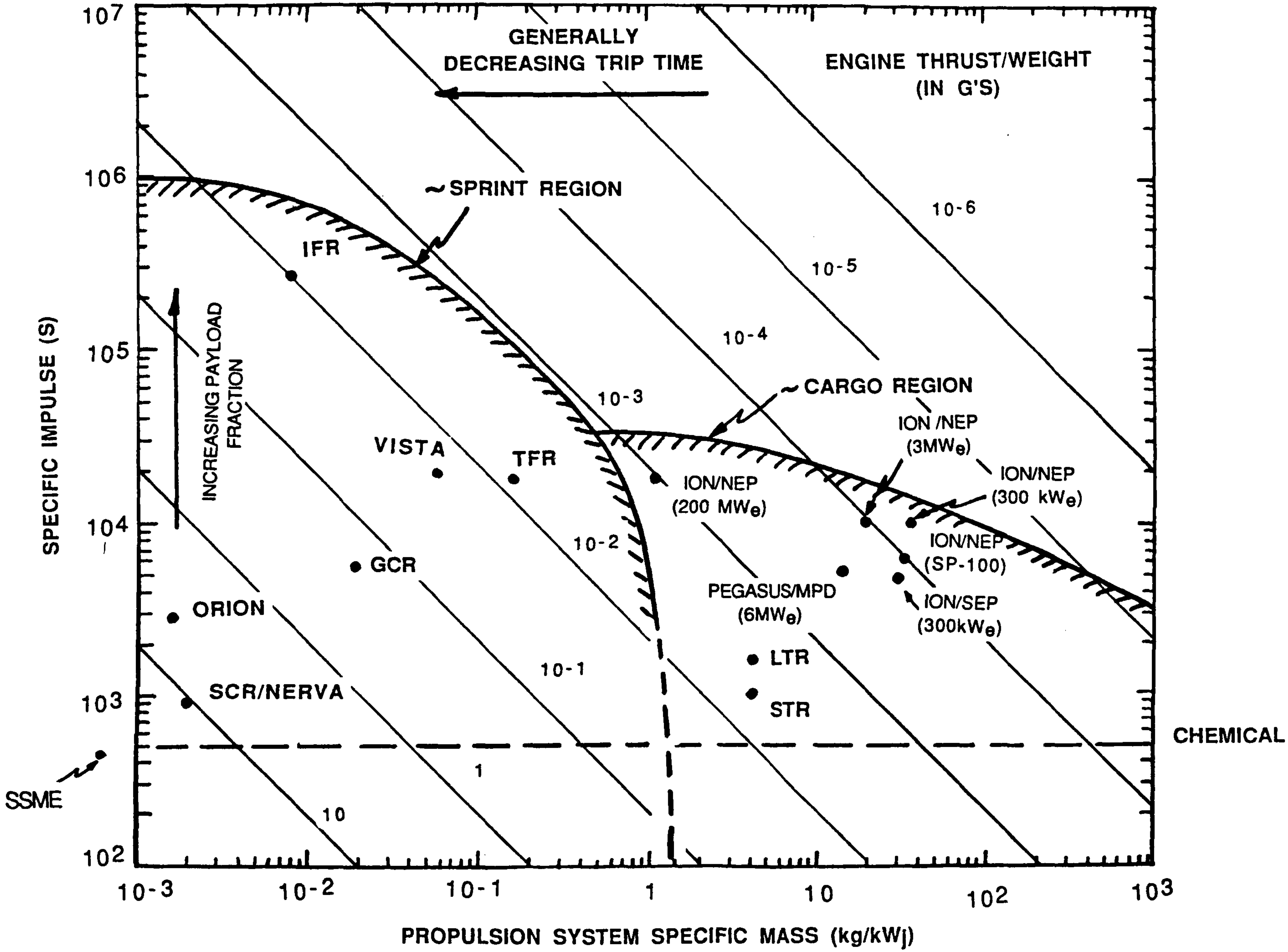
\+ ROUND TRIP UNLESS OTHERWISE INDICATED
* TO DESTINATION ONLY

REFERENCE ENGINE DESIGNS (CONTINUED)

	SYSTEM	DESTINATION	TRIP TIME+	PAYLOAD MASS* / INITIAL MASS	APPENDIX REFERENCES
8)	SCR/NERVA (5000 MW)	MARS	720 DAYS	0.23	40
9)	NEP/ION (200 MW_e)	MARS	~ 180 DAYS	0.0	39
10)	GCR (8500 MW)	MARS	80 DAYS / 280 DAYS	0.075 / 0.25	15
11)	ORION (43,000 MW)	MARS	250 DAYS	0.23	16
12)	TFR (7500 MW)	MARS	77 DAYS	0.06	17
13)	VISTA (225,000 MW)	MARS	100 DAYS	0.017	20
14)	IFR (200,000 MW)	MARS / PLUTO	55 DAYS / 20 MONTHS	0.26 / 0.14	17,18

\+ ROUND TRIP UNLESS OTHERWISE INDICATED
* TO DESTINATION ONLY

MISSION REQUIREMENTS REGIONS



ATTRACTIVE SPRINT SYSTEM CANDIDATES

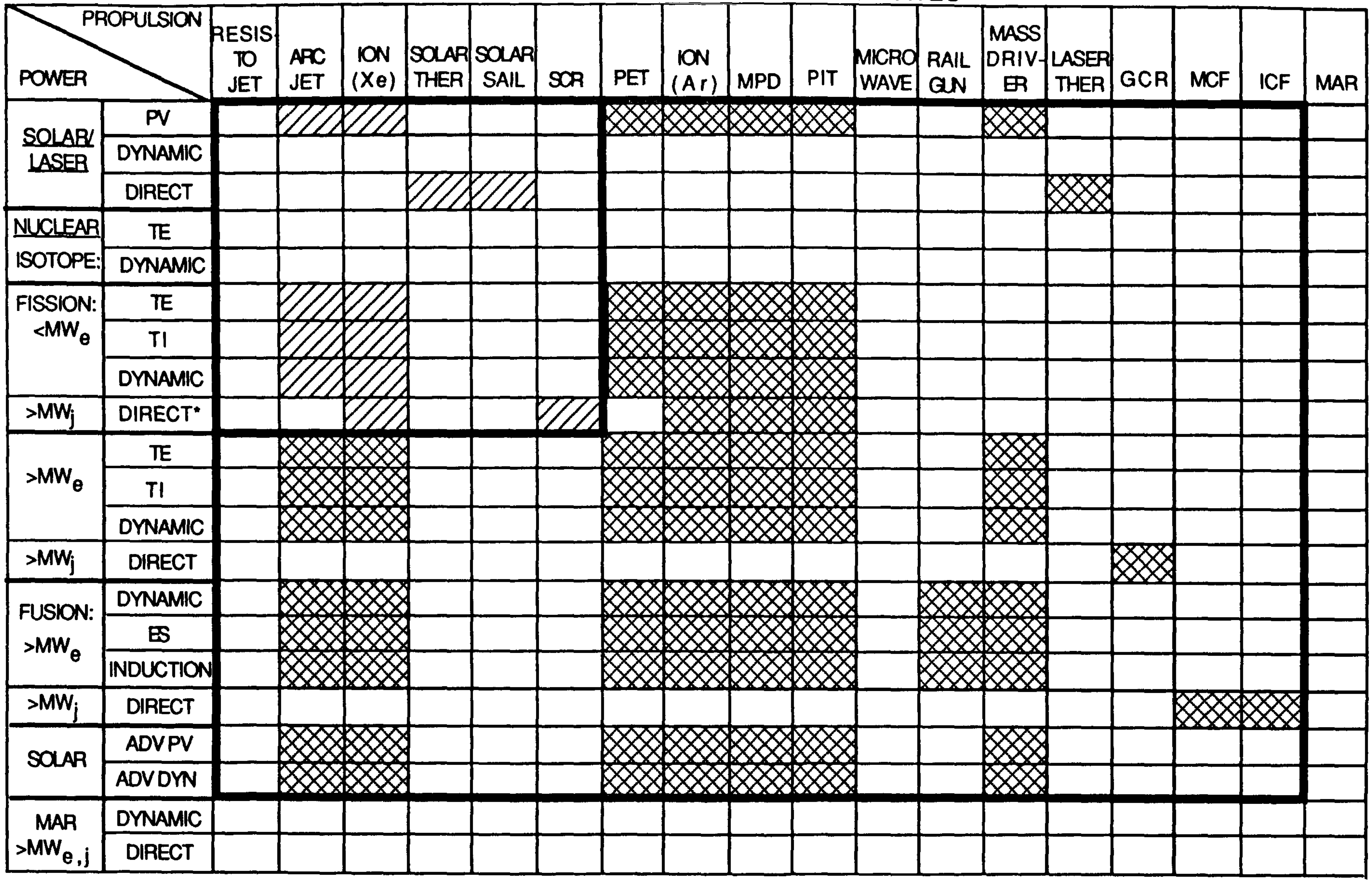
POWER	PROPULSION	RESISTO JET	ARC JET	ION (Xe)	SOLAR THER	SOLAR SAIL	SCR	PET	ION (Ar)	MPD	PIT	MICRO WAVE	RAIL GUN	MASS DRIVER	LASER THER	GCR	MCF	ICF	MAR
SOLAR/ LASER	PV																		
	DYNAMIC																		
	DIRECT																		
NUCLEAR ISOTOPE:	TE																		
	DYNAMIC																		
FISSION: $<MW_e$	TE																		
	TI																		
	DYNAMIC																		
$>MW_j$	DIRECT*																		
$>MW_e$	TE																		
	TI																		
	DYNAMIC																		
$>MW_j$	DIRECT																		
FUSION: $>MW_e$	DYNAMIC																		
	ES																		
	INDUCTION																		
$>MW_j$	DIRECT																		
SOLAR	ADV PV																		
	ADV DYN																		
MAR $>MW_{e,j}$	DYNAMIC																		
	DIRECT																		

*NOTE: DIRECT NUCLEAR ROCKETS CAN BE INTEGRATED WITH POWER CONVERSION AND EP TO OBTAIN A "HYBRID" SYSTEM WITH THE ADVANTAGES OF BOTH SYSTEMS.

NEAR TERM (10 - 15 YEARS)

LATER TERM (> 15 YEARS)

ATTRACTIVE CARGO SYSTEM CANDIDATES



*NOTE: DIRECT NUCLEAR ROCKETS CAN BE INTEGRATED WITH POWER CONVERSION AND EP TO OBTAIN A "HYBRID" SYSTEM WITH THE ADVANTAGES OF BOTH SYSTEMS.

NEAR TERM (10 - 15 YEARS)   LATER TERM (> 15 YEARS)

ADDITIONAL CONSIDERATIONS FOR SURVIVING SYSTEMS

SOLAR ELECTRIC PROPULSION (SEP)

ADVANTAGES

- SPACE SOLAR POWER IS A WELL-DEVELOPED TECHNOLOGY
- NO RADIOACTIVITY

DISADVANTAGES

- POWER LEVEL DECREASES WITH INCREASING DISTANCE FROM SUN
- DEGRADATION IN VAN ALLEN RADIATION BELT
- SOLAR ENERGY COLLECTION REQUIRES LARGE SURFACE AREA, PARTICULARLY AT HIGH POWER
- OPERATION IN PLANETARY SHADOW REQUIRES ENERGY STORAGE (A MASS PENALTY) OR REDUCES AVERAGE POWER
- SUN TRACKING AND ARRAY POINTING ADD SYSTEM AND TRAJECTORY COMPLEXITY
- LOW THRUST/WEIGHT REQUIRES EXTENDED PERIODS OF OPERATION
- RELIABILITY DEPENDS ON COLLECTOR, POWER CONVERSION, AND THRUSTERS, IMPOSING STRINGENT DESIGN REQUIREMENTS

ADDITIONAL CONSIDERATIONS FOR SURVIVING SYSTEMS

NUCLEAR ELECTRIC PROPULSION (NEP)

ADVANTAGES

- COMPACT POWER SOURCE
- POWER LEVEL INDEPENDENT OF ENVIRONMENT (EXAMPLE: SUN, RADIATION BELT)
- FAVORABLE MASS SCALING TO HIGH POWERS
- ADVANTAGEOUS FOR MISSIONS WHERE REACTOR IS PART OF THE PAYLOAD
- HYBRID SYSTEM SHOWS ADVANTAGES OVER DIRECT NUCLEAR OR NEP

DISADVANTAGES

- RADIATION REQUIRES SHIELDING FOR MEN AND INSTRUMENTS
- RADIATORS REQUIRE LARGE SURFACE AREA, PARTICULARLY AT HIGH POWER
- LOW THRUST/WEIGHT REQUIRES EXTENDED PERIODS OF OPERATION
- RELIABILITY DEPENDS ON REACTOR, POWER CONVERSION, AND THRUSTERS, IMPOSING STRINGENT DESIGN REQUIREMENTS

ADDITIONAL CONSIDERATIONS FOR SURVIVING SYSTEMS

SOLID CORE THERMAL ROCKET

ADVANTAGES

- TECHNOLOGY DEVELOPED (NERVA PROGRAM: 10 YEARS/$1.5 BILLION INVESTED)
- OPERATION OF 19 REACTORS DEMONSTRATED:
  - CORE DESIGN
  - NON-NUCLEAR COMPONENTS
  - A VARIETY OF CONTROL/SAFETY SYSTEMS
- FLIGHT ENGINE DESIGNED BUT NOT TESTED
- FACTOR OF 2 ADVANTAGE IN I_{sp} OVER CHEMICAL PROVIDES ROBUSTNESS (LOWER INITIAL MASS, SHORTENED TRIP TIME, PROPULSIVE BRAKING OPTION)

DISADVANTAGES

- LARGER ENGINE MASS THAN CHEMICAL
- RADIATION POSES PROBLEMS FOR LOCAL MANNED OPERATION IMMEDIATELY AFTER SHUTDOWN
- "SPECIAL" DOCKING FACILITIES REQUIRED
- RADIATION RAISES ENVIRONMENTAL CONCERNS ABOUT ENGINE SAFETY DURING STARTUP IN LEO AND THE USE OF AEROBRAKING

ADDITIONAL CONSIDERATIONS FOR SURVIVING SYSTEMS

HYBRID NUCLEAR THERMAL ROCKET

ADVANTAGES

- HYBRID POSSESSES ADVANTAGES OF BOTH HIGH THRUST/WEIGHT AND HIGH PAYLOAD CAPABILITY SYSTEMS
- TECHNOLOGIES DEMONSTRATED IN NERVA/EP PROGRAMS
- DUAL-MODE OPERATION (FOR THRUST/POWER GENERATION) MORE EFFECTIVELY UTILIZES INHERENT ENERGY POTENTIAL OF ENGINE
- AT HIGH POWER LEVELS, CHARACTERISTICS OF ADVANCED FISSION ROCKETS (GCR) MAY BE POSSIBLE (SHORT TRIP TIME)

DISADVANTAGES

- POWER CONVERSION SYSTEM ADDS WEIGHT TO SOLID CORE ROCKET
- HIGH POWER (MWe) OPERATION CAN SIGNIFICANTLY ALTER REACTOR DESIGN
- TECHNOLOGICAL FEASIBILITY REMAINS TO BE DEMONSTRATED

ADDITIONAL CONSIDERATIONS FOR SURVIVING SYSTEMS

SOLAR/LASER THERMAL PROPULSION

ADVANTAGES

- NO POWER SOURCE ON BOARD
- NO RADIOACTIVITY
- SOLAR THERMAL CONCEPT EXPERIMENTALLY DEMONSTRATED

DISADVANTAGES

- POINTING AND TRACKING REQUIREMENTS ADD SYSTEM AND TRAJECTORY COMPLEXITY
- LASER SOURCE REQUIRES LARGE INPUT POWER LEVELS
- MULTIPLE LASERS REQUIRED FOR CONTINUOUS ACCELERATION NEAR EARTH
- LASER CONCEPT STILL EXPERIMENTAL
- SOLAR THERMAL I_{SP} LIMITED BY MATERIALS

ADDITIONAL CONSIDERATIONS FOR SURVIVING SYSTEMS

GAS CORE THERMAL ROCKET

ADVANTAGES

- OPERATION OF URANIUM FUEL IN HIGH TEMPERATURE GASEOUS STATE ALLOWS HIGH I_{sp} OPERATION WITH SIMULTANEOUS HIGH THRUST
- ONLY FISSION ROCKET CONCEPT IDENTIFIED CAPABLE OF 60 DAY ROUND TRIP COURIER MISSION TO MARS
- THE GCR'S ABILITY TO DISPOSE OF RADIOLOGICALLY DANGEROUS FISSION PRODUCTS IN SPACE (WITH WIDE DISPERSAL) SIGNIFICANTLY REDUCES POST-SHUTDOWN HAZARDS
- BOTH CLOSED CYCLE (NO FUEL LOSS) AND OPEN CYCLE CONCEPTS HAVE BEEN STUDIED ANALYTICALLY AND EXPERIMENTALLY (NERVA PROGRAM)
- KEY OPERATIONAL FEATURES OF BOTH CONCEPTS DEMONSTRATED IN EXPERIMENTS (CRITICALITY OF GASEOUS UF_6 DEMONSTRATED)

DISADVANTAGES

- LOSS OF URANIUM (~ 1/4 - 1% OF PROPELLANT FLOW RATE) IN OPEN CYCLE CONCEPT INCREASES FUEL COSTS AND LEADS TO ENVIRONMENTAL CONCERNS FOR SYSTEMS USED IN NEAR EARTH ORBIT
- SIGNIFICANT CONCEPT TESTING UNDER NUCLEAR CONDITIONS REMAINS TO BE DONE

ADDITIONAL CONSIDERATION FOR SURVIVING SYSTEMS

FUSION PROPULSION (MAGNETIC/INERTIAL)

ADVANTAGES

- ABUNDANT FUSION FUEL (10 TRILLION TONS OF DEUTERIUM IN EARTH'S OCEANS)
- HIGH I_{sp} INHERENT IN FUSION-GRADE PLASMAS
- THE HIGH SPECIFIC POWER/IMPULSE OF IFR'S CAN RESULT IN ROUND TRIP TRAVEL TIMES TO PLUTO OF < 20 MONTHS
- ADVANCED-FUEL FUSION SYSTEMS (USING DEUTERIUM AND HELIUM {He-3}) PRODUCE MINIMAL NEUTRON RADIATION (LEADS TO LOWER INITIAL WEIGHTS FOR MFR'S AND HIGHER SPECIFIC POWERS)
- THE USE OF "SPIN POLARIZED" DHe^3 FUEL MAY RESULT IN "CLEAN" FUSION ROCKETS (NO NEUTRON RADIATION)
- NASA CAN CAPITALIZE ON LARGE RESEACH EFFORT IN FUSION (~ $1 BILLION WORLDWIDE/YEAR)

DISADVANTAGES

- MAGNETIC FUSION SYSTEMS REQUIRE HEAVY SUPERCONDUCTING MAGNETS AND COMPLICATED PLASMA EXTRACTION TECHNIQUES
- INERTIAL FUSION SYSTEMS REQUIRE DRIVERS WITH HIGH EFFICIENCY, ENERGY, OPERATING TEMPERATURE AND REP RATE (DRIVERS POSSESSING ALL OF THESE FEATURES DO NOT CURRENTLY EXIST)
- PROPULSION-SPECIFIC (LIGHTWEIGHT) TECHNOLOGIES ARE AT LEAST TWO DECADES AWAY

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ADDITIONAL CONSIDERATIONS FOR SURVIVING SYSTEMS

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PULSED FISSION ROCKET (ORION)

ADVANTAGES

- HIGH I_{sp} OPERATION POSSIBLE WITHOUT MAGNETIC NOZZLE DUE TO SHORT INTERACTION TIME OF PULSE WITH VEHICLE
- CAN UTILIZE ESTABLISHED TECHNOLOGIES
- CONCEPT FEASIBILITY DEMONSTRATED USING CONSECUTIVE CHEMICAL EXPLOSIONS

DISADVANTAGES

- CRITICALITY REQUIREMENTS LIMIT MINIMUM BOMBLET YIELD (~0.01 kTONS TNT)
- LARGE VEHICLES REQUIRED TO EFFICIENTLY HANDLE ENERGY
- POLITICAL/ENVIRONMENTAL PROBLEMS ASSOCIATED WITH FISSION BOMB USAGE IN SPACE
- ENGINE TESTING VIRTUALLY RULED OUT DUE TO NUCLEAR TEST BAN TREATY

SUMMARY

- THIS TASK HAS BEEN RESPONSIVE TO THE REQUEST TO EVALUATE ADVANCED PROPULSION/POWER CONCEPTS. AN EXTENSIVE DATA BASE HAS BEEN ASSEMBLED AND USED TO DERIVE TWO SUMMARY MATRICES (ONE SPRINT, ONE CARGO) OF CREDIBLE PROPULSION SYSTEMS

- CONCEPTS WERE IDENTIFIED USING THREE FILTERS

  (1) TIME: NEAR TERM
  LATER TERM

  (2) SYSTEM PERFORMANCE: COMPARISONS MADE TO CHEMICAL PROPULSION AND TO OTHER ADVANCED PROPULSION SYSTEMS

  (3) MISSION COMPATIBILITY: SYSTEM CAPABILITIES FOR MISSION GOALS (HIGH PAYLOAD FRACTION, SHORT TRIP TIME)

- FURTHER SCREENING IS MISSION SPECIFIC; HOWEVER, SOME POTENTIAL ADVANTAGES/DISADVANTAGES IN THIS REGARD HAVE BEEN PRESENTED

Are We Being Too Timid In Our Development of Space Propulsion Technology?

Aeropropulsion	Space Propulsion
• **Subsonic (~500 mph)** ex: Commercial Jets	• **Chemical Rockets ($\lesssim$ 500 s)** ex: LOX/LH2 Engines
• **Supersonic (Mach 1–2)** ex: Boeing SST Concorde	• **Solid Core NTRs (~800 – 1000 s)** ex: NERVA/ROVER PROGRAMS - NRX/XE ~1100 MWt - PHOEBUS – 2A ~5000 MWt
• **Hypersonic (~5000 mph)** ex: NASP Orient Express	• **Gas Core NTRs (~1500–6000 s)** ex: Closed Cycle Nuclear Light Bulb Engine Open Cycle GCR (W/ext. Space Radiator)

SUPPORTING INFORMATION
(APPENDIX)

- REFERENCES
- DEFINITIONS OF KEY PROPULSION SYSTEM PERFORMANCE PARAMETERS
- STATE-OF-THE-ART DATA SUMMARY (FOR VARIOUS PROPULSION SYSTEMS AND TECHNOLOGIES)
- PROS AND CONS CHARTS FOR EP AND POWER SYSTEM TECHNOLOGIES
- ORGANIZATIONS (GOVERNMENT AGENCIES/INDUSTRY/UNIVERSITIES) ADVOCATING/STUDYING DIFFERENT CONCEPTS
- DATA USED IN SYSTEMS ASSESSMENT

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KEY PROPULSION SYSTEM PERFORMANCE PARAMETERS

- SPECIFIC IMPULSE: $I_{sp}(s)$; $V_{ex} \equiv g_o I_{sp}$ = EXHAUST VELOCITY
- THRUST: $F(N) = \dot{m}_p(kg/s) g_o I_{sp}$; $\dot{m}_p$ = PROPELLANT FLOW RATE
- JET POWER: $P_{jet}(kW_j) = 4.9 \times 10^{-3}\ F(N)\ I_{sp}(s)$
- ENGINE SPECIFIC POWER

  - DIRECT THRUST: $\alpha_p(kW_j/kg) \equiv P_{jet}/M_e$; M_e = TOTAL ENGINE SYSTEM MASS
  - EP SYSTEM: $\alpha_p(kW_j/kg) \equiv \eta_{ej} P_e / M_e \equiv \eta_{ej}/\alpha_m (kg/kW_e)$
  - BEAMED ENERGY SYSTEM: $\alpha_p(kW_j/kg) \equiv (\eta_{elj} P_l$ OR $\eta_{sj} P_s)/M_e$

  WHERE -

  $P_{e,l,s}$ = ELECTRICAL, LASER, OR SOLAR POWER
  $\eta_{e,l,sj}$ = CONVERSION EFFICIENCY OF ELECTRICAL, LASER, OR SOLAR POWER TO JET POWER

KEY PROPULSION SYSTEM PERFORMANCE PARAMETERS (Continued)

- THRUST-TO-ENGINE WEIGHT RATIO

$$F/g_o M_e \equiv \frac{2000}{g_o 2} \quad \frac{\alpha_p (kW_j/kg)}{I_{sp}(s)}$$

- GROSS PROPULSION SYSTEM POWER

$P_g = P_{jet}/\eta_j$ FOR DIRECT THRUST SYSTEMS

$= P_e/\eta_e$ FOR ELECTRIC PROPULSION SYSTEMS

$= P_l$ OR P_s FOR LASER OR SOLAR BEAMED ENERGY SYSTEMS

$\eta_{e,j}$ ARE THE ELECTRICAL AND JET POWER CONVERSION EFFICIENCIES

- PROPULSION SYSTEM REQUIRED OPERATIONAL LIFETIME: τ_{op}

NOTE: THE FUNCTIONAL LIFETIME OF AN ENGINE CAN BE LESS THAN τ_{op} IMPLYING A NEED FOR A MULTIPLE ENGINE PROPULSION SYSTEM

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NEAR TERM PROPULSION SYSTEM STATE-OF-THE-ART DATA SUMMARY

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PROPULSION CONCEPT / PERFORMANCE / *[DEMONSTRATED]*	I_{sp} (ks)	α_m (kg/kW$_e$)	$\eta_{e,l,sj}$ (%)	α_p (kWj/kg)	F/g_oM_e	F/N[a] (N)	P_{jet}/N[a] (kWj)	P_g (MW)	τ_{op} (hrs.)
CHEMICAL: SHUTTLE MAIN ENGINES	***0.455***	NA[b]	NA	***1635***	***75***	***2.08x10^6***	***4.64x10^6***	***≥5000***	7.5
DIRECT FISSION: SCR/NERVA PROGRAM- Ref.2	***0.85*** -1.2	NA	NA	***~120*** ***-435***	***3.3*** ***-10.7***	***2.45x10^5*** ***-1.12x10^6***	***9.12x10^5*** ***-3.6x10^6***	***1100*** ***-5000***	***1.0*** ***-10.0***
LASER THERMAL (OTV-Ref.4)	0.9-1.5 (1.5)	NA	10-80 (46)	(0.26)	($3.7x10^{-3}$)	(63)	(466)	(1.0)	(684)
SOLAR THERMAL (OTV-Ref.6)	0.8-1.2 (0.9)	NA	(50)	(0.28)	($6.4x10^{-3}$)	(222)	(980)	(2.0)	(720)
SOLAR ELECTRIC:ION (LUNAR FERRY-Ref.7)	***3.5-10.0*** (4.5)	(26)	***60-80*** (75)	($3x10^{-2}$)	($1.4x10^{-4}$)	***.2***-10 (10/10)	***3.5***-500 (225/10)	(0.3)	***≤10^4*** (~$9x10^3$)
NUCLEAR ELECTRIC:ION (NEPTUNE PROBE-Ref.8)	***3.5-10.0*** (5.3)	(44)	***60-80*** (77)	($1.7x10^{-2}$)	($6.6x10^{-5}$)	***.2***-10 (2.84/5)	***3.5***-500 (74/5)	(1.4)	***≤10^4*** ($5x10^4$)
NUCLEAR ELECTRIC:ARCJET (OTV-Ref.9)	***0.8***-1.2 (1.0)	(30)	***30-50*** (45)	($7.5x10^{-3}$)	($1.6x10^{-4}$)	***10*** (8.4/4)	***10***-50 (41/4)	(1.4)	***500***-2000 (~$3x10^3$)
DUAL MODE NUCLEAR (SCR/ION-Ref.11)	(0.895) / 5.3	(34.5)	***77***	***(64)*** / $2.1x10^{-2}$	***(1.5)*** / $8.4x10^{-5}$	***(5x10^4)*** / 2.84	***(2.2x10^5)*** / 74	***(263)*** / 0.4	NA

[a] $F, P_{jet}/N$ - THRUST, JET POWER PER UNIT ENGINE

[b] NOT APPLICABLE OR DATA NOT AVAILABLE

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PROJECTED PERFORMANCE FOR LATER TERM PROPULSION SYSTEMS

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PROPULSION CONCEPT / PERFORMANCE [DEMONSTRATED]	I_{sp} (ks)	α_m (kg/kW_e)	$\eta_{e,l,sj}$ (%)	α_p (kWj/kg)	F/g_oM_e	F/N[a] (kN)	P_{jet}/N[a] (MWj)	P_g (MW)	τ_{op} (hrs.)
SOLAR SAIL (MARS CARGO MISSION-Ref.21[c])	30,600	NA[b]	NA	(292)	$(2x10^{-4})$	$(3.7x10^{-2})$	(5600)	NA	$(3.7x10^4)$
NUCLEAR ELECTRIC:MPD (PEGASUS-Ref.13)	2-5 (5)	5-10 (8.5)	30-50 (50)	$(6x10^{-2})$	$(2.5x10^{-4})$	0.01-0.2 (0.122)	0.25-.5 (3)	2-40 (27)	10^4-$2x10^4$ $(1.24x10^4)$
DIRECT FISSION: GCR/MMM (Ref.14)	3-7 (5.7)	NA	NA	10-50 (47)	.04-0.2 (0.17)	22-440 (220)	540-11000 (6150)	750-15000 (8500)	(109)
DUAL MODE NUCLEAR (SCR/MPD-Ref.11)	(0.98) / 5	(11)	50	(20) / $4.6x10^{-2}$	(0.42) / $1.9x10^{-4}$	(45.4) / $2x10^{-2}$	(218) / 0.5	(260) / 4	NA
PULSED FISSION (ORION-Ref.16)	2-1.0 (2.5)	NA	NA	(470)	(3.9)	(3500)	(43,000)	(NA)	(~1)
FUSION PROPULSION:MCF (MCF/MMM-Ref.17)	10-200 (20)	NA	NA	~2-10 (5.75)	$(6.3x10^{-3})$	(60)	(6000)	(7500)	$(2x10^3)$
ICF (ICF/RT PLUTO MISSION-Ref.18)	20-1000 (270)	NA	NA	(110)	$(8.4x10^{-3})$	(40)	(53,000)	(200,000)	$(1.4x10^4)$

a $F, P_{jet}/N$ - THRUST, JET POWER PER UNIT ENGINE
b NOT APPLICABLE OR DATA NOT AVAILABLE
c PERFORMANCE PERAMETERS GIVEN AT 1 A.U.

NOTATION: SCR,GCR-SOLID,GAS CORE ROCKET
MCF, ICF-MAGNETIC, INERTIAL FUSION ROCKET
MMM-MANNED MARS MISSION

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DATA SUMMARY FOR NEAR/LATER TERM POWER TECHNOLOGIES

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SOURCE	SYSTEM	AVAIL	PWR RANGE (kWe)	EFF (%)	SP MASS (kg/kWe)	COMMENTS
ISOTOPE	RTG	NEAR	.01 - 1	7	120	MODULAR RTG WITH GPHS
	DIPS	NEAR	1 - 10	28	125	CBC, DUEL LOOP, GPHS
SOLAR	PV-Si	NEAR	1 - 100	6	90	SS MISSION, NO STORE, TOT STRUCT
	PV-AdvPln	NEAR	1 - 100	25	3	LeRC, NO STORAGE, MIN STRUCTURE
	PV-CONC	LATE	1 - 100	25	5	LeRC, NO STORAGE, MIN STRUCTURE
	SD-SOTA	NEAR	10 - 50	33	200	SS MISSION, NO STORE, TOT STRUCT
	SD-Adv	LATE	10 - 50	34	16	LeRC, NO STORAGE, FP STIRLING
NUCLEAR	T/E	NEAR	100 - 1000	7	40	SP-100, SiGe (GaP) CELLS
	DYNAMIC	NEAR	100 - 300	26	40	SP-100, FP STIRLING
	AMTEC	LATE	1000 - 40,000	22	4	MMWe, LIQ METAL REACTOR
	T/I	LATE	1000 - 10,000	13	5	MMWe, IN-CORE
	DYNAMIC	LATE	1000 - 50,000	21	4	MMWe, LMR, K-RANKINE
	MHD	LATE	10,000-100,000	40	5	NERVA DERIVATIVE REACTOR LeRC

NEAR - NEAR TERM TECHNOLOGY READINESS
LATE - FAR TERM TECHNOLOGY READINESS

ADVANCED SPACE ANALYSIS OFFICE

NASA

Lewis Research Center

Space Flight Systems Directorate

DATA SUMMARY FOR NEAR/LATER TERM ELECTRIC PROPULSION TECHNOLOGIES

ASAO

THRUSTER CONCEPT / PERFORMANCE *[DEMONSTRATED]*	I_{sp} (ks)	α_{ep}[a] (kg/kW_e)	η_{ej} (%)	F (N)	P_e (kW_e)	τ[b] (hrs.)
ELECTROTHERMAL: RESISTOJET-Ref.22,23	*0.1-0.4*	5	*50-80*	*0.05-1*	*0.01-30*	*4000*
ARCJET-Ref.9,10	*0.4-1.0*-1.2	5	*25-55* 50-60	*1-10*	*1-100*	*750* 1500
MICROWAVE-Ref.27	*0.1-0.6*	NA	*5-45*	*0.02-0.1*	*0.2-2*	NA
ELECTROSTATIC: ION(X_e)-Ref.29,30	*3.0-5.0*	*10*	*60-70*	*0.2* 2	*3-7*	*>1000* 10,000
ION(A)-Ref.31	7.0-10.0	5	78-80	1-10	50-500	10,000
ELECTROMAGNETIC: MPD-Ref.32,33,34	*1.0-10.0* 1.0-20.0	1	*30* 50	*10-200*	*100-1000*	*500* 2000
PIT-Ref.35	*3.0-5.0*	1	50	10-100	1000- 5000	NA
PET-Ref.28	*1.0-2.0*	5	50	*0.05-1.0*	0.5-50	NA

[a] SPECIFIC MASS OF THRUSTER

[b] OPERATIONAL LIFETIME OF THRUSTER

ADVANCED SPACE ANALYSIS OFFICE

POWER SYSTEM TECHNOLOGIES PROS AND CONS

PHOTOVOLTAIC: ADVANTAGES/DISADVANTAGES

- ADVANTAGES
  - PROVEN, LOW RISK TECHNOLOGY
  - LOW ARRAY SPECIFIC MASS
  - SINGLE CELL FAILURE WILL NOT LEAD TO SYSTEM FAILURE
  - CAPABILITY FOR DEPLOYMENT & RETRACTION
  - CONCENTRATED PV CAN DECREASE SOLAR COLLECTION AREA

- DISADVANTAGES
  - LARGE SOLAR COLLECTION AREA
  - POWER LEVELS RESTRAINED BY AREA
  - LOW CONVERSION EFFICIENCY
  - SOLAR RADIATION DEGRADATION
  - HEAVY BATTERIES REQUIRED FOR SHADE PERIODS

POWER SYSTEM TECHNOLOGIES PROS AND CONS

SOLAR DYNAMIC: ADVANTAGES/DISADVANTAGES

- ADVANTAGES
  - HIGH SYSTEM EFFICIENCY
  - HIGH GROWTH POTENTIAL
  - EVOLUTIONARY STEP TOWARDS NUCLEAR DYNAMIC
  - TERRESTRIAL EXPERIENCE

- DISADVANTAGES
  - UNPROVEN IN SPACE
  - HIGH SYSTEM COMPLEXITY
  - ROTATING MACHINERY
  - COLLECTOR POINTING ACCURACY
  - SYSTEM PERFORMANCE DICTATED BY COMPONENTS

POWER SYSTEM TECHNOLOGIES PROS AND CONS

RADIOISOTOPE THERMOELECTRIC GENERATORS: ADVANTAGES/ DISADVANTAGES

- ADVANTAGES
  - STATIC CONVERSION (NO MOVING PARTS)
  - LONG LIFE CAPABILITY
  - DEMONSTRATED, RELIABLE TECHNOLOGY (APOLLO & VOYAGER PROGRAMS)
  - NO CRITICAL MASS, NO ACTIVE CONTROL REQUIREMENTS
- DISADVANTAGES
  - HEAT SOURCE FUEL IS EXPENSIVE AND SCARCE
  - POWER LEVELS RESTRAINED BY FUEL AVAILIBILITY
  - ENVIRONMENTALLY HAZARDOUS FUEL (ISSUE OF SAFETY AT LAUNCH)
  - LOW CONVERSION EFFICIENCY
  - SHIELDING REQUIRED FOR MANNED APPLICATIONS AND RADIATION SENSITIVE EQUIPMENT

POWER SYSTEM TECHNOLOGIES PROS AND CONS

DYNAMIC ISOTOPE POWER SYSTEMS: ADVANTAGES/DISADVANTAGES

- ADVANTAGES
  - HIGH CONVERSION EFFICIENCY
  - HIGHER POWER LEVELS DUE TO INCREASED EFFICIENCY
  - NO CRITICAL MASS, NO ACTIVE CONTROL REQUIREMENTS
- DISADVANTAGES
  - FUEL REQUIRED IS COSTLY, SCARCE, AND VERY HAZARDOUS
  - DYNAMIC CONVERSION (MOVING PARTS, COMPLEX DESIGN)
  - SYSTEM PERFORMANCE DICTATED BY COMPONENTS
  - SHIELDING REQUIRED FOR MANNED APPLICATIONS AND RADIATION SENSITIVE EQUIPMENT

POWER SYSTEM TECHNOLOGIES PROS AND CONS

NUCLEAR STATIC: ADVANTAGES/DISADVANTAGES

- ADVANTAGES
  - NO MOVING PARTS
  - CONVERTER CONSISTS OF MANY CELLS ARRANGED IN PARALLEL AND SERIES TO INSURE REDUNDANCY AND RELIABILITY
  - SEMICONDUCTING THERMOELECTRIC MATERIALS ARE TECHNOLOGICALLY READY

- DISADVANTAGES
  - LOW CONVERSION EFFICIENCY
  - REQUIRES START-UP POWER
  - REACTOR SWELLING LEADS TO DIFFICULTY IN MAINTAINING CELL GAP CLEARANCE
  - SHIELDING REQUIRED FOR MANNED APPLICATIONS AND RADIATION SENSITIVE EQUIPMENT

POWER SYSTEM TECHNOLOGIES PROS AND CONS

NUCLEAR DYNAMIC: ADVANTAGES/DISADVANTAGES

- ADVANTAGES
  - HIGH CONVERSION EFFICIENCY
  - HIGH POWER DENSITY (kW_e/m^3)
  - ABILITY TO PROVIDE FOR PEAK POWER REQUIREMENTS
- DISADVANTAGES
  - DYNAMIC CONVERSION (MOVING PARTS, COMPLEX DESIGN)
  - LARGE RADIATOR MASS
  - SYSTEM PERFORMANCE DICTATED BY COMPONENTS
  - REQUIRES START-UP POWER
  - SHIELDING REQUIRED FOR MANNED APPLICATIONS AND RADIATION SENSITIVE EQUIPMENT

ELECTRIC PROPULSION TECHNOLOGY PROS AND CONS

ELECTROTHERMAL THRUSTER
RESISTOJETS: ADVANTAGES/DISADVANTAGES

- ADVANTAGES
  - ESTABLISHED TECHNOLOGY
  - FLIGHT READY
  - PRECISE THRUST LEVELS
  - "OMNIVOROUS" - CAN USE NEARLY ANY PROPELLANT
  - COMPACT/SIMPLE
  - DC POWER
- DISADVANTAGES
  - ISP LIMITED BY MATERIAL TEMPERATURES
  - LONGER LIFETIMES REQUIRED

ELECTRIC PROPULSION TECHNOLOGY PROS AND CONS

ELECTROTHERMAL THRUSTERS (CONT'D)
ARCJETS: ADVANTAGES/DISADVANTAGES

- ADVANTAGES
  - COMPACT/SIMPLE
  - DC POWER
  - OMNIVOROUS
  - NEUTRAL PLASMA

- DISADVANTAGES
  - PRESENTLY UNDER DEVELOPMENT
  - LIFETIME IN QUESTION
  - ISP LIMITED BY MATERIAL TEMPERATURES

ELECTRIC PROPULSION TECHNOLOGY PROS AND CONS

ELECTROTHERMAL THRUSTERS (CONT'D)
MICROWAVE: ADVANTAGES/DISADVANTAGES

- ADVANTAGES
  - ELECTRODELESS - LONGER LIFE
  - EFFICIENT MICROWAVE GENERATION
  - POTENTIALLY SCALABLE TO HIGH POWER LEVELS
  - NEUTRAL PLASMA

- DISADVANTAGES
  - LABORATORY DEVICE
  - SYSTEM COMPLEXITY
  - THRUSTER WEIGHT AN ISSUE

ELECTRIC PROPULSION TECHNOLOGY PROS AND CONS

ELECTROSTATIC THRUSTERS
ION: ADVANTAGES/DISADVANTAGES

- ADVANTAGES
  - HIGH EFFICIENCY
  - HIGH ISP
  - ESTABLISHED TECHNOLOGY
  - FLIGHT READY
  - "OMNIVOROUS": Hg, Xe, Kr, A
  - POSSIBLE LONG LIFE

- DISADVANTAGES
  - COMPLEX POWER CONDITIONING
  - MUST INCREASE AREA TO INCREASE THRUST
  - LONG LIFE DURING HIGH POWER OPERATION REMAINS UNDEMONSTRATED

ELECTRIC PROPULSION TECHNOLOGY PROS AND CONS

ELECTROMAGNETIC THRUSTER
MPD (STEADY STATE): ADVANTAGES/DISADVANTAGES

- ADVANTAGES
  - COMPACT/SIMPLE
  - NEUTRAL PLASMA
  - POTENTIAL FOR HIGH ISP AND HIGH EFFICIENCY

- DISADVANTAGES
  - LABORATORY DEVICE
  - QUESTIONABLE LIFETIME OF ELECTRODES

WHO'S DOING WHAT?

A VARIETY OF ORGANIZATIONS (GOVERNMENT AGENCIES/INDUSTRY/UNIVERSITIES) ARE ADVOCATING/STUDYING DIFFERENT PROPULSION CONCEPTS.

EXAMPLE: A CONSORTIUM HAS BEEN FORMED TO PROMOTE AND DEVELOP A NERVA-DERIVATIVE ADVANCED NUCLEAR THERMAL ROCKET ENGINE. (WORK SUPPORTED BY AIR FORCE ASTRONAUTICS LABORATORY.)

Idaho National Engineering Laboratory
Program Integration/Test Facility

MARTIN MARIETTA
Stage Integration/Mission Analysis

Rockwell International
Rocketdyne Division
Engine System

SAIC - Science Applications International Corporation
Flight Safety/Mission Analysis

Westinghouse Electric Corporation
Advanced Power Systems Divisions
Nuclear Subsystem

WHO'S DOING WHAT? (CONTINUED)

NUCLEAR FISSION THERMAL ROCKET (OTHERS INCLUDE)

- LOS ALAMOS NATIONAL LABORATORY (LANL)
- ARGONNE NATIONAL LABORATORY (ANL)
- BROOKHAVEN NATIONAL LABORATORY (BNL)
- GRUMMAN AEROSPACE
- GENERAL ELECTRIC COMPANY
- BABCOCK AND WILCOX COMPANY
- AEROJET TECH SYSTEMS COMPANY
- MCDONNEL DOUGLAS

HYBRID NUCLEAR THERMAL ROCKET (THRUST PLUS ELECTRICITY)

- ROCKWELL INTERNATIONAL (ROCKETDYNE DIVISION)
- WESTINGHOUSE (ADVANCED POWER SYSTEMS DIVISION)
- MARTIN MARIETTA
- INEL
- AEROJET TECH SYSTEMS COMPANY (DURING NERVA PROGRAM)

GAS CORE FISSION THERMAL ROCKET

- INNOVATIVE NUCLEAR SPACE POWER INSTITUTE (INSPI) AT CALIFORNIA STATE UNIVERSITY - LONG BEACH
- SDIO
- INEL
- UNIVERSITY OF FLORIDA
- UNITED TECHNOLOGY RESEARCH LABORATORIES

WHO'S DOING WHAT? (CONTINUED)

PULSED FISSION ROCKET (ORION)

- GENERAL ATOMIC (DID SIGNIFICANT WORK DURING ORION PROGRAM FUNDED BY BOTH NASA & ASAF).

FUSION PROPULSION (BOTH MAGNETIC AND INERTIAL)

- AIR FORCE ASTRONAUTICS LAB (AFAL)
- JET PROPULSION LABORATORY (JPL)
- LAWRENCE LIVERMORE NATIONAL LAB (LLNL)
- NASA/LeRC
- ROCKETDYNE
- UNIVERSITY OF WISCONSIN
- GENERAL ATOMIC
- MCDONNELL DOUGLAS

WHO'S DOING WHAT? (CONTINUED)

ARCJETS

- NASA LeRC
- AEROJET TECH SYSTEMS/TECHNION
- ROCKET RESEARCH CENTER (RRC)
- AIR FORCE ASTRONAUTICS LAB (AFAL)
- JPL
- PRINCETON
- JAPAN

ION

- NASA LeRC
- JPL
- HUGHES
- BOEING
- ELECTRIC PROPULSION LABORATORY
- JAPAN
- GERMANY - GIESSEN INSTITUTE OF PHYSICS
- ENGLAND
- COLORADO STATE

WHO'S DOING WHAT? (CONTINUED)

MPD
- NASA LeRC
- JPL
- ESA
- GERMANY (UNIVERSITY OF STUTTGART)
- JAPAN
- PRINCETON
- MIT

SOLAR THERMAL
- AFAL
- ROCKETDYNE

SOLAR SAIL
- WORLD SPACE FOUNDATION
- JPL

WHO'S DOING WHAT? (CONTINUED)

PIT
- TRW

PET
- NASA LeRC
- GT DEVICES, INCORPORATED

MICROWAVE
- NASA LeRC
- MICHIGAN STATE
- PENN STATE

LASER THERMAL
- LAWRENCE LIVERMORE LAB (LLL)
- PHYSICAL APPLICATION, INCORPORATED
- KMS FUSION
- LOCKHEED MISSILES & SPACE
- UNIVERSITY OF TENNESSEE SPACE INSTITUTE
- UNIVERSITY OF ILLINOIS

WHO'S DOING WHAT? (CONTINUED)

RTG

- US DOD
- NASA - LeRC
- GENERAL ELECTRIC COMPANY
- TELEDYNE ENERGY SYSTEMS
- JPL

DIPS

- US DOD
- NASA - LeRC
- ROCKWELL, ROCKETDYNE DIVISION
- GARRETT FLUID SYSTEMS
- SUNDSTRAND CORPORATION
- GENERAL ELECTRIC COMPANY
- GRUMMAN SPACE SYSTEMS

WHO'S DOING WHAT? (CONTINUED)

SOLAR DYNAMICS

- NASA - LeRC
- ROCKWELL, ROCKETDYNE DIVISION
- GARRET FLUID SYSTEMS
- SUNDSTRAND CORPORATION
- GRUMMAN SPACE SYSTEMS
- MECHANICAL TECHNOLOGY, INCORPORATED

PV

- NASA - LeRC
- ROCKWELL, ROCKETDYNE DIVISION
- FORD AEROSPACE
- LOCKHEED MISSILES AND SPACE
- NASA - MSFC

WHO'S DOING WHAT? (CONTINUED)

SP-100

- NASA - LeRC
- US DOD
- GENERAL ELECTRIC COMPANY
- US DOD
- THERMO ELECTRON CORPORATION
- GA TECHNOLOGIES
- MECHANICAL TECHNOLOGY, INCORPORATED

SNAP-DYN

- ROCKWELL, ROCKETDYNE DIVISION
- GARRET FLUID SYSTEMS COMPANY
- SUNDSTRAND CORPORATION

WHO'S DOING WHAT? (CONTINUED)

MMWe

- GENERAL ATOMIC TECHNOLOGIES
- WESTINGHOUSE CORPORATION
- GARRET
- ORNL
- ROCKWELL
- FORD AEROSPACE
- THERMACORE
- JPL
- NASA - LeRC
- THERMO-ELECTRON CORPORATION

DATA USED IN SYSTEMS ASSESSMENT

INPUT STATUS
D

SPECIFIC MASS OF PROPULSION SYSTEMS

	CONV.	CONFIG.	Resistojet	Arcjet	Ion(Xe)	Sol Therm.	Solar Sail	ECR
ISOTOPE	T/E	PbTe	733.00	1000.00	633.43			
		SiGe(GaP)	290.33	212.90	189.71			
		AMTEC						
	RANKINE	CPC	300.03	300.04	183.74			
	HYBRID	CPC/TE						
	BRAYTON	CBC	310.07	329.00	192.00			
SOLAR	DIRECT	DIRECT						
	PV	Si-PLN	448.07	867.81	300.04	4.00	3.4E-03	
		Si-PLN	166.21	247.82	146.79			
		Si-PLN	19.00	22.90	29.00			
		GaAs-PLN	13.83	26.79	19.00			
		GaAs-CONC						
		Si-ALPHA						
		InP						
		CASCADE						
		THERMAL						
	RANKINE	CPC	418.40	818.00	300.01			
	RANKINE	CPC	400.33	800.00	400.00			
	BRAYTON	CBC	300.23	840.38	318.01			
	BRAYTON	CBC	421.17	631.79	300.14			
	STIRLING	FP	439.00	932.00	300.00			
	STIRLING	FP						
SNAP-DYN	RANKINE	CPC	194.17	201.20	172.87			
	BRAYTON	CBC	142.20	213.30	129.03			
	STIRLING	FP						
SP-100 FISSION INDIRECT	T/E	SiGe(GaP)	90.11	84.10	99.24			
		SiGe(GaP)	75.00	112.00	71.43			
	T/I	IN-CORE	97.00	86.93	90.00			
	STIRLING	FP	79.24	118.00	75.00			
	BRAYTON	CBC	77.00	116.00	73.00			
MMW ECR	RANKINE	KRC						
	BRAYTON	CBC						
	T/I	IN-CORE						
	T/E	AM						
	MHD	TURBOMHD						
ECR	MHD	BRAYTON						
FUSION INDIRECT	ELECTROSTATIC							
FUSION DIRECT	ECR							
	GCR							6.3E-03
FUSION DIRECT	MCF							
	ICF							
Isp (s)			100.00	400.0	3000.00	800.0	1000.0	800.0
			400.00	1200.0	5000.00	1000.0	30000.0	1200.0

	Ion(Ar)	MPD	PIT	PET	Microwave	Laser Ther*	ECR	MCF	ICF	RAILGUN	MASS DRIVER
Isp (s)	7000.00	1000.00	3000.00	1000.0	100.0	1000.0	3000.0	10000.0	20000.0	1000.0	1000.0
	10000.00	20000.00	5000.00	2000.0	800.0	2000.0	7000.0	100000.0	1000000.0	2000.0	2000.0

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INPUT STATUS
P

SPECIFIC MASS OF PROPULSION SYSTEMS

	CONV.	CONFIG.	Resistojet	Arcjet	Ion(Xe)	Sol Therm.	Solar Sail	SCR	Ion(Ar)	MPD	PIT	PET	Microwave	Laser Ther'	GCR	MCF	ICF	RAILGUN	MASS DRIVER
ISOTOPE	T/E	PbTe							648.90	871.50	871.00	870.00	1402.07					1018.20	900.00
		SiGe(GaP)							133.00	242.00	242.00	250.00	433.33					800.00	280.00
		AMTEC	100.33	102.50	100.00				66.00	122.00	122.00	130.00	233.33					320.00	160.00
	RANKINE	CPC							153.27	238.23	238.23	247.23	420.72					804.47	277.22
	HYBRID	CPC/TE	102.34	273.51	103.43				135.50	210.81	210.81	218.81	381.39					407.01	240.81
	BRAYTON	CBC							101.25	252.00	252.00	260.00	490.00					800.00	200.00
SOLAR	DIRECT	DIRECT												10.00					
	PV	Si-PLN							337.90	520.00	520.00	534.00	808.02					1120.10	864.00
		Si-PLN							132.00	100.20	100.24	108.20	347.08					496.01	328.20
		Si-PLN							10.00	10.00	10.00	18.00	46.07					80.00	40.00
		GaAs-PLN							9.13	8.00	8.00	18.00	44.33					82.20	40.00
		GaAs-CONC	16.67	25.00	21.43				11.25	12.00	12.00	20.00	50.00					100.00	80.00
		Si-ALPHA	13.03	20.75	16.00				9.13	8.00	8.00	18.00	44.33					83.30	40.00
		InP	13.03	20.75	19.00				9.13	8.00	8.00	18.00	44.33					83.30	40.00
		CASCADE	15.00	22.50	20.00				10.00	10.00	10.00	18.00	46.67					56.00	40.00
		THERMAL	91.67	137.50	83.71				67.50	102.00	102.00	110.00	200.00					300.00	140.00
	RANKINE	CPC							300.35	404.40	404.40	402.40	837.40					1044.80	822.40
	RANKINE	CPC							343.25	343.20	343.20	551.20	833.33					1182.40	381.20
	BRAYTON	CBC							298.93	424.20	424.20	432.20	737.13					824.80	462.30
	BRAYTON	CBC							314.03	407.40	407.40	505.40	850.00					1070.80	535.40
	STIRLING	FP							325.90	514.00	514.00	522.00	866.07					1104.00	852.00
	STIRLING	FP	35.00	52.50	37.14				25.00	34.00	34.00	42.00	86.07					144.00	72.00
SNAP-DYN	RANKINE	CPC							144.30	225.00	225.06	233.00	465.00					820.00	203.00
	BRAYTON	CBC							105.40	162.04	162.04	170.04	301.07					481.30	208.04
	STIRLING	FP	134.37	201.55	122.31				89.53	133.24	133.24	181.24	205.40					382.40	151.24
SP-100	T/E	SiGe(GaP)							40.83	80.33	80.33	87.33	126.00					104.80	97.33
FISSION INDIRECT		SiGe(GaP)							55.00	82.00	82.00	88.00	160.07					240.00	120.00
	T/I	IN CORE							42.01	81.22	81.22	89.22	137.03					180.44	80.22
	STIRLING	FP							56.10	87.00	87.00	95.00	175.14					230.17	125.00
	BRAYTON	CBC							57.24	85.50	85.50	93.50	172.85					247.10	125.00
MMW SCR	RANKINE	HRC	15.65	23.75	20.71				10.03	11.00	11.00	19.00	48.33					60.00	40.00
	BRAYTON	CBC	16.33	27.50	22.00				12.50	14.00	14.00	22.00	83.33					104.00	32.00
	T/I	IN-CORE	16.33	27.50	22.00				12.50	14.00	14.00	22.00	83.33					104.00	32.00
	T/E	AM	14.63	22.25	18.00				8.88	8.00	8.00	17.00	48.33					28.00	47.00
SCR	MHD	TURBO&MHD	16.67	25.00	21.43				11.25	12.00	12.00	20.00	50.00					100.00	80.00
FUSION INDIRECT	MHD	BRAYTON	8.83	13.75	14.71				5.30	2.00	2.00	10.00	34.33					81.30	40.50
FISSION DIRECT	ELECTROSTATIC		16.00	15.00	13.71				6.25	4.00	4.00	12.00	30.67					84.00	42.00
	SCR																		
FUSION DIRECT	GCR														3.0E-02				
	MCF															3.0E-01			
	ICF																3.0E-03		
Isp (s)			100.00	400.0	3000.00	800.0	1000.0	850.0	7000.00	1000.00	3000.00	1000.0	150.0	1000.0	3000.0	10000.0	20000.0	1000.0	1000.0
			400.00	1200.0	5000.00	1000.0	30000.0	1200.0	10000.00	20000.00	5000.00	2000.0	800.0	2000.0	7000.0	100000.0	1000000.0	2000.0	2000.0

ADVANCED PROPULSION DATA MATRIX

	Resistojet	Arcjet	Ion(Xe)	Sol Therm'	Solar Sail	SCR	Ion(Ar)	MPD	PIT	PET	Microwave	Laser Ther'	GCR	MCF	ICF	RAILGUN	MASS DRIVER
Isp (s)	100.0	400.0	3000.00	800.0	1000.0	850.0	7000.00	1000.00	3000.00	1000.0	150.0	1000.0	3000.0	10000.0	20000.0	1000.0	1000.0
Isp (s)	400.0	1200.0	5000.00	1000.0	30000.0	1200.0	10000.00	20000.00	5000.00	2000.0	800.0	2000.0	7000.0	100000.0	1000000.0	2000.0	2000.0
α(kg/kWe)	5.00	8.00	10.00	2.0	3.42E-03	8.30E-03	4.00	1.00	1.00	5.0	10.0	5.0	0.020	0.2	0.009	20.0	20.0
Thrust eff (%)	60.00	40.00	70.00	50.0	100.0	72.0	80.00	50.00	50.00	50.0	30.0	50.0	72.0	80.0	26.5	25.0	50.0
POWER (kWe)	1.00	1.00	1.00	1600.0	5.60E+06	1.10E+06	10.00	10.00	1.0	0.5	0.1	1000.0	7.50E+06	7.50E+06	5.00E+07	100000.0	1000000.0
POWER (kWe)	10.00	100.00	10.00	1600.0	5.60E+06	5.00E+06	500.00	10000.0	5000.0	100.0	2.0	10000.0	1.50E+06	7.50E+06	5.00E+07	1000000.0	10000000.0
Thrust (N)	0.5	0.1	0.10	100.0	37.3	2.45E+06	1.00	1.0	10.000	0.1	0.010	100.0	22000.0	60000.0	40000.0	10.0	10.0
Thrust (N)	1.0	10.0	0.20	200.0	37.3	1.12E+06	10.00	200.0	200.0	1.0	1.0	500.0	440000.0			100.0	1000.0
Life (hrs)	10000.0	1500.0	10000.00		3.68E+04	10.0	10000.00	2000.0									
Status	D	D	D	D	D	D	P	P	P	P	P	P	P	P	P	P	P

POWER DATA MATRIX

SOURCE	CONV.	POWER CONFIG.	DATE	SP MASS kG/kWe	THERMAL EFF	RANGE kWe	RANGE kWe	POWER kWe	TEMP K	RAD AR m2/kWe	COLL AR m2/kWe	LIFE YEARS	REMARKS
ISOTOPE	T/E	PbTe	D	434.8	5.0	0.01	1.0	0.07	900.0	28.57		7.0	SNAP 27, APOLLO 12-17
		SiGe(GaP)	D	120.0	7.6	0.01	1.0	0.34	1273.0	4.67		5.0	GE, MOD-RTG (GPHS)
		AMTEC	P	60.0	22.0	1.0	10.0					7.0	LeRC PTD, MARS SURFACE
	RANKINE ORC	ORC	D	116.6	23.3	1.0	10.0	8.0	873.0	8.86		12.0	DIPS, SUNDSTRAND
	HYBRID	ORC/TE	P	104.4	26.8	1.0	10.0	6.0	1100/850	7.40		12.0	DIPS, SUNDSTRAND
	BRAYTON	CBC	D	125.0	28.0	1.0	10.0	6.0	1034.0	5.57		12.0	DIPS, GARRETT
SOLAR	DIRECT		D										
	PV	Si-PLN	D	262.0	3.0	1.0	100.0	20.2	325.0	2.57	24.76	10.0	DR-O2, 2 ARRAYS, 78 PANELS/ARRAY, NIH BATTERIES
		Si-PLN	D	94.1	5.8	1.0	100.0	30.4	325.0	1.32	12.67	10.0	DR-O2, 2 ARRAYS, 78 PANELS/ARRAY, NO STORAGE
		Si-PLN	D	4.00	20.0	1.0	100.0	10.0	325.0			10.0	LeRC PTD, NO STORAGE, MINIMAL STRUCTURE
		GaAs-PLN	D	3.30	25.0	1.0	100.0	10.0	325.0				LeRC PTD, NO STORAGE, MINIMAL STRUCTURE
		GaAs-CONC	P	5.00	25.0	1.0	100.0	10.0	350.0				LeRC PTD, NO STORAGE, MINIMAL STRUCTURE
		Si-ALPHA	P	3.30	15.0	1.0	100.0	10.0	325.0				LeRC PTD, NO STORAGE, MINIMAL STRUCTURE
		InP	P	3.30	23.0	1.0	100.0	10.0	325.0				LeRC PTD, NO STORAGE, MINIMAL STRUCTURE
		CASCADE	P	4.00	30.0	1.0	100.0	10.0	325.0				LeRC PTD, NO STORAGE, MINIMAL STRUCTURE
		THERMAL	P	50.00	30.0	1.0	100.0	25.0	2000.0	0.6		10.0	LeRC PTD, NO STORAGE, MINIMAL STRUCTURE
	RANKINE	ORC	D	241.24	27.5	10.0	50.0	28.2	875.0	6.77	7.86	10.0	ROCKETDYNE, DR-02, NO THERMAL STORAGE
	RANKINE	ORC	D	270.60	27.5	10.0	50.0	28.2	875.0	6.8	7.7	10.0	ROCKETDYNE, DR-02, WITH THERMAL ENERGY STORAGE
	BRAYTON	CBC	D	211.14	33.3	10.0	50.0	28.2	1012.0	5.27	6.32	10.0	ROCKETDYNE, DR-02
	BRAYTON	CBC	D	247.70	33.3	10.0	50.0	28.2	1012.0	5.3	6.3	10.0	ROCKETDYNE, DR-02, WITH THERMAL ENERGY STORAGE
	STIRLING	FP	D	286.00	34.0	10.0	50.0	25.0	1080.0	4.6	6.8	10.0	LeRC PTD WITH THERMAL ENERGY STORAGE
	STIRLING	FP	P	16.0	34.0	10.0	50.0		1080.0	4.6		10.0	LeRC PTD, LUNAR SURFACE, NO THERMAL ENERGY STORAGE
SNAP-DYN	RANKINE	ORC	D	111.50	17.7	10.0	50.0	30.0	728.0	6.07		10.0	SNAP10A, SUNDSTRAND/ROCKETDYNE
	BRAYTON	CBC	D	80.32	19.2	10.0	50.0	50.0	922.0	4.68		10.0	SNAP10A, GARRET/ROCKETDYNE
	STIRLING	FP	P	75.62	19.3	10.0	50.0	50.0	922.0	4.04		10.0	SNAP10A, MTI/ROCKETDYNE
SP-100	T/E	SiGe(GaP)	D	28.66	5.5	100.0	300.0	105.6	1350.0			7.0	GE, ASTRO SPACE DIV
		SiGe(GaP)	D	40.00	7.0	100.0	300.0	105.6	1350.0			7.0	LeRC PTD
	T/I	IN-CORE	D	29.61	8.1	100.0	300.0	100.0	2200.0			7.0	MARTIN MARIETTA
	STIRLING	FP	D	42.54	26.0	100.0	300.0	100.3	1350.0	1.66		7.0	GE, ASTRO SPACE DIV
	BRAYTON	CBC	D	41.79	18.5	100.0	300.0	100.3	1150.0	2.77		7.0	GE, ASTRO SPACE DIV
MMW/SCR	RANKINE	NRC	P	4.50	20.5	1000.0	5.0E+04	1.0E+04	1450.0	0.14		10.0	LMR, ORNL/ROCKWELL
	BRAYTON	CBC	P	6.00	22.0	1000.0	5.0E+04	1.0E+04	1500.0	0.43		10.0	LMR-NERVA DERIVATIVE, WESTINGHOUSE/GARRETT
	T/I	IN-CORE	P	6.00	12.0	1000.0	5.0E+04	1.0E+04	1900.0	0.30		10.0	GENERAL ATOMICS
	T/E	AM	P	3.90	22.0	1000.0	5.0E+04	1.0E+04	1350.0	0.24		10.0	LMR, WESTINGHOUSE/FORD
	MHD	TURBO/MHD	P	5.00	40.00	1.0E+04	1.0E+05	1.0E+04	2500.0	0.2			NDR(NERVA DERIVATIVE REACTOR), LeRC 1972
GCR	MHD	BRAYTON	P	0.30	40.0	1.0E+05	1.0E+06						ROSA AND MYRABO, 1984
FUSION INDIRECT	ELECTROSTATIC		P	1.00	80.0	1.0E+05	1.0E+06						SOAR, FUS.TECH. INST., U. WIS.
FISSION DIRECT	SCR		D	8.30E-03	70.0	1.00E+06	5.00E+06	1.00E+06	2500.0			10.0	NERVA PROGRAM
	GCR		P	2.00E-02	70.0	7.50E+06	1.50E+06						
FUSION DIRECT	MCF		P	0.2		7.50E+06	7.50E+06	7.50E+06					
	ICF		P	9.00E-03		5.00E+07	5.00E+07	5.00E+07					

STATUS (S) D - DEMONSTRATED
P - PROJECTED

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IN-SITU PROPELLANT LEVERAGE ANALYSIS

Alan Willoughby - Analex Corporation
Bruce Cordell (Team Leader) - General Dynamics Space Systems Division

OBJECTIVE:

The objective of this study is to determine the effects of using Lunar, Phobos/Deimos (Ph/D), and Mars propellants in Lunar, Mars, and Lunar/Mars blended scenarios.

METHODOLOGY:

Background:

Current concepts for Lunar/Mars blended scenarios involve the production of Lunar oxygen (lunox), and in situ propellant production on Phobos, Deimos, or Mars (surface or atmosphere). It is important to ascertain which space resource production and use scenarios have the most performance, cost, and strategic benefits.

Key Assumptions:

The key assumptions of this study include: 1) In situ propellants are transported by cryogenic tanker vehicles similar to the Orbital Transfer Vehicle (OTV) designed by General Dynamics (I_{sp} = 485 sec; insulation and meteoroid protection require 10% increase in inert tank weight scaling relations; zero boiloff losses assumed), 2) Aerobrakes are scaled to 15% of entry weight, 3) Payload weights are from Code Z Scenario Requirements Document (SRD) (Lunar-Outpost-to-Mars-Outpost scenario); 3) Both hydrogen (H) and oxygen (O) are produced at Phobos/ Deimos; 4) Conjunction class trajectories are used; 5) Infrastructure build-up is not included in initial cases; and 6) Mass penalties, not operations costs are calculated in these initial cases.

Approach:

GDSS computer models relating in situ propellant production products and sites, delivery trajectories, users and economics are being utilized to provide quantitative insights into space resource utilization for the Moon and Mars. Space Operations Analysis Resource (SOAR) is an interactive, user-friendly computer program (including a library of orbital mechanics routines) used to perform multi-vehicle mission planning. SOAR was initially an Earth-centered simulation. The consideration of Mars and Lunar missions has led to the development of SOAR versions for Lunar and Mars orbital operations (Moon- and Mars-centered, respectively), and interplanetary trajectories (Sun-centered). Space Transportation and Resources (STAR) is a user friendly spreadsheet model that simulates personnel and cargo transportation between an arbitrary number of space transportation nodes. STAR models refueling, staging, and/or payload changes at any node. Vehicle scaling (including aerobrake) relations are input into STAR as are trajectory data (e.g.

ΔV's from SOAR). A Mars-Lunar transportation/resource cost model is being expanded and fully integrated into STAR.

PAYLOAD DELIVERED / VEHICLE WEIGHT RATIOS

Figures 1 and 2 show performance data for each payload destination. For each destination, data is shown for each potential site of propellant production. In all cases the payload delivered is 100 metric tons of propellant. Aerobraking is used whenever possible and all vehicles are reusable. The vehicles used in these figures are: 1) Lunar Transfer Vehicle (LTV), which operates between Low Earth Orbit (LEO) and Low Lunar Orbit (LLO), 2) Lunar Lander (LL), which shuttles payloads between the Lunar surface and LLO, 3) Phobos Tanker Lander (PTL), which operates between Phobos or Deimos (Ph/D) and LLO, 4) Mars Lander (ML), which launches payloads from the Mars surface into Mars orbit, 5) Mars Tanker Vehicle (MTV), which operates between Mars orbit and LLO, and 6) Expendable Launch Vehicle (ELV), which launches payloads from Earth's surface into LEO.

Figures 1a-d depict the ratio between the payload delivered to a given destination (100 mT for all cases) and the total weight of each vehicle involved, taken at the node where the vehicle is heaviest with propellant. For these figures (each indicating a destination) the propellant production sites considered are shown along the x-axis, with each vehicle involved in transporting the propellant represented by a bar. The vehicles that are labeled in bold type are of major interest in determining which scenarios are preferred in terms of performance, because they are the ones transporting propellant directly from the production sites.

LOW EARTH ORBIT

For LEO (Figure 1a) the Ph/D propellant case is favorable since it involves only one vehicle, the PTL, which carries 27% of its total weight as payload. Also favorable is the Lunar surface propellant case in which the payload delivered to LEO is 42% of the LTV weight and 30% of the Lunar Lander weight; this example assumes the availability of hydrogen on the Moon, however. A less favorable, but more realistic case utilizes Lunar oxygen with the hydrogen originating on Earth. This is more complex logistically since hydrogen must be transferred to the LL and oxygen transferred to the LTV in Low Lunar Orbit. This also results in somewhat decreased payload/vehicle weight ratios (33% of the LTV and 25% of the LL) since the LL must descend to the Lunar surface with its ascent hydrogen which increases the load that the LTV must carry. However, the Lunar oxygen scenario fares better than the situation in which all propellants are brought to LEO from Earth's surface (payload is 3% of ELV total weight). Interestingly, transporting propellant from Mars' surface to LEO (7% of the ML and 27% of the MTV) is more favorable in terms of mass than from Earth's surface to LEO.

LOW LUNAR ORBIT

For LLO (Figure 1b) the Lunar surface propellant case is very favorable since the Lunar Lander carries 59% of its total weight as payload; again, this case is constrained by the availability of Lunar hydrogen. Transporting propellant from the Lunar oxygen case is operationally similar to the case mentioned above where the

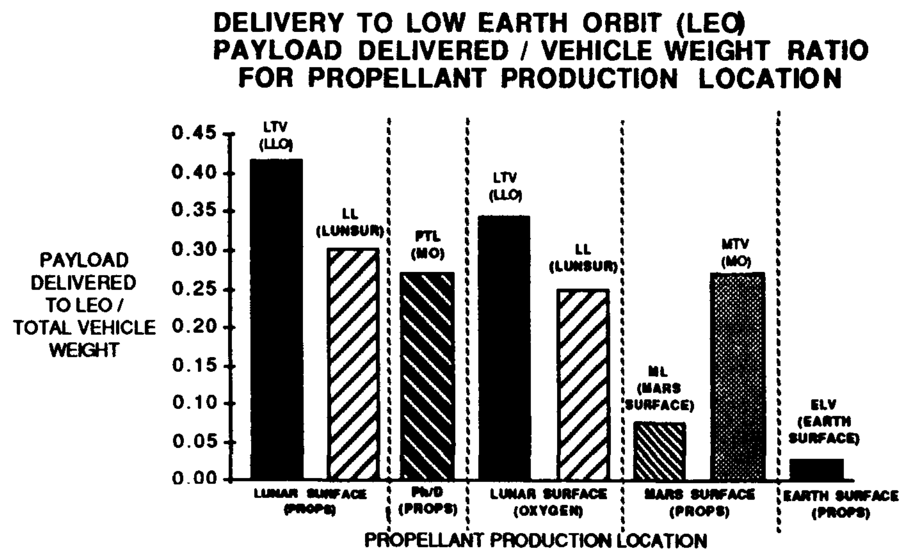

Figure 1a.

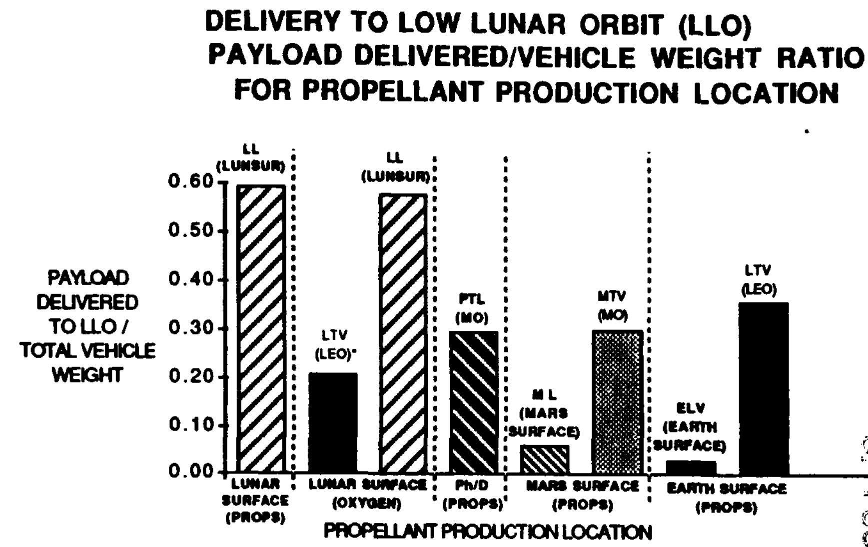

Figure 1b.

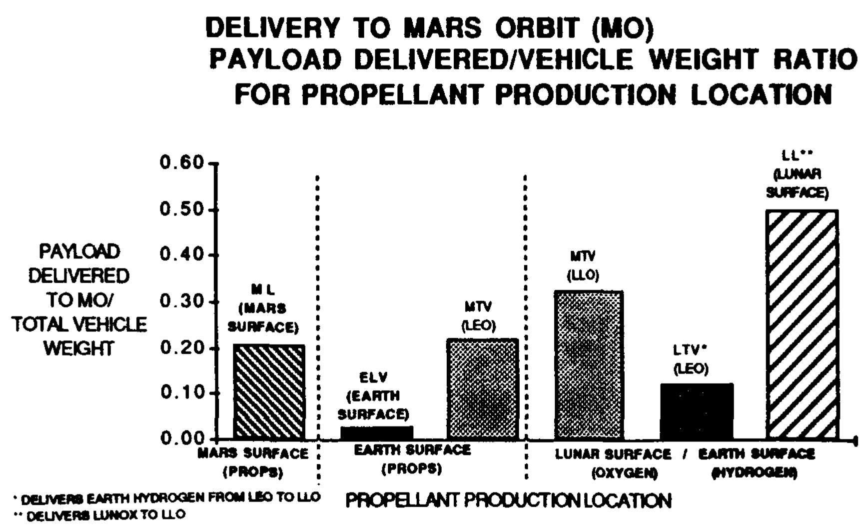

Figure 1c.

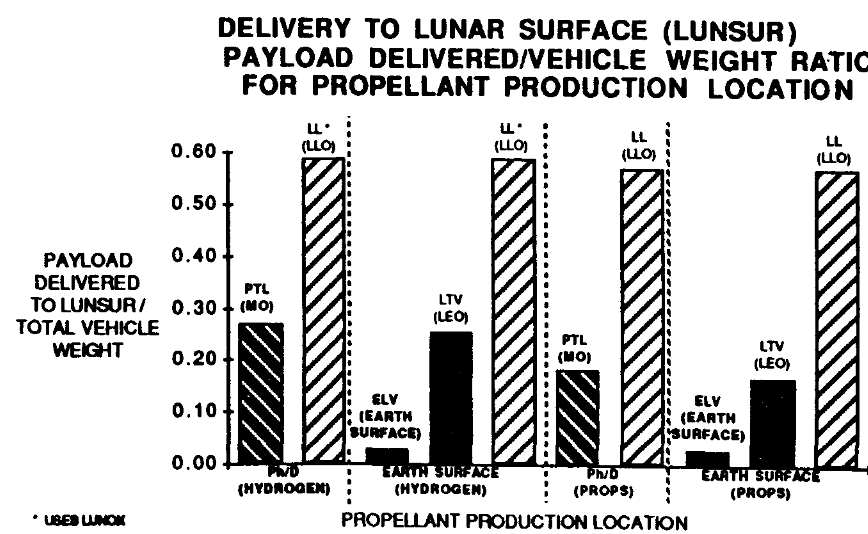

Figure 1d.

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destination for the lunox is LEO, and the payload/vehicle ratio for the LL is 58%. Ph/D is also favorable with the PTL carrying 30% of its total weight as payload. For Mars surface propellants the payload/vehicle weight ratio for the ML is 6% while for the MTV it is 30%. As with the other destinations, transport from Earth's surface is very difficult, with a ratio of 3% for the ELV and 36% for the LTV.

MARS ORBIT

For Mars orbit (Figure 1c), aside from using Ph/D as an obvious propellant source, the most favorable case uses a Mars Lander to transport propellant from Mars' surface (payload is 21% of the ML weight). Transportation of Lunar oxygen and Earth hydrogen to Mars orbit from LEO involves three vehicles. In addition to a Mars Transfer Vehicle, an LTV delivers hydrogen to LLO (payload to vehicle weight ratio is 12%) and an LL transports oxygen to LLO (ratio is 50%). As in the LEO or LLO delivery cases involving lunox and Earth H2, the logistics are somewhat more complicated because the propellants originate at different sources. With the terrestrial and Lunar propellants collected in LLO the MTV is fueled there for the flight to Mars orbit (ratio is 33%). Transporting the propellant payload from Earth's surface is operationally simpler but less efficient in propellant utilization, the ELV having a payload /vehicle ratio of 3% and the MTV having a ratio of 22%.

LUNAR SURFACE

Two propellant sources are assumed for delivery to the Lunar surface; Ph/D and the Earth (see Figure 1d). Depending on the status of the Lunar propellant production infrastructure, these sources can be assumed to supply either all the propellants or just the hydrogen component to the Lunar surface. Figure 1d indicates a strong advantage for Ph/D over Earth's surface as the propellant source, and a mild advantage for transportation of hydrogen only over all the propellant (for either source). Assuming a viable lunox production capability, this mild propellant leverage advantage for hydrogen transport only, translates to a strong operational advantage since 100 mT of hydrogen transported can be utilized as part of 800 mT of propellant. Lunox is used to power the Lunar Landers in the cases where hydrogen only is delivered to the Lunar surface, with the other vehicles using oxygen from Earth. For the Ph/D hydrogen case, the PTL has a payload /vehicle weight ratio of 28% while the Earth surface hydrogen case ratio for the ELV is 3%. The Ph/D hydrogen scenario is of particular interest because it is an example of complementary interplanetary resource retrieval; i.e. the volatile elements (believed to be) abundant on Ph/D are economically transported to the extremely volatile-deficient Lunar surface for use in Lunar base operations and industrialization.

PROPELLANT UTILIZATION

Figure 2a shows LEO as the payload destination. Propellants transported from Ph/D are most favorable with 95 mT of propellant used to transport 100 mT to LEO. If Lunar hydrogen is available (as well as lunox) this would also be favorable (215 mT used), and it is instructive to compare this to the situation where lunox is available but H2 must be transported from Earth. In this case, propellant used in the LTV and LL are increased since hydrogen must be transported to the Moon as well as from the Moon,

DELIVERY TO LOW EARTH ORBIT (LEO) PRODUCTION REQUIRED vs PRODUCTION LOCATION

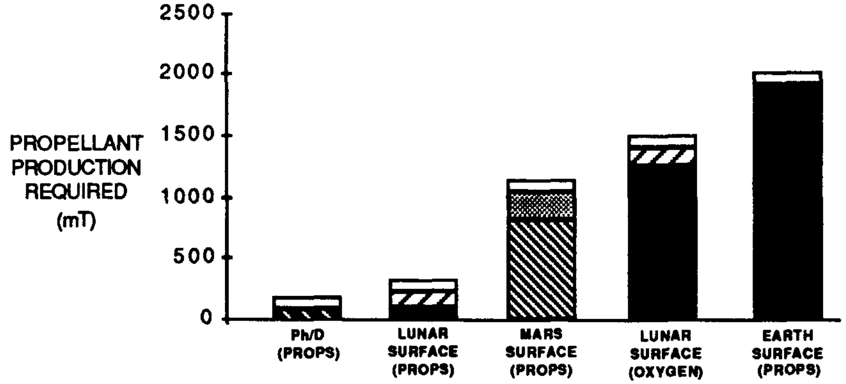


FIGURE 2a

DELIVERY TO LOW LUNAR ORBIT (LLO) PRODUCTION REQUIRED vs PRODUCTION LOCATION

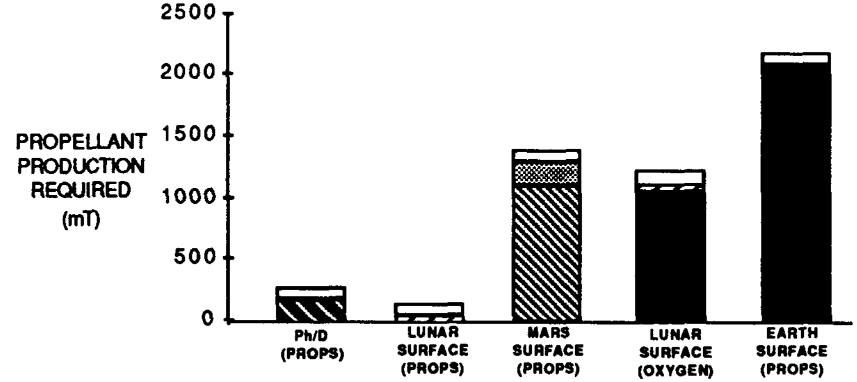
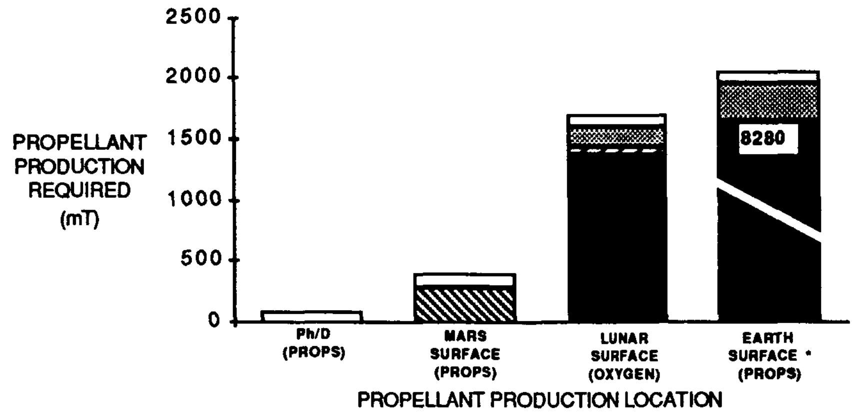


FIGURE 2b

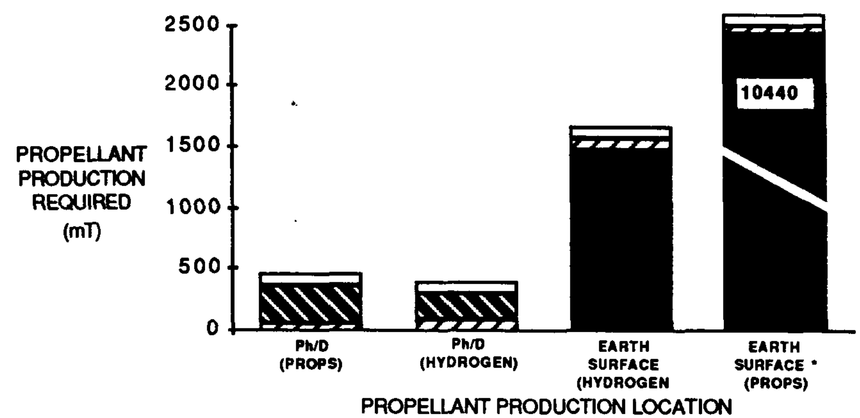
DELIVERY TO MARS ORBIT (MO) PRODUCTION REQUIRED vs PRODUCTION LOCATION



* ACTUAL CONTRIBUTION OF ELV IS 5 TIMES LARGER

FIGURE 2c

DELIVERY TO LUNAR SURFACE (LUNSUR) PRODUCTION REQUIRED vs PROPELLANT LOCATION



* ACTUAL CONTRIBUTION OF ELV IS 5 TIMES LARGER

FIGURE 2d

but the major inefficiency results from transporting hydrogen to LEO from Earth's surface (1106 mT, for a total of 1409 mT). Propellant from the surface of Mars uses 1037 mT for transport. Propellant transfer from Earth's surface (1926 mT to place 100 mT in LEO) requires the most propellants.

Figure 2b shows the Low Lunar Orbit destination. Lunar propellants (lunox & Lunar H) are most favorable (60 mT). This compares to 1133 mT if hydrogen must be transported from Earth. Propellant from Ph/D is very favorable with 187 mT used in transport. Mars surface propellants requires 1292 mT. Again, highest propellant requirement is from Earth's surface,using 2089 mT.

Figure 2c shows Mars Orbit as the destination. To transport propellant from Mars' surface uses 294 mT. The lunox/Earth H case is relatively complex, transporting hydrogen from Earth and lunox to drive all vehicles, with LLO serving as the staging point; this uses 1608 mT (1264 mT for the ELV). If all propellant comes from Earth's surface then 8588 mT are used (8280 mT for the ELV).

Figure 2d shows the Lunar Surface destination. Lunox/Ph/D H is most favorable (296 mT used). Here, lunox (& Ph/D H) is used to drive the LL with Ph/D propellant used in the PTL. With Ph/D propellant for both vehicles, use is 373 mT. Lunox/Earth H requires 1568 mT (1245 mT for the ELV). If all propellant comes from Earth's surface then 10854 mT are used (10440 mT for the ELV).

Figure 3 shows propellant production and use locations for several possible scenarios including the seven Lunar and Ph/D cases investigated at this time. The IMS ratio figure of merit is a measure of the LEO weight savings for a given amount of in situ propellant production versus a scenario in which all propellants originate on the Earth. As in all cases in this report, data are for steady-state scenarios and do not reflect space infrastructure buildups.

The largest IMS (4.3) is for the use of Ph/D propellants in Mars orbit. The negative IMS for Lunar oxygen export to LEO suggests it will be difficult for chemical propulsion systems to make this case profitable. Using Lunar oxygen in LLO (IMS = 1.8) appears to be a very good case although it has lower leverage than using Ph/D propellants near Mars. The apparent good leverage for exporting Lunar oxygen to Mars orbit is reflective of the fact that only one of the three vehicles required in Case 7 - the Lunar tanker which delivers hydrogen to LLO - ever appears in LEO. The extremely favorable leverage of Ph/D propellants in LMO and the surprisingly good performance in LEO and LLO suggests strongly that in situ propellant production on Ph/D may be the most profitable early space resource scenario.

Our current conclusions and recommendations are summarized in Figure 4. Ph/D propellant production and exportation is an attractive prospect that may support space operations near the Moon and LEO,while providing an economic incentive to explore Mars.

PLACE OF APPLICATION	Ph/D H2 + O2	LUNAR O2	MARS O2 + FUEL
LEO	#3 IMS = .4	#6 IMS = -.6	TBD
LUNAR SURF	TBD	TBD	TBD
LLO	#4 IMS = .4	#5 IMS = 1.8	TBD
E-M LIB	TBD	TBD	TBD
MARS SURF	TBD	TBD	TBD
MARS ORB	#1, #2 IMS = 4.3	#7 IMS =1.8	TBD
E-S LIB	TBD	TBD	TBD

FIGURE OF MERIT: Ideal marginal savings ratio: MT of LEO savings per MT in situ propellant production. Above results based on conjunction class LH/LOX freighters w/ aerobrakes

FIGURE 3 - ISP PROPELLANT LEVERAGE ANALYSIS RESULTS - STATUS

Where should the priority be for leveraging chemical propulsion systems to Mars?

On lunar LOX for outbound savings?
On Ph/D propellants for homeward bound savings?

Conclusions:

- Ph/D propellants offer substantially more LEO mass savings.
- More leverage from Ph/D because:
  - Ph/D is regularly "closer" to LEO than is the lunar surface, via conjunction class chemical freighters.
  - Ph/D potentially offers easily available fuels, as well as oxidizers.

Should in-situ propellants be exported to LEO?

Conclusions:

- Ph/D might be beneficially exported to LEO, given adequate excess production levels, good propellant storage, and highly automated plants.
- Lunar oxygen export to LEO provides no apparent benefit, using chemical freighters.

IN-SITU PROPELLANT LEVERAGE ANALYSIS CONCLUSIONS

- In-situ propellants offer great potential for space exploration cost savings.
- Phobos and Deimos offer by far the greatest potential leverage on all missions involving Mars.
  - Exploitation at the earliest possible opportunity is strongly indicated.
  - The Ph/D leverage will be very high even if hydrogen is not found.
- Labor intensive operations of ISPP plants will markedly degrade their advantage. Strong emphasis should be given to optimizing automation.
- Lunar oxygen can be attractive for lunar vicinity operations, returns to earth, and outbound sprints to Mars.
- Phobos/Deimos propellant production may offer a low technology alternative to multi-megawatt nuclear cargo vehicles which allows an evolutionary buildup of mass through LEO in the Lunar outpost to Mars outpost scenario.

POTENTIAL HIGH PAYOFFS

- Ph/D fuels (hydrogen or other) with lunar LOX for lunar vicinity operations
- Ph/D hydrogen for nuclear thermal rockets
- Ph/D excess oxygen is ideal for Mars vicinity life support

FIGURE 4 - CONCLUSIONS AND RECOMMENDATIONS

ISSUES OF MARS ORBITAL REFUELING

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Brue Cordell (Team Leader) - General Dynamics Space Systems Division

RENDEZVOUS/PROXIMITY OPERATIONS ISSUES

Rendezvous and proximity operations are of paramount importance for mission success of the Mars Orbital Refueling mission. The optimum method will provide maximum operational flexibility. An understanding of the Mars environment and the perturbation effects upon rendezvous and proximity operations is critical. This study investigated sample rendezvous options, the effects of perturbations, and rendezvous guidelines and operations, based upon STS experience and requirements. Figure 1 summarizes the issues identified for these operations in a Mars Orbital Refueling scenario.

EXAMPLE OF RENDEZVOUS OPTION, AND DELTA V's

- RENDEZVOUS WITH CARGO VEHICLE AT PHOBOS;
  PILOTED VEHICLE AT INITIAL 500 x 33,000 km (24 HOUR), 15 ° INCLINATION PARKING ORBIT:
- PLANE CHANGE AT APOAPSIS: 0.12 km/sec
  PHOBOS Δ V: 0.8 km/sec

MARS ORBITAL PERTURBATIONS

- MAXIMUM EFFECTS ON LOW MARS ORBITS: NODAL REGRESSION = -9.4°/DAY; APSIDAL ROTATION = 19°/DAY
- TYPICAL EFFECTS ON HIGHLY ELLIPTICAL 24 HOUR ORBITS: NODAL REGRESSION = -.25°/DAY; APSIDAL ROTATION = .49°/DAY
- ADDITIONAL SMALLER PERTURBATIONS DUE TO HIGHER ORDER TERMS IN THE GRAVITY FIGURE, INFLUENCE OF THE SUN, OTHER PLANETS, AND THE MARTIAN MOONS

SHUTTLE EXPERIENCE AND MARS ORBITAL REFUELING

- CURRENT RENDEZVOUS AND PROXIMITY OPERATION TECHNIQUES DEVELOPED PROVIDE THE BACKGROUND, OPERATIONS, AND PROCEDURES FOR A MARS ORBITAL RENDEZVOUS MISSION
- CREW SAFETY CONCERNS RECOMMENDS VISUAL CONTACT WITH FAVORABLE LIGHTING DURING PROXIMITY OPERATIONS AND BAILOUT CAPABILITY FOR CRITICAL FLIGHT PHASES

Figure 1: RENDEZVOUS AND PROXIMITY OPERATIONS

The following is an example of the rendezvous and ΔV's for the Piloted Vehicle (PV) to rendezvous with the Cargo Vehicle (CV), which has landed ("rendezvoused") on the surface of Phobos. The PV is assumed to be initially in a Mars orbit of 500 km X 300,000 km (24 hour orbit), 15° inclination parking orbit. The initial plane change maneuver occurs at apoapsis, where a 0.12 km/sec impulse is applied. A final ΔV of 0.8 km/sec, to rendezvous with Phobos, occurs at the apoapsis of this transfer orbit.

This study investigated a few parking orbit options, and determined perturbations affecting them. Low Mars circular orbits (e.g. 500 km x 500 km), have nodal regressions up to -9.4°/day and apsidal rotations up to 19°/day. A highly elliptical Mars equatorial orbit of 500 km x 33,000 km, has a nodal regression of -.25°/day with an apsidal rotation of 0.49°/day. Higher order terms were not included in this analysis, but have large perturbation effects during lengthy on-orbit times. These include higher order harmonics, solar influence, and Martian moon effects.

Rendezvous and proximity operations can be derived from Gemini, Apollo, and STS experience. To date, six Shuttle missions have performed deployment and retrieval of passive, free flyer vehicles. Future planning includes extensive operations with the Space Station. Based on this, current rendezvous and proximity operation techniques developed for the STS provide the necessary background, operational constraints, and procedures for a Mars Orbital Refueling mission.

Several areas were identified for further requirement definition of this phase of the mission. Crew safety should be considered foremost in the consideration of any Mars scenario. This dictates requirements for procedure design/implementation ease, position determination/target tracking methods, and visibility, propellant consumption, and plume impingement requirements.

In addition, various issues were identified. They include the following: vehicle design, cargo vehicle orbital prediction uncertainties, Cargo Vehicle maneuverability, docking methodology, and Remote Manipulator System (RMS) capabilities.

OPERATIONS ON PHOBOS/DEIMOS

Several issues pertaining to operations on Phobos and Deimos are described in this section. Though for some issues no distinction between operations on Phobos and Deimos is significant, wherever distinctions can be made they will be spelled out. These issues are summarized in Figure 2.

ISSUE	COMMENTS
RENDEZVOUS	• CV LOCATION ASSURED IF EARLY RENDEZVOUS with Ph/D • EARLY CV-Ph/D RENDEZVOUS MUST BE AUTOMATED • BETTER ACCESS TO DEIMOS
PROXIMITY OPERATIONS/DOCKING	• USE MILLI-G FIELD OR ESTABLISH "HANDLES" (e.g. with PENETRATORS) • EFFECTS OF IRREGULAR SURFACE AND DUST ENVIRONMENT • PHOBOS' HIGHLY VARIABLE G-FIELD vs DEIMOS MORE UNIFORM FIELD
CONTINGENCIES	• OPERATING ON AND/OR NEAR Ph/D GIVES MAXIMUM FLEXIBILITY IN OPERATIONS SEQUENCE AND POTENTIAL EMERGENCY OPERATIONS
CRYOGENICS	• HIGH MARS ORBIT COOLER THAN LOW MARS ORBIT • ALTHOUGH Ph/D ARE WARM, OFFER OPPORTUNITY FOR SHADING FROM SOLAR AND MARS RADIATIONS
PROPELLANT TRANSFER	• MILLI-G FIELD MAY PROVIDE BACKUP MODE FOR ZERO-G FLUID ACQUISITION/TRANSFER
METEOROID ENVIRONMENT	• POTENTIALLY MORE HAZARDOUS NEAR PHOBOS

Figure 2: OPERATIONS ON PHOBOS/DEIMOS

RENDEZVOUS

A prime advantage to storing the CV on Ph/D rather than in Mars orbit is that the CV's location is thereby assured, the moons being much easier objects for the PV to track than the small CV, especially if the CV beacon or transponder system were to become inoperative. Even if the CV's location on Ph/D is unknown, it would be much easier to find than if it were lost in free space. For the CV's part, an automated rendevous with Ph/D is required, which is considerably more complex than establishing an orbit in free space around Mars. The initial mission would have to rely on optical sightings of Ph/D and perhaps radar ranging as it got closer. For later missions this could be supplemented by a transponder or beacon emplanted on the surface of Ph/D. Access to Deimos would be easier since it is higher in Mars' gravity well and maneuvers would require less delta v than for Phobos.

PROXIMITY OPERATIONS AND DOCKING

Since the gravity field of Ph/D is non-negligible for objects that are nearby and have low relative velocities, the effect is to considerably complicate the calculation of relative motion for proximity operations. A limited three body problem treatment (CV or PV being third body with negligible mass) is required. In addition, the gravity fields of the moons, and in particular Phobos, are highly non-spherical which would affect relative motion in ways that are currently difficult to predict in detail. Thus proximity operations involving the unmanned CV may pose control technology challenges since real-time human-guided control of the vehicle will not be available in the scenario where the CV docks with Ph/D for storage before the PV is sent.

For the final "docking" with Ph/D a couple options are possible. After nullifying the relative velocity at a short distance the vehicle can be allowed to settle to the surface under the influence of Ph/D gravity. This operation is very simple but if the underlying surface is sloping or highly irregular there is a possibility that the unsecured vehicle can slide downhill and hit an object that would damage it, or settle into a position that is unstable or highly tilted. An alternative would be for the vehicle to establish "handles" to grip the Ph/D surface by firing penetrators into the surface and then reel itself into a secure docking by cable. A problem with this approach is that the surface may be very friable. Unless the penetrators bury themselves quite deeply they are likely to shatter, fracture, and break free of the regolith under the loading of a several hundred thousand pound vehicle. Of course this can be avoided if the load applied to the cables is limited, but then the ability to control the motion of the vehicle as it approaches the surface would also be limited. Perhaps penetrators can be useful if they are used mainly to anchor the vehicle as it settles to the surface rather than as a means of reeling in and controlling the motion of the vehicle.

CONTINGENCIES

Operating on or near Ph/D offers a backup contingency for fuel transfer which isn't available in low Mars orbit. Though zero-g transfer of fuel between the CV and PV may be the baseline method, the milli-g gravity of Ph/D might be used for propellant settling and/or siphoning in case of hardware failure.

CRYOGENICS

The issue of thermal environment is very important in determining how much and what type of insulation is needed by the CV to minimize boiloff of its cryogenic fuel load during its long period of storage in Mars orbit. Direct radiation from the Martian surface is a significant source of thermal energy and so a high Mars orbit is a cooler environment (and thus a better storage environment) than a low Mars orbit. Accordingly, storage in an orbit at the distance of Deimos will provide a cooler environment than that for Phobos. Both Phobos and Deimos have synchronous orbits (i.e., rotation rates that are tidally locked to their periods, as is the Earth's moon). Thus a CV stored on the farside of Ph/D would be shielded from the radiation load of the Martian surface. However, the Ph/D surfaces are in themselves relatively warm radiators (though cooler than the Martian surface), and heat conduction to the CV by contact with the Ph/D surface would also have to be minimized. Placing the CV anywhere on Ph/D, except perhaps the poles, would shield the vehicle for half its period from direct solar radiation, which is an advantage over storage in a free space orbit. In summary, storage of the CV at a sub-polar latitude on the side of Ph/D facing away from Mars would provide a fairly benign thermal environment as long as radiation and conduction from the Ph/D surface itself can be minimized adequately.

PROPELLANT TRANSFER

Issues pertaining to propellant transfer on Ph/D were examined in the section on Refueling Operations. For the option of Refill Tanks the milli-g environment of Ph/D could offer advantages for the tanking operation by allowing propellant settling. The Tank Change-out and Crew Module Transfer options may be workable on Ph/D, though the milli-g environment and proximity of the hard, abrasive regolith could introduce risk factors that make manipulation of large tanks or modules undesirable there. For the Redundant Vehicles option there should be no particular obstacle to operation on Ph/D since all that is involved is transfer of the crew and some equipment from one vehicle to the other.

METEOROID ENVIRONMENT

The escape velocities of Phobos and Deimos being very small, there could be a belt of material that has been ejected from the moons by impacts, concentrated along the orbit of each moon and having fairly low relative velocities. This would be especially likely for Phobos which orbits near the Roche limit for Mars so that material ejected from it would have a lower probability of reimpacting and being reabsorbed by the parent body within a given period of time than for Deimos. In other words, the dust belts, if they exist, would likely be denser and more stable in the orbit of Phobos than for Deimos. These belts could affect rendevous and proximity operations by causing one to choose approach paths that either avoid the dust belt or pass through it as quickly as possible to minimize the chance for a collision. It may also be desirable to choose a vehicle orientation that minimizes the seriousness of damage if a collision were to occur. Extra shielding may need to be employed for parts of the CV since it will be stored in this environment for a long time. If mining for propellants commences to almost any degree, then a dust belt is likely to be created, even if there wasn't a significant one before. Methods for containment of most ejecta may need to be developed in conjunction with methods for blasting or drilling on Ph/D.

MARS ORBIT REFUELING - OPERATIONS METHODS

The sequence for a Mars Orbit Refueling scenario depends upon the location at which the propellant transfer operations are accomplished. This study considered four locations; Low Mars Orbit (LMO), High Mars Orbit (HMO), and on the surface of a Martian moon: Phobos or Deimos (Ph/D). Each requires distinctively different methods to transfer propellants, resulting largely from the gravity fields encountered. In both cases, it is assumed that the Cargo Vehicle (CV) has been launched and is either in Mars orbit (LMO or HMO), or on the surface of one of the moons. The Piloted Vehicle (PV) must locate the CV, rendezvous, dock, and perform the propellant transfer. Figure 3 summarizes the operations for each location.

LOCATION	GRAVITY	OPERATIONS SEQUENCE
LMO, HMO	Zero G	Dock PV with CV/Descend/Transfer/Remove CV/To Destination
On Ph/D	Milli-G	Dock CV with Ph/D (Auto)/Descend-Dock PV to CV/Invert(?)/Transfer/Seperate/ To Destination
		Dock CV with Ph/D (Real Time or Remote)/Descend-Dock PV to CV/Invert(?)/Transfer/ Separate/To Destination
		Dock PV with Ph/D /Descend-Dock CV to PV/Transfer/Remove CV/To Destination

Figure 3: MARS ORBIT REFUELING - OPERATIONS METHODS

MARS ORBIT

In both LMO and HMO, operations are conducted in a zero-G environment. The PV rendezvous with the CV and the two vehicles dock on-orbit. The mated configuration descends, and the propellants are transferred from the CV to the PV by directly pumping the fluids or transferring vehicle components (i.e. propellant tanks). The CV is undocked, and the PV continues it's mission.

PHOBOS or DEIMOS

If the rendezvous with the CV is to take place on Ph/D, operations are conducted in a milli-G environment and three scenarios were identified to transfer propellants.

1) The CV "lands" (i.e. rendezvous, docks, anchors) on Ph/D via autonomous command and control. Upon arrival at Ph/D, the PV would descend to the surface and dock with the CV. Depending upon the orientation that the CV landed on the moons surface, the mated configuration may have to invert itself to attain the proper attitude for propellant transfer. Propellants are transferred by directly pumping or transferring vehicle components (i.e. propellant tanks). The PV separates and continues it's mission.

2) The CV "lands" on Ph/D as before, only this scenario utilizes real-time or remote control. Final PV docking, propellant transfer and continued operations are identical to 1).

3) The CV "lands" on Ph/D as before. The PV would descend to the surface and dock with the CV. Propellants are transferred by directly pumping, or transferring vehicle components (i.e. propellant tanks). The PV separates, continues it's mission, then returns to Earth.

REFUELING OPERATIONS

Several options have been identified for the transfer of fuel for the return trip from Mars to Earth. These form a near continuum of options ranging from transfer of the fuel to transfer of the crew, and are shown diagrammatically in Figure 4. The options are described as follows:

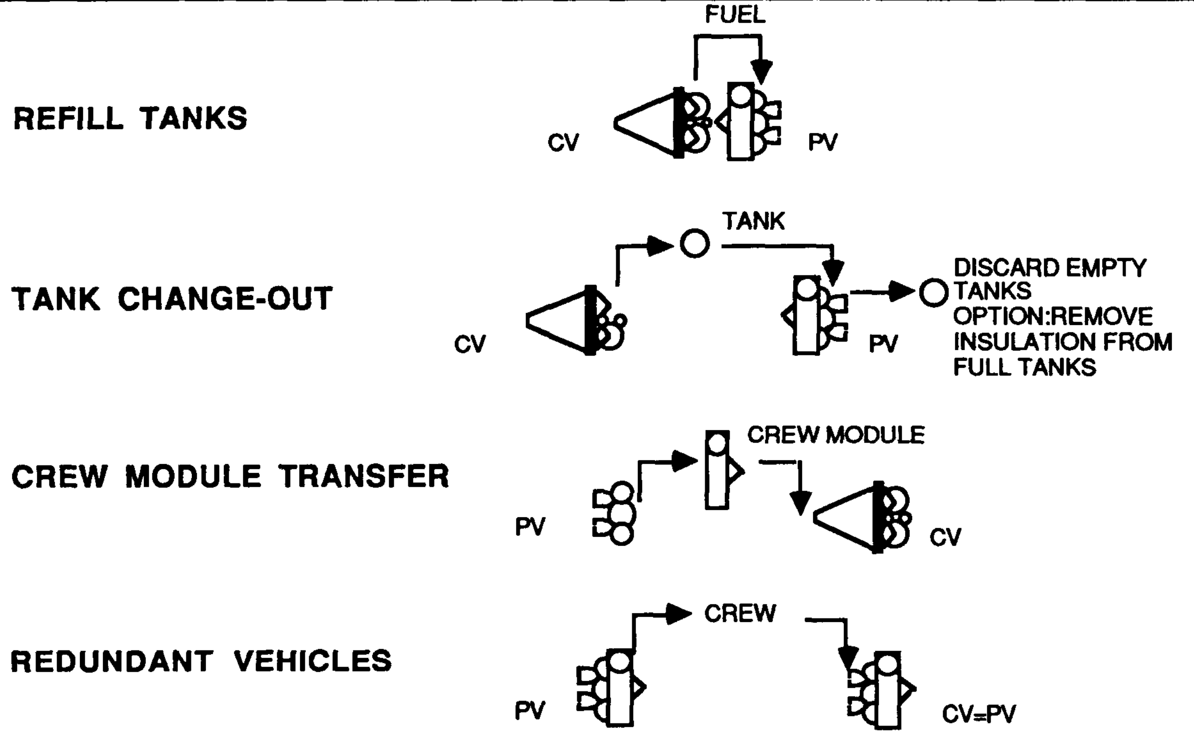


Figure 4: REFUELING OPERATIONS - CONTINUUM OF OPTIONS

REFILL TANKS

Hard docking between the PV and the CV as well as maintenance of near inertial attitude will be necessary to ensure fluid transfer with a minimum of slosh. The time required for transfer of several hundred thousand pounds of propellant will vary depending on the power available, as indicated in the Propellant Transfer Operations chart. The number and size of fluid hose connections also affects the rate at which fluid transfer is possible, as well as the Ops complexity. The milli-g environment of Phobos/Deimos could offer advantages for the tanking operation by allowing propellant settling. However, even if propellant transfer while on a moon becomes the baseline, the system should be capable of transfer in a zero-g environment. Since the final resting position of an unmanned cargo vehicle on a moon could be highly tilted with respect to the local vertical, allowance for this should be made in a system designed for milli-g operation. Though it is a relatively simple operation requiring the attention of perhaps not more than one crewmember, refilling could nonetheless occupy a large fraction of the total stay time in Mars orbit. For Phobos/Deimos missions the effect of this would be minimized if refilling took place on the target moon, thereby allowing the majority of the crew to be engaged in productive exploration and research while it took place. The importance of refueling may dictate that it be successfully completed before commiting the vehicles and crew to any other significant activity. For missions involving Mars landing and/or refueling in zero-g, the time required for refilling tanks could thus greatly affect the productive time spent in Mars orbit.

TANK CHANGE-OUT

An alternative to transferring the contents of tanks from one to another is to change-out the tanks themselves. Such a maneuver may not require hard docking between the vehicles but may require that the operation be performed in zero-g rather than on a moon, for safety. Though avoiding issues involved in fluid transfer and requiring a minimal expenditure of energy, this operation would be relatively complex, involving at least two interface break and makes per tank. It would probably require less total time than fluid transfer but would be much more manpower intensive during the operation. From a development and manufacturing standpoint it would be desirable to make identical the tanks on both the PV (which are disposed of in Mars orbit) and the CV (which get transferred to the PV). From a performance standpoint the fuel tanks on the PV, having a shorter storage time and more benign thermal environment than those on the CV, require less insulation and could thus be made lighter (with corresponding performance gains) than those on the CV. A possible compromise would be to make at least part of the insulating system for the CV tanks from a material that could be easily removed in Mars orbit, such as MLI blankets, and thus have a weight savings for the return leg. The underlying tank could then be identical for both PV and CV. Even if this removable insulation isnt quite as efficient as a system whereby the insulation is entirely integral to the tank structure (thus resulting in a heavier tank), the quality of removability in Mars orbit still offers the potential for a net reduction in mass transported to LEO.

CREW MODULE TRANSFER

This option bypasses the issue of how to get fuel from the CV to the PV by transferring instead the crew module to the CV while in Mars orbit. As with tank change-out, a zero-g environment may be required for safety. The power required is minimal, and the operation relatively simple, possibly involving a single crew module interface. This simplicity also allows a minimal amount of time to be spent on the operation. A negative aspect is that the crew module would be transferred to a relatively unknown vehicle for the return trip, the CV having been stored unattended for a long period of time in Mars orbit or on Ph/D. Besides being a viable primary method, crew module transfer offers a useful role as a backup method in the event that the PV propulsion system becomes damaged or disabled after entering Mars orbit.

REDUNDANT VEHICLES

This option is the simplest of any listed here since it involves no manipulation of hardware or fuel, but only the transfer of the crew and their equipment from the PV to the CV. As such it is the least demanding in time and energy, and could be accomplished either in zero-g or on Ph/D. Disadvantages include the fact of transferring to a vehicle in which the crew module as well as the propulsion system are relatively unknown. Another negative aspect is the necessity of hauling an extra crew module to Mars orbit. Though this option might be justified as a primary transfer method, the performance penalty incurred by bringing a second crew module to Mars orbit makes this an unlikely backup option. In the case where the crew module is made up of several smaller connected modules, such as space station modules, an additional back-up option exists. The CV could ferry to Mars orbit a single crew module unit which would be used in the event that one of the PV crew module units was damaged during the trip from Earth. This module could be loaded for the trip out with supplies and equipment needed in Mars orbit. In the event that this replacement option is not needed, the extra module would be detached from the CV if Crew Module Transfer is being used, or can remain attached to the CV for the other options. For any of the options the unused module could be available as a component for a future Ph/D outpost. The probability of failure of more than one crew module unit in a crew habitat consisting of, say, four units is quite small if arising from events such as meteorite puncture. Thus the reliability of the system can be increased significantly by providing replacement for a single crew module unit, which is a small fraction of the entire crew habitat.

PROPELLANT TRANSFER OPERATIONS

Three options for transferring propulsion capability to the crew module for return to Earth from Mars orbit were considered in this preliminary evaluation. These options include: Fluid (propellant) transfer from the cargo vehicle to the piloted vehicle (Figure 5), individual resupply tank transfer to the piloted vehicle (Figure 6), and crew module transfer from the piloted vehicle to the cargo vehicle (to accomplish a vehicle transfer for return operations; Figure 9).

TASK	*TIME (HOURS) 20 40 60 80 100	TOTAL MHRS	IVA	EVA
RENDEZVOUS & DOCKING/CARGO VEHICLE STATUS MONITORING	4:20	8:40	8:40 (2)	
SCHEDULED MAINTENANCE ACTIVITIES	8:00	16:00	16:00 (2)	
UNSCHEDULED – CONTINGENCY MAINT • REMOVE & REPLACE 2 ENGINES	12:00	24:00	24:00 (2)	
• REMOVE & REPLACE AVIONICS BOXES	4:00	8:00	8:00 (2)	
TANK CONDITIONING PROPELLANT TRANSFER (500K PROP)	5:00 (40:00)** (EXTRAPOLATED FROM 50K PROPELLANT FOR SBOTV)	7:00 (42:00)	7:00 (42:00) (1-2)	
DOCK CV W/PH/D SECURE	1:00	2:00	2:00 (2)	
PRELAUNCH	4:00	16:00	16:00 (4)	
LAUNCH	0:30	2:00	2:00 (4)	
TOTAL	HRS 38:50 (73:30)	83:40 (118:40)	83:40 (118:40)	0:00

( ) NUMBER OF CREWMAN *MAJOR TASKS DON'T HAVE TO BE PERFORMED ON CONSECUTIVE DAYS
** FUEL TRANSFER TIME DEPENDENT ON POWER AVAILABLE

Figure 5: CHECKOUT/MAINTAIN/REFUEL MANNED VEHICLE FOR EARTH RETURN (PROPELLANT TRANSFER)

The estimated times associated with performing each of the proposed operations are provided in the following figures, along with any additional electrical power requirements. The fluid transfer option was identified as the only one with substantial power requirements. This power is required to operate fluid pumps to maintain flow rates and heaters to provide sufficient ullage pressures. The variations in power requirements to support the forty and five hour transfer times is due to pump efficiency ranges. We used efficiencies of 0.2 and 0.6 to calculate the power requirements.

TASK	*TIME (HOURS)	TOTAL MHRS	IVA	EVA
	20 40 60 80 100			
RENDEZVOUS & DOCKING/CARGO VEHICLE STATUS MONITORING	4:20	8:40	8:40 (2)	
SCHEDULED MAINTENANCE ACTIVITIES	8:00	16:00	16:00 (2)	
UNSCHEDULED – CONTINGENCY MAINT • REMOVE & REPLACE 2 ENGINES	12:00	24:00	24:00 (2)	
• REMOVE & REPLACE AVIONICS BOXES	4:00	8:00	8:00 (2)	
DOCK CV W /PH/D SECURE	1:30	3:00	3:00	
TANK CONDITIONING PROPELLANT TRANSFER (500K PROP)	14:25	28:50	28:50	
PRELAUNCH	4:00	16:00	16:00 (4)	
LAUNCH	0:30	2:00	2:00 (4)	
TOTAL	HRS 48:45 →	106:30	106:30	0:00

( ) NUMBER OF CREWMAN *MAJOR TASKS DON'T HAVE TO BE PERFORMED ON CONSECUTIVE DAYS
** FUEL TRANSFER TIME DEPENDENT ON POWER AVAILABLE

Figure 6: CHECKOUT/MAINTENANCE/REFUEL MANNED VEHICLE FOR EARTH RETURN (TANK CHANGE-OUT)

Operational complexities were also considered. The fluid transfer and crew module options can be accommodated with single interface designs, however, the tank transfer method is complicated with handling requirements and additional interface connections. It is necessary to break and make four primary (resupply tank) and four secondary (empty tank) interfaces during operation. We assumed that all components (empty tanks, abandoned vehicle, etc.) should be left intact and anchored to either Phobos or Deimos, rather than individual dispersion.

Preliminary conclusions indicate that operations can be accomplished either in Mars orbit or on the surface of Phobos or Deimos and that all three options for propulsion system/subsystem transfer should be made available to the Mars crew.

TANK EXCHANGE

Figure 7 presents the tasks required to perform the tank exchange that was described on Figure 6. The task time is given for each task. The total operation requires 15 hours 55 minutes. The times are for tasks performed by teleoperations. The tanks will require quick releases and modularity.

The difference between step 2 and step 7, both tank removal tasks, accounts for different distances that the RMS may have to move for the beginning task.

	TASK HOURS		TASK HOURS
1. SECURE MATED PV & CV TO PH/D SURFACE	1:30	8. TRANSPORT AND INSTALL PV EMPTY TANK #2 ON CV	:45
2. REMOVE PV EMPTY TANK #1 WITH 1ST RMS	:50	9. CHECKOUT RESUPPLY TANK #1 ON PV	:15
3. POSITION PV EMPTY TANK & RMS AWAY FROM VEHICLE	:10	10. REPEAT STEPS 5 THRU 9 FOR RESUPPLY TANK #2	3:25
4. POSITION RMS #2 AT CV	:15	11. REPEAT STEPS 5 THRU 9 FOR RESUPPLY TANK #3	3:25
5. REMOVE CV RESUPLY TANK #1 WITH 2ND RMS	:40	12. REPEAT STEPS 5, 6 & 9 FOR RESUPPLY TANK #4	2:00
6. TRANSPORT AND INSTALL RESUPPLY TANK #1 ON PV	1:05	13. INSTALL PV EMPTY TANK #1 ON CV WITH RMS #1	:45
7. REMOVE PV EMPTY TANK #2 WITH 2ND RMS	:40	14. STOW AND SECURE RMS #1 & 2	:10
		TOTAL	15:55

Figure 7: TANK CHANGEOUT

TANK EXCHANGE SEQUENCE DIAGRAM

Figure 8 illustrates the steps for exchanging the empty propellant tanks on the piloted vehicle (PV) with the full resupply tanks on the cargo vehicle (CV). The PV (including the crew module) is mated to the CV. This mating can be performed either in space or on Ph/D surface. Exchange of the tanks can also be performed in space or on Ph/D surface. For this study, the vehicle were mated in space and then secured to Ph/D surface where the tanks were exchanged.

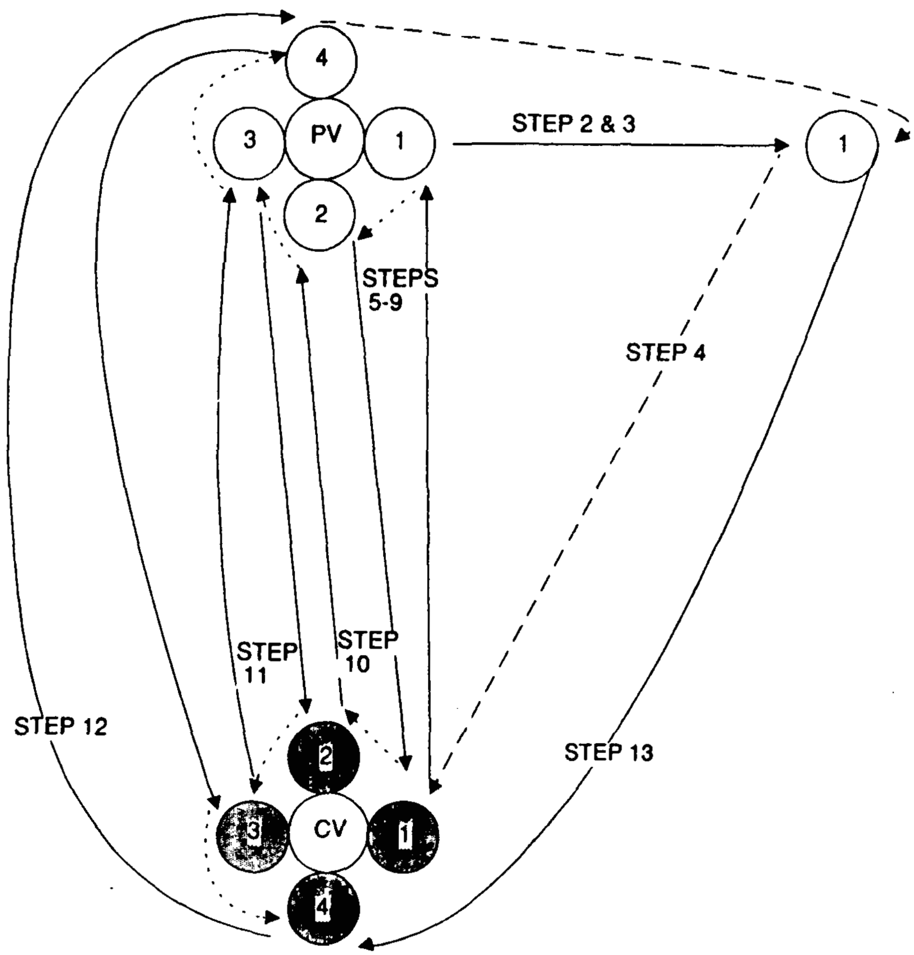


Figure 8: TANK CHANGE-OUT SEQUENCE DIAGRAM

The tank exchange includes removing an empty tank on the PV and replacing it with a full tank from the CV. The empty tank is then installed on the CV for storage. The first tank to be exchanged is removed from the PV and placed in a holding fixture by the RMS. Full tank #1 is then installed on the PV where the first tank was removed. The RMS then moves to empty tank #2, where it is grasped, released, and then transported to the CV where it is installed in position #1. The RMS now moves to full tank #2, where it is grasped, released, and transported to and installed in PV position #2. This procedure is repeated for tanks #3 and #4.

Last empty tank #1 is installed on the CV. The tanks have all been exchanged and checked out. The vehicles are now ready for separation. If the tank exchange operation was performed on Ph/D, then the CV with the tanks can be left on the surface. All empty tanks were installed on to the CV but could have been placed directly on Ph/D surface or left to float in space. It was felt that the tanks should be secured.

CREW MODULE TRANSFER

One method for refueling the Mars vehicle for the return flight to Earth is to transfer the CM from the empty PV to the CV as shown in Figures 9 through 11. The scenario for this method described in this chart requires two complete vehicle interfaces on the CM. This includes all fluid and electrical connections. This method keeps the PV and CM mated until the CV is mated and its system checkout is performed. This would eliminate a remating of the PV and CM if a contingency propellant transfer method was required.

If there were only one CM/vehicle interface, then the CM would require its own power and communications source for use during the transfer operation. The CM would be released from the PV and the PV would move out of the area. The CV would then mate to the CM. To mate the CM/CV, the RMS on the CM would be used or the CM would require its own RCS for docking. This last scenario could leave the CM vulnerable.

TASK	*TIME (HOURS) 20 40 60 80 100	TOTAL MHRS	IVA	EVA
RENDEZVOUS & DOCKING/CARGO VEHICLE STATUS MONITORING	4:20	8:40	8:40 (2)	
SCHEDULED MAINTENANCE ACTIVITIES	8:00	16:00	16:00 (2)	
UNSCHEDULED – CONTINGENCY MAINT • REMOVE & REPLACE 2 ENGINES	12:00	24:00	24:00 (2)	
• REMOVE & REPLACE AVIONICS BOXES	4:00	8:00	8:00 (2)	
TRANSFER CREW MODULE TO CV	2:50	5:40	5:40 (2)	
DOCK CV W/PH/D SECURE	1:00	2:00	2:00 (2)	
PRELAUNCH	4:00	16:00	16:00 (4)	
LAUNCH	0:30	2:00	2:00 (4)	
TOTAL	HRS 36:40 →|	82:20	82:20	0:00

( ) NUMBER OF CREWMAN *MAJOR TASKS DON'T HAVE TO BE PERFORMED ON CONSECUTIVE DAYS
** FUEL TRANSFER TIME DEPENDENT ON POWER AVAILABLE

Figure 9: CHECKOUT/MAINTAIN/REFUEL MANNED VEHICLE FOR EARTH RETURN (CREW MODULE TRANSFER)

		TASK HOURS
1.	PV MATED TO CV	•
2.	CHECKOUT ALL SYSTEMS ON CV	2:00
3.	RELEASE PV FROM CREW MODULE	:05
4.	MANEUVER PV AWAY FROM CV/CREW MODULE	:15
5.	SHORT BURST CARGO VEHICLE ENGINES	:30
6.	DOCK PV WITH PH/D AND SECURE	1:00
	TOTAL	3:50

Figure 10: CREW MODULE TRANSFER

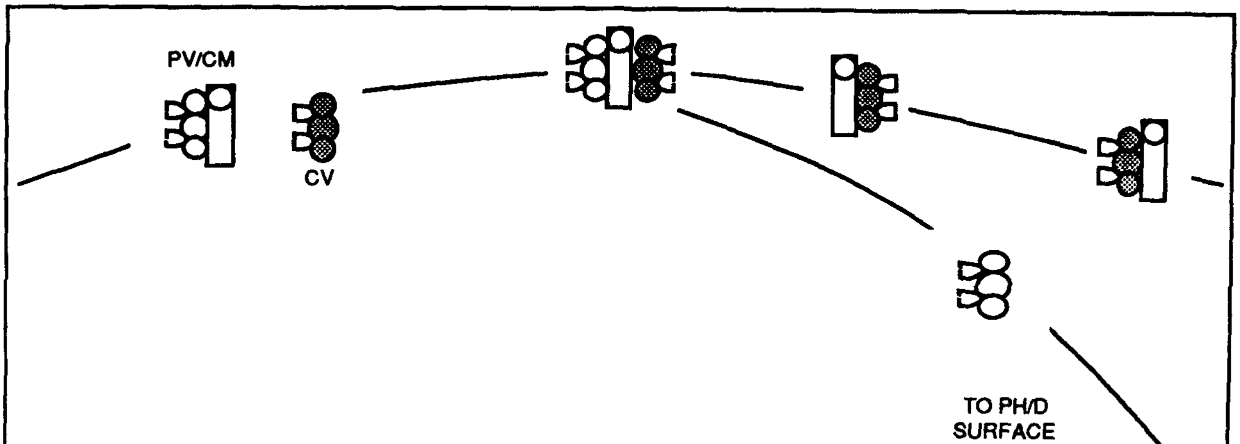


Figure 11: CREW MODULE TRANSFER SEQUENCE DIAGRAM

MAINTENANCE PHILOSOPHY

Our maintenance philosophy and expected needs is extrapolated from our studies and experience with upper stages and cryogenic storage facilities in Earth orbits; it includes the following:

1) Automated health monitoring, operational flight instrumentation, built-in test (bit), isolate faults to orbital replacement unit (ORU),

2) Fail operational/fail safe (FO/FS) units,

3) Remove/replace capability,

4) Some repair capability aboard crew module spacecraft,

5) Stock spare parts based on reliability and criticality,

6) Modular construction,

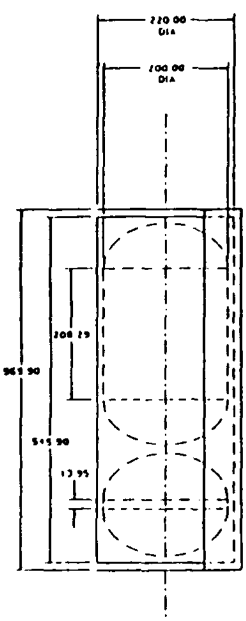
7) Minimize EVA vehicle maintenance operations using teleoperations to remove and replace units, TV inspection plus bit/health monitoring, and EVA back-up for all teleoperations (support equipment is required),

8) Maintenance records/predictions/procedures stored on board in computer - can get some help from ground on maintenance problems, what to do,

9) Have two RMS' aboard manned vehicle for maintenance, and,

10) For Mars, the question of which vehicle is suitable for Earth return should be determined first, and then the propellant transfer accomplished.

A propellant storage tank concept and weight summary is shown in Figure 12.



160,000 LB
FLUID CAPACITY

PROPELLANT STORAGE TANK EARTH TO LEO
WEIGHT SUMMARY
– HEAVY LIFT LAUNCH VEHICLE

	WT (LBS)
STRUCTURES	30237
ACTIVE THERMAL CONTROL	0
PASSIVE THERMAL CONTROL	5954
FLUID SYSTEM	2839
DATA MANAGEMENT	205
ELECTRICAL	200
ORBITAL FACILITY DRY WEIGHT	39435
PAYLOAD ADAPTER WEIGHT	700
FLUID WEIGHT	160000
RESIDUAL WEIGHT	5264
UPPER STAGE WEIGHT	21744
GROUND SUPPORT INTERFACE	300
TOTAL SYSTEM WEIGHT	227443

Figure 12: PROPELLANT STORAGE TANK CONCEPT & WEIGHT SUMMARY (REFERENCE LONG TERM CRYOGENIC STORAGE FACILITY STUDY)

DESIGN AND DEVELOPMENT SCHEDULE FOR MARS STV

A Mars STV program of this magnitude will utilize all resources available to ensure safety and mission success. It will require an operational Shuttle and/or Expendable Launch Vehicle (ELV) program, a timely completion of the Space Station, numerous technology demonstrations, and a precursor Phobos and/or Deimos imaging mission. It will be a very aggressive and intensive schedule, culminating in an Initial Operational Capability (IOC) in the first quarter of 2001. Figure 13 depicts the schedule for a Mars STV program, including the precursor programs and requirements.

The Space Station would be an integral part of any Mars scenario, providing a Low Earth Orbit (LEO) platform upon which to conduct the numerous technology demonstrations required for this mission. The Space Station program is currently in Phase C/D, with a goal of the first launch of hardware occurring in the first quarter of 1994. It should become operational (Phase 1) at the start of 1996, and would be able to support any Mars development necessary after that. In addition to its research function, the Space Station would act as the construction platform the for Mars STV, the LEO node of departure for the actual mission, and the point of return upon mission completion.

The Mars STV Program itself would consist of three phases. The first two phases would have 2 year durations each, with Phase C/D commencing in the first quarter of 1996, culminating in an IOC in the first quarter of 2001. The mission would result in the first crew arriving in the fourth quarter of 2001. A precursor to an actual landing upon a Martian moon would be the requirement for an imaging mission of the moons themselves. This would occur in the 1996 time-frame, which would allow sufficient time to analyze the data and determine a suitable landing location.

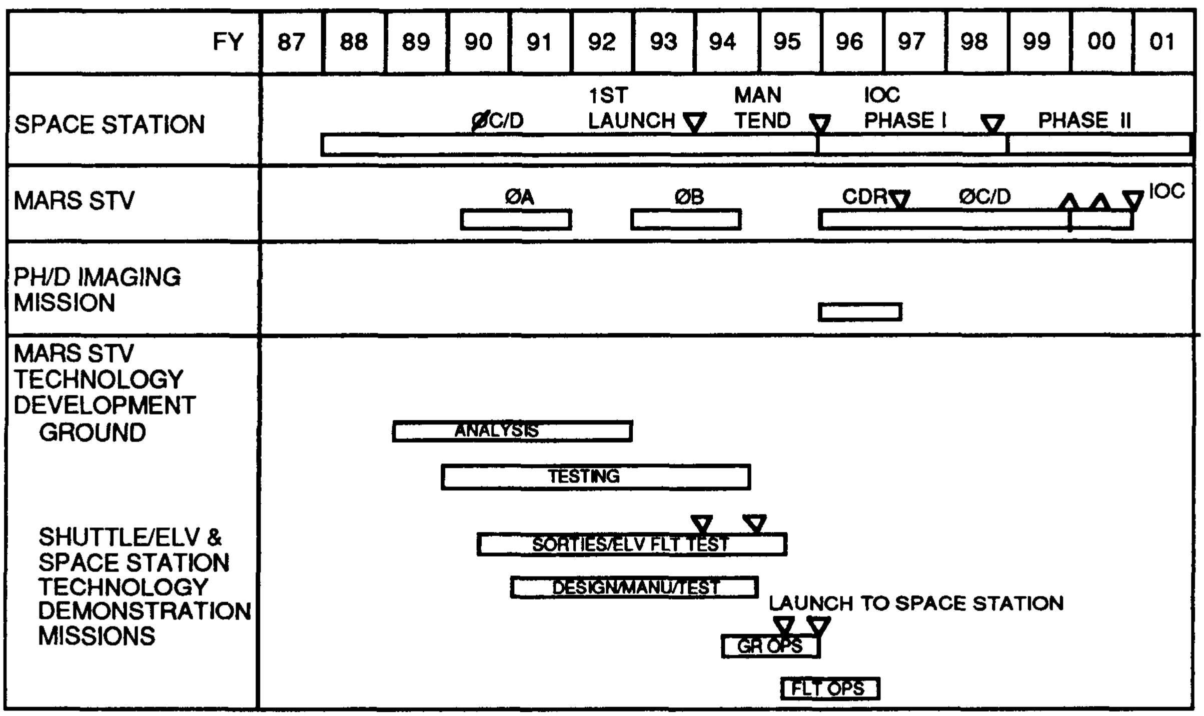


Figure 13: DESIGN AND DEVELOPMENT SCHEDULE FOR MARS STV

A thorough pre-mission analysis must be accomplished to define and solve all aspects of a Mars mission, and to develop the necessary hardware, technology, and operations which must be demonstrated before the actual flight. The pre-mission planning would initially begin in the first quarter of 1989. This analysis would develop scenarios, operations, training, contingencies, hardware, software, and schedules.

Testing of components would begin one year later and continue until the completion of the final technology demonstration.

The Shuttle and ELV flights would act as platforms for the flight test of required hardware, procedures, and hardware and software test articles, in addition to providing the means to deliver actual mission hardware components for the on-orbit construction of the STV. Final construction of the STV would occur at the Space Station, which will also conduct numerous technology demonstration missions for the effects on long-duration flight operations.

NASA
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Space Flight Systems Directorate

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OFFICE OF EXPLORATION 1988 ANNUAL REPORT
(VOLUME 3)

ADVANCED SPACE PROPULSION WORKSHOP PROCEEDINGS
NASA LEWIS RESEARCH CENTER
APRIL 12-13, 1988

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PARTICIPATING ORGANIZATIONS

NASA HEADQUARTERS
NASA AMES RESEARCH CENTER
NASA LANGLEY RESEARCH CENTER
NASA LEWIS RESEARCH CENTER
NASA MARSHALL SPACE FLIGHT CENTER
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LARGE SCALE PROGRAMS INSTITUTE
PRINCETON UNIVERSITY
SCIENCE APPLICATIONS INTERNATIONAL CORP.
SVERDRUP TECHNOLOGY, INC.
UNIVERSITY OF TEXAS

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Trajectory Analysis

The workshop section on trajectory analysis included a general discussion on the methods of trajectory analysis as they pertain to mission analysis. Also presented were comparisons of high and low thrust trajectories for the manned Mars mission generally. Further presentations showed specifically how key mission parameters can affect the trajectories.

The workshop brought into the fore many ideas and concerns about the trajectory analysis needed for a manned Mars mission. Some of the key observations voiced are the following: the staging of missions for Mars from nodes such as the space station, the moon, lunar libration points, etc., is not fully understood. More study is required for impulsive systems to determine if a broken plane maneuver must be made at the mid point of the trajectory. This has a direct bearing on the propulsion systems and masses of the Mars vehicle. The concern comes about because the inclination of the staging node (in the ecliptic plane) probably will not match that of the outgoing asymptote of the escape trajectory. SAIC and LeRC studies on space shuttle utilization already point to this problem for a relatively simpler system.

Secondly, it is very desirable to provide long duration launch opportunities because the consequence of missing a launch may very well mean wasting a vehicle. This can happen because the synodic periods of Earth to Mars are such that a totally new vehicle design could be required for the next launch opportunity. It was noted that low thrust vehicles may provide this sort of mission availability because they can modulate thrusting and coasting periods during flight. We need to provide an understanding of all propulsion systems capabilities so informed decisions can be made.

Lastly, multimode vehicles seem to be largely unstudied. Such a vehicle would incorporate a high thrust system and a low thrust system. For example, a nuclear thermal propulsion system which could also provide electricity for an ion or mpd thruster system in an alternate mode of operation would be a multimode vehicle. More study should be given these concepts in order to properly ascribe any merit to them. The analysis techniques for such study work probably does not exist and will have to be written.

Analytic Tools

The complexity of the Code Z missions, which combine high and low thrust propulsion options, aerobrakes, lunar and Mars transfers, various transportation modes, lunar and Mars ascents and descents etc., exceeds what has been done for any other seriously contemplated mission. The consensus of the workshop was that while the results of studies to this point have been adequate to meet current requirements, more in-depth studies will require more sophisticated analytic tools than are currently available. Currently available studies make many appropriate simplifying assumptions. Comparison between competing propulsion technologies does not require precision to indicate trends. It appears probable that when decision are in order to choose between the technologies, more precision will be necessary.

The low thrust codes available are in most cases at least fifteen years old. They were developed for computers of that era. The more precise they are, the more difficult solutions are to acquire. Some of the older codes have been adapted to the CRAY X-MP and perform better than they did on the machines for which they were intended, but are still difficult to use. With the capabilities of modern machines, techniques for solving some of the low-thrust problems that previously would overwhelm available machines should be explored.

As discussed in the aerobrake section of this review, the assumption regarding the characteristics of aerobrake have been simplified. The weight of a aerobrake has been assumed to be 15% of the weight braked. It appears that this assumption is probably low for an aerobrake for a Mars return. For more than conceptual studies, aerobrake models that are sensitive to ΔV requirements, plane changes, and packaging characteristics are needed to perform trade studies.

Although optimum solutions to the complex problems associated with the Code Z missions will be difficult to obtain, failure to obtain such solutions will make the results of sensitivity analyses suspect. For instance, if the benefits of improvements or changes in a propulsion technology are desired, approximate solutions to trajectory optimization problems could make these data suspect. The results could be more indicative of the vagaries of approximate solutions than of the change in propulsion technology. These are not important in conceptual studies, but will become important as propulsion technologies are explored in more depth.

An effort should be made to survey the analytic tools available to solve these problems, particularly low-thrust problems. Industry, academia, and government should be queried as to the tools available. This information should be considered at a workshop and a program initiated to develop the tools required to efficiently analyze the mission being considered for the future.

Trade Studies

During the last forty years, a tremendous number of space transportation trade studies have been done by academia, government, and industry. Inevitably, a different set of assumptions were used to do each study. The result of different mission, configuration, and performance ground rules lead to irreconcilable studies and conclusions. In addition, the diversity of independent variables, along with the equally vast numbers of permutations of each variable, exacerbates the dilemma of propulsion system selection using previous studies.

A consistent set of mission, configuration, and performance ground rules must be defined for each of the missions now in vogue. They should be the product of the overall space propulsion community and represent a consensus of their expertise. The rules need to be documented, referenced, and disseminated to the community so that varying propulsion concepts can be exercised and subsequently assessed against some standard.

Ground rule definition is an essential part of initiating the formidable array of trade studies already identified to support the Civil Space Leadership Initiatives (CSLI). These studies include, but are certainly not limited to: propellant selection and acquisition, conventional versus advanced space propulsion, multi-mission utilization, trajectory definition, and abort mode definition. As mission scenarios evolve, trade studies originally oriented towards propulsion will start impacting other systems, thereby necessitating interdisciplinary studies.

Specific trade studies to be initiated in the near term include storable versus cryogenic propellant for descent and/or ascent propulsion, aerobrake versus all propulsive, in-situ propellant manufacturing and usage, and combined versus split missions. Although a significant number of studies have been done to date, some very profound questions remain and represent obstacles to the refinement of the CSLI's.

Figures of Merit

Most previous space propulsion studies have attempted to compare dissimilar propulsion concepts using a limited number of Figures-of-Merit (FOM's). These "goodness criteria" upon which propulsion systems should be subjected to (during an evaluation process for a given mission), should be a variety: cost (R&D, acquisition, operation and support), initial mass in LEO, mass ratio, human factors and environments, trip time/stay time, engine/stage modularity, sizing for delivery to LEO, launch opportunity, abort mode tolerance, reusability, and probably others. In addition, the FOM's are of varying priority and interdependent. This will drive the propulsion system selection process to use a combination of the FOM's.

How the FOM's are quantified is a function of the mission, maturity of the technologies, pre-supposed space infrastructure, and others. Therefore, it is vital that the mission, configuration, and performance ground rules be unambiguously defined. Their numerical values for each system should be arrived at by actual flight article data (if available) at best and disciplined calculation at least. "Kentucky windage" should be exhaustively sought and weeded out, to be replace with dependable, referenceable data. FOM definition and assessment, as well as its origin, must be subjected to, and survive, a peer review process.

Embarking on propulsion system trade studies prior to a community agreement on FOM's will produce an incomplete understanding of each system's attributes and shortcomings. A consensus may develop on the suitability of a certain system based on selected, better understood FOM's -- only to have a rude awakening at a later time when different, less favorable FOM findings materialize. Clearly, a forum is needed to discuss, identify, and quantify FOM's within the space propulsion community. This is particularly important for those associated with the preliminary assessment of the manned lunar/Mars expeditions. The scale and complexity of these missions call for a global understanding of relevant FOM's, how they are to be quantified, and how their use should aid propulsion system selection.

High Power Electric Propulsion

Advanced Electric Propulsion (EP) was considered in the context of technology and mission analysis. Emphasis was placed on high power/high Isp concepts such as Ion and Magnetoplasmadynamic (MPD) propulsion, which show the greatest promise for high energy planetary exploration missions.

Technology Issues: The consideration of high power EP options in the 100-10000 kWe range, while basing their feasibility upon low power or short duration laboratory tests is an area of concern. Ion engines have been operated at power levels of 10 kWe or less for thousands of hours; however, only a small amount of data exists for short duration, 100 kWe tests. Similarly, MPD thrusters have been run for 100's of hours at 100 kWe power levels in the past, as well as in MWe pulses; none of these data are satisfactory proof of the utility of these devices for actual long duration, high power missions. High power operation, which is required for reasonable trip times to the Moon or the planets, introduces questions of electrode lifetime and system design. Electric propulsion systems are unique in that they also require advanced high power supplies, which have also yet to be demonstrated.

The ion engines' apparently superior efficiency relative to the MPD (80% for ion vs. 60% for MPD) has led to the question, 'Why even consider the MPD thruster?'; in consideration of the lack of either system in the necessary high power regime needed for advanced missions, this question can only be answered once the actual feasibility and performance of these devices has been demonstrated at the conditions required for space travel.

The sensitivity of mission performance to the performance of the power/EP system used has been amply demonstrated by the trajectory analyses presented at this workshop; primarily in the specific power of the system, and the efficiency of the thrusters. A final choice will also include questions of reliability, size, and complexity. Presently, both forms of high power electric propulsion are on the brink of demonstrating sustained, high power operation; until tnis is accomplished, no one system should be chosen on the basis of technology projections.

Nuclear Thermal

Compared to most advanced space propulsion concepts, nuclear thermal has a leg-up when it comes to technology maturity and test data. Programs in the 1960's and early 1970's lead to hardware development and ground testing of major components. This technology also offers attractive performance capabilities by combining relatively high thrust with moderately high specific impulse. These qualities make it a leading candidate for consideration in the ongoing manned lunar/Mars scenario definition.

This interest has spawned concern over how much of the knowledge generated by the earlier programs is still retrievable. Because of the length of time that has transpired, it is not readily apparent how much of the expertise is still in place and how much has either moved on, retired, or died. Only after the state of the technology is assessed can the work yet to be done be addressed.

A workshop should be convened that would gather the existing sum of knowledge: people, reports, mothballed hardware, if any, and salvageable test facilities. Only then can trade studies be defined and values of figures of merit be arrived at. It could also drive reactivation of former programs and facilities if payload performance requirements for such a system hold firm.

Space Nuclear Power Operation Issues

Pound for pound, nuclear power probably represents the leading candidate for a light-weight, high-power supply in space applications. Nuclear thermal and nuclear electric propulsion systems continue to be leading contenders for future missions. Nuclear reactors appear unique in their ability to supply power at levels expected by most lunar or Mars base scenarios. The apparent inevitability of reactors in space raises many thorny issues that must be overcome if civil use of them is ever to get off the ground.

Guidelines on reactor-Space Station proximity, and the amount and orientation of shielding required, will be a formidable task. Conditions for start-up of a vehicle using nuclear propulsion also need to be defined. Both concerns are not well understood. Compatibility of aerobrakes and reactors, whether it (the reactor) is part of the propulsion system or merely cargo, and the concomitant human safety issues are yet to be explored.

Still another set of concerns centers around the initial launch of reactors to LEO. It is expected that the ongoing DOD sponsored SP-100 program will be the vanguard for subsequent space reactor programs in this and other areas. It remains to be seen what requirements will be levied by Range Safety and other government entities outside of NASA on the launch vehicle community. Indeed, the non-technical concerns will no doubt dominate this technology, much like it has with its sister field of earth-based nuclear power. How the agency plans to manage the safety, emotional, and political problems associated with this field should be of primary interest to all.

Several action items call for attention and require participation from more than just the propulsion and power communities. Incorporating the tasks inevitably associated with reactors in or around the Space Station should be folded into the Space Station requirements documents at the earliest opportunity. Shielding requirements, both in the Space Station neighborhood and interplanetary space, must be defined. Reactor-aerobrake compatibility is a big unknown that deserves attention. Developments in the SP-100 program should be closely monitored so as not to duplicate efforts or mistakes. Finally, and perhaps most importantly, a plan on how to approach the image problem nuclear power has with the public should be a fundamental concern to the agency.

Aerobrake

Mass Fractions for Mars or Lunar Mission Aerobrakes Appear to be Extrapolated from LEO-GEO Studies and May be Quite Optimistic

The mass of an aerobrake, composed of the structure elements and thermal protection system (TPS), is a function of the predicted peak heating rate, dynamic pressure and total heating. These parameters are influenced by the amount of energy removed from the trajectory, the entry velocity (Ve), and the vehicle parameters such as the lift/drag ratio and ballistic coefficient (beta=mass/coefficient of drag X area). Figures 1 and 2, from NASA TM-100031 by Gene Menees, illustrate how these parameters (entry velocity, L/D, and the ballistic coefficient) affect minimum altitude, maximum heating rates, g-loads and maximum total pressure.

Extrapolating aerobrake mass fractions from the LEO-GEO studies may be optimistic for a number of reasons. First of all, the entry velocities and orbital energy of GEO to LEO trajectories are lower than typical Earth aerocapture scenarios and, therefore, undergo lower heating rates and loading. Also the heating mechanisms for a LEO-GEO mission may be very different than what a Mars or Moon mission may encounter. For example, some LEO-GEO studies predict a heating phenomenon called nonequilibrium radiation because of the blunt shape of the aerobrake and the high perigee altitudes. This phenomenon may not be present in a planetary aerocapture. Furthermore, because of their relatively lower heating rates, LEO-GEO aerobrakes utilize flexible TPS materials which are substantially lighter than ablatives or rigid TPS. This is an option a Mars or Lunar mission may not be able to use.

Can Nuclear Systems be Combined with Aerobrakes?

Combining an aerobrake with nuclear systems poses several challenges. Nuclear reactors normally are separated from the payload by long trusses. Can a heavy structure, such as a reactor, cantilevered on a long truss withstand the g-loads and forces of an aeropass? How will it be protected from the hot gases encountered? Obviously a hot reactor can not be pulled in close to the cargo or crew area without extensive shielding. Finally, what are the political sensitivities of a nuclear system performing an aeropass about the Earth or a planet?

What is the Feasibility of Packaging a Manned Spacecraft Behind an Aerobrake

Experts must rigorously test any aerobrake concept before it is used for a planetary mission. The presence of man will require even greater reliability. Can a reliable system be designed in a timely fashion to be used on these missions? Furthermore, after long periods of weightlessness, can the crew and equipment tolerate the high g-loads, the buffeting and the maneuvering encountered during an aeropass? An injury sustained by a crew member during this portion of the mission could jeopardize the mission. Some method of monitoring, and repairing if necessary, the integrity of the aeroshield is especially important for a manned mission. Can small holes from space debris and micrometeoroids be detected? Will the aerobrake require the additional mass of a shield to protect it during the heliocentric portion of the mission? In addition, can the Martian atmosphere be characterized well enough to attempt a manned aeropass? Finally, because the Martian atmosphere is not very dense, an aeromaneuver might have to be performed at relatively low altitudes. Can the position and state vectors be determined accurately enough at Mars to prevent catastrophe?

Mission Analysis for Missions Including Aerobrakes Appears to Include Several Soft Assumptions about the Capabilities of Aerobrakes

Can mission analysts simply assume an aerobrake maneuver can replace a propulsive burn? There may be some constraints placed on an actual mission such as rendezvous conditions that may be difficult to obtain. The aerobraking community and TPS material experts must define the range of feasible aerobrake entry conditions for manned planetary missions. With these conditions, defined by aerobraking performance predictions and material limits and characteristics, mission analysts can then tailor heliocentric trajectories to satisfy these constraints.

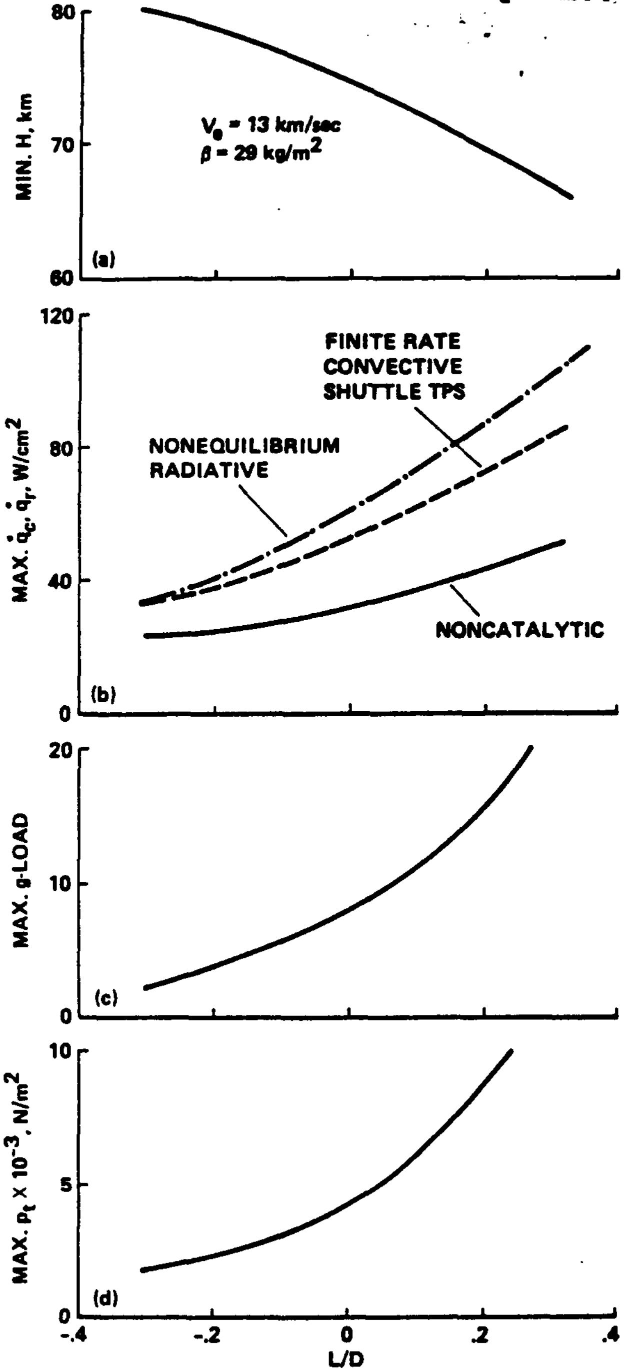

Fig. 15 Earth L/D effects at V_e = 13 km/sec: a) minimum altitude; b) maximum heating; c) maximum g-load; d) maximum total pressure.

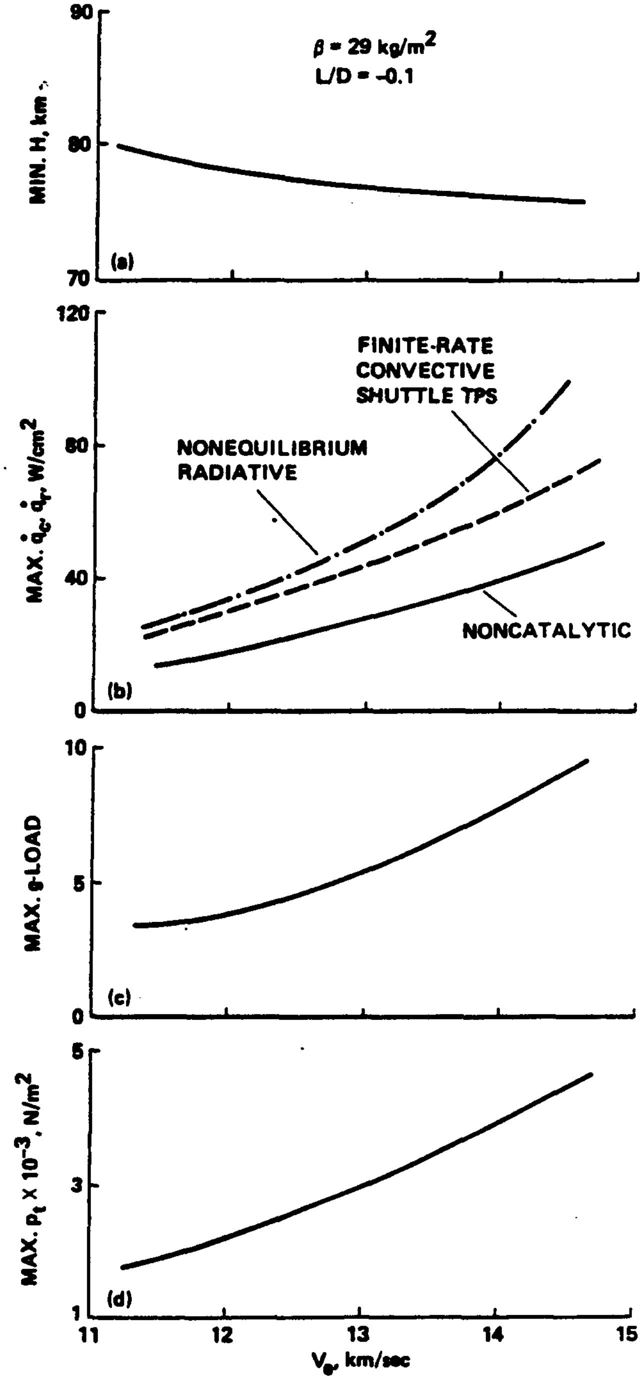

Fig. 16 Earth V_e effects for L/D = -0.1: a) minimum altitude; b) maximum heating; c) maximum g-load; d) maximum total pressure.

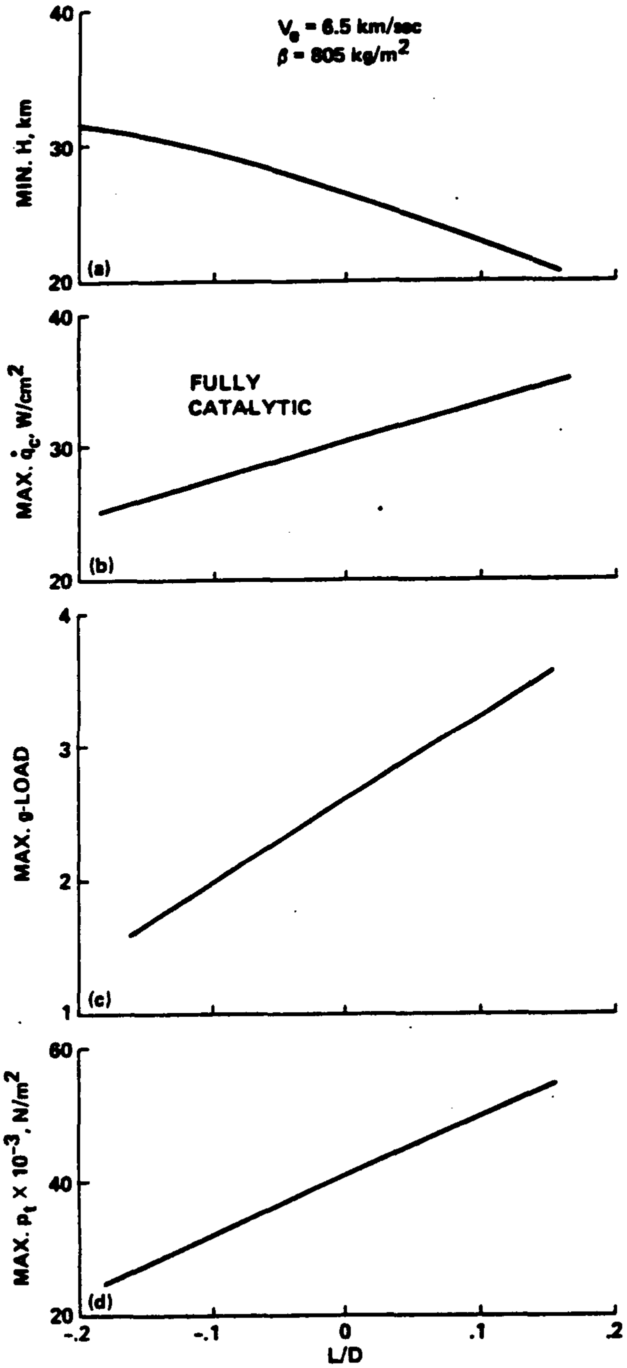

Fig. 7 Mars L/D effects at V_e = 6.5 km/sec: a) minimum altitude; b) maximum heating; c) maximum g-load; d) maximum total pressure.

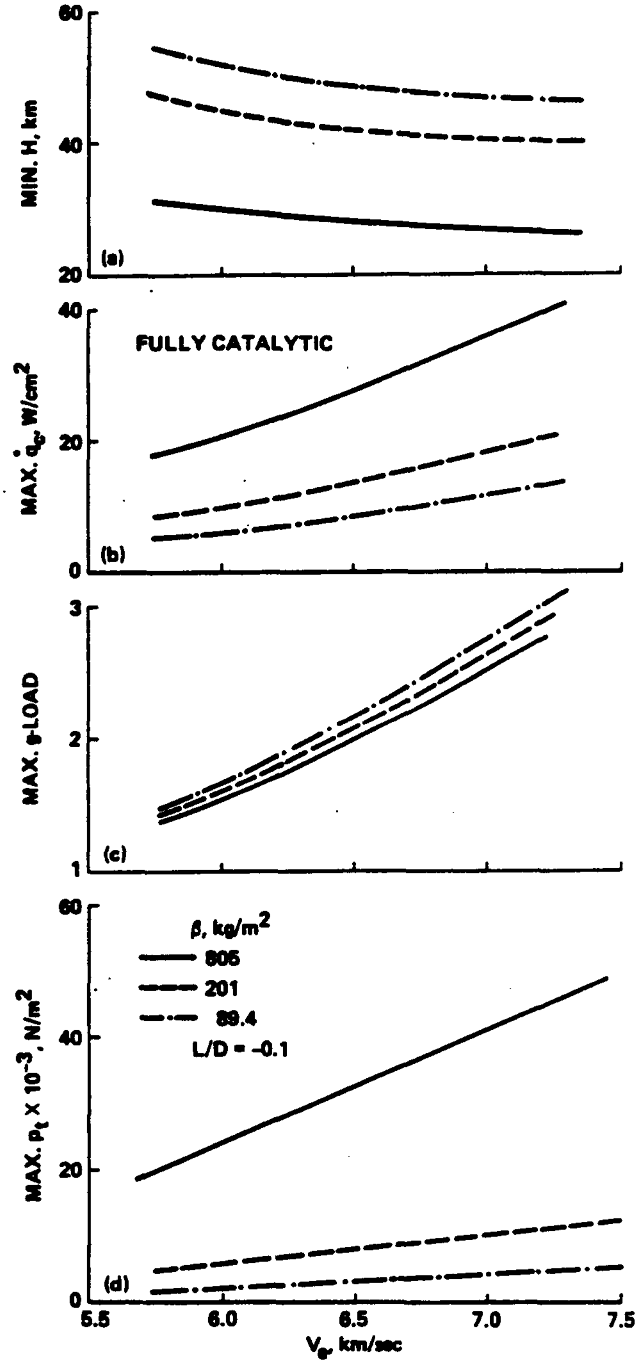

Fig. 8 Mars V_e effects at L/D = -0.1: a) minimum altitude; b) maximum heating; c) maximum g-load; d) maximum total pressure.

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TRAJECTORY ANALYSIS

OBSERVATIONS

- IMPACT OF INITIATING MARS MISSIONS FROM SPACE STATION, THE MOON, OR OTHER NODES IS NOT WELL UNDERSTOOD. FOR INSTANCE, LAUNCHING FROM SPACE STATION COULD REQUIRE LARGE BROKEN PLANE MANEUVERS MIDWAY IN FLIGHT TO MARS, AFFECTING SPACECRAFT PROPULSION REQUIREMENTS

- LAUNCH OPPORTUNITIES FOR HIGH AND LOW THRUST SCENARIOS ARE DIFFERENT. LONG OPPORTUNITIES ARE HIGHLY DESIRABLE TO AVOID WAITING FOR NEXT OPPORTUNITY

- MULTI-MODE PROPULSION CONCEPTS MAY OFFER ADVANTAGES

ACTIONS

- SYSTEMATICALLY STUDY AND UNDERSTAND IMPLICATIONS OF INITIATING BOTH MARS AND LUNAR MISSIONS FROM TRANSPORTATION NODES

- UNDERSTAND LAUNCH OPPORTUNITY CHARACTERISTICS OF HIGH AND LOW THRUST PROPULSION SYSTEMS AND ADVANCED CONCEPTS (E.G. GAS CORE ROCKET) THAT MAY BE CAPABLE OF BOTH HIGH THRUST/HIGH SPECIFIC IMPULSE OPERATION

- INVESTIGATE MULTI-MODE PROPULSION SCENARIOS

ANALYTIC TOOLS

OBSERVATIONS

- A VARIETY OF KNOWN COMPUTER PROGRAMS EXIST FOR BOTH HIGH AND LOW THRUST SYSTEMS, BUT MANY OF THESE PROGRAMS ARE LIMITED IN CAPABILITY RELATIVE TO THE CURRENTLY PERCEIVED MISSION ANALYSIS REQUIREMENTS
- THERE IS A NEED FOR MULTI-MODE MISSIONS ANALYSIS CAPABILITY
- THERE NEEDS TO BE A BETTER UNDERSTANDING OF THE TYPES AND CAPABILITIES OF AVAILABLE ANALYTIC TOOLS, ESPECIALLY FOR LOW THRUST SYSTEMS

ACTIONS

- CONDUCT A SURVEY OF INDUSTRY AND ACADEMIA TO DETERMINE CAPABILITY OF CURRENT LOW THRUST ANALYTIC TOOLS
- INITIATE WORK TO REVISE PROGRAMS AND/OR BUILD NEW ANALYTIC CAPABILITIES TO SATISFY MISSION ANALYSIS NEEDS

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TRADE STUDIES

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OBSERVATIONS

- A LARGE NUMBER OF TRADE STUDIES HAVE BEEN IDENTIFIED, I.E. STORABLE VS. CRYO FOR ASCENT/DESCENT, MULTI-MODE STUDIES, TECHNOLOGY/MISSION COMPATIBILITY, COMBINED VS. SPLIT MISSIONS, AEROBRAKING VS. PROPULSIVE, IN SITU PROPELLANT USAGE, ETC.

- TO PERFORM THESE TRADE STUDIES, A COMMON AND CONSISTENT SET OF GROUND RULES AND ASSUMPTIONS NEEDS TO BE ESTABLISHED TO ASSURE MEANINGFUL AND CREDIBLE COMPARISON

ACTIONS

- COORDINATE THE ESTABLISHMENT AND UNDERSTANDING OF COMMON GROUND RULES AND ASSUMPTIONS

- INITIATE TRADE STUDIES

- DETERMINE THE IMPACT OF IN-SITU PROPELLANT USAGE ON ENGINE DESIGN

FIGURES OF MERIT

OBSERVATIONS

- FIGURES OF MERIT FOR LUNAR AND MARS MISSIONS ARE NUMEROUS AND COMPLICATED BY THEIR INTERACTIONS
- BRIEFLY, FIGURES OF MERIT INCLUDE COST (OPERATIONAL VS. LIFE CYCLE VS. DEVELOPMENT DESIGN AND TESTING), MASS IN LEO, MASS RATIO, HUMAN FACTORS AND ENVIRONMENTS, TRIP TIME/STAY TIME, MODULARITY, PACKAGING, LAUNCH OPPORTUNITY, ABORT TOLERANCE, REUSABILITY, ...

ACTION

- CONDUCT WORKSHOP TO PROMOTE COMMON UNDERSTANDING OF FIGURES OF MERIT AND FACILITATE THE PROCESS OF COMPARING SCENARIOS

HIGH POWER ELECTRIC PROPULSION

OBSERVATION

- POTENTIAL OF HIGH POWER ELECTRIC PROPULSION SYSTEMS (MPD & ION) REMAINS TO BE DEMONSTRATED OVER THE PARAMETER RANGES OF INTEREST. HIGH POWER OPERATION AND THRUSTER LIFETIMES ARE KEY ISSUES

ACTION

- SERIOUS SUPPORT BY THE PATHFINDER PROGRAM IS REQUIRED TO DETERMINE THE FEASIBILITY AND PRACTICALITY OF HIGH POWER MPD AND ION SYSTEMS

NUCLEAR THERMAL PROPULSION

OBSERVATIONS

- MISSION STUDIES INDICATE NUCLEAR THERMAL PROPULSION HAS ADVANTAGES FOR MARS AND LUNAR MISSIONS
- NUCLEAR PROPULSION INTRODUCES SAFETY/EMOTIONAL/POLITICAL QUESTIONS

ACTION

- CONDUCT WORKSHOP TO ESTABLISH STATE OF NUCLEAR THERMAL TECHNOLOGY AND ASSESS POTENTIAL

SPACE NUCLEAR POWER OPERATION ISSUES

OBSERVATIONS

- IMPLICATIONS AND CONSTRAINTS OF OPERATING NUCLEAR SYSTEMS NEAR THE SPACE STATION IS NOT WELL UNDERSTOOD AND REQUIRES FURTHER STUDY
- COMPATIBILITY OF AEROBRAKING AND NUCLEAR SYSTEMS IS LESS UNDERSTOOD
- ACCEPTABILITY OF LAUNCHING REACTORS IS CURRENTLY BEING STUDIED FOR SP-100. RESULTS OF SP-100 EXPERIENCE WILL IMPACT CONSIDERATION OF OTHER SYSTEMS

ACTIONS

- INITIATE STUDY OF USING NUCLEAR SYSTEMS IN PROXIMITY TO SPACE STATION OR OTHER MANNED SPACECRAFT
- CONDUCT SHIELDING STUDIES IN SUPPORT OF MANNED OPERATIONS
- INVESTIGATE COMPATIBILITY OF NUCLEAR SYSTEMS WITH SAFETY AND PACKAGING CONSTRAINTS OF AEROBRAKES

AEROBRAKE

OBSERVATIONS

- MASS FRACTIONS FOR MARS MISSION OR LUNAR MISSION AEROBRAKES APPEAR TO BE EXTRAPOLATED FROM LEO-GEO STUDIES AND MAY BE QUITE OPTIMISTIC
- CAN NUCLEAR SYSTEMS BE COMBINED WITH AEROBRAKES?
- WHAT IS THE FEASIBILITY OF PACKAGING A MANNED SPACECRAFT BEHIND AN AEROBRAKE?
- MISSION ANALYSIS FOR MISSIONS INCLUDING AEROBRAKES APPEARS TO INCLUDE SEVERAL "SOFT" ASSUMPTIONS ABOUT CAPABILITIES OF AEROBRAKES

ACTION

- CRITICALLY EVALUATE RANGE OF ASSUMPTIONS OF AEROBRAKE CAPABILITIES

THE MISSION AND TRAJECTORY DESIGN PROCESS

FOR

MANNED LUNAR AND MARS MISSIONS

Paul A. Penzo
Jet Propulsion Laboratory

Presentation to the
Space Systems and Technology
Advisory Committee

Briefing at
NASA Lewis Research Center
Cleveland, Ohio

April 12 and 13, 1988

TRAJECTORY ANALYSIS AND MISSION DESIGN

Trajectory Analysis

This is the generation of trajectory capabilities, energy requirements, flight times, and geometric parameters, independent of specific mission constraints and requirements.

Examples: APOLLO circumlunar trajectories
Multiple gravity assist (Voyager, Mariner 10, Pioneer 11)
Circulating trajectories

(Extensive handbooks of data are sometimes generated.)

Mission Design

This is a detailed study to determine the acceptable Earth launch and planet arrival conditions based on specific launch year and capability, and on celestial, science, systems, and operational constraints.

Then for specific sets of trajectories, parametric data is generated to aid spacecraft and systems design, and science observations.

MISSION DESIGN PHASES

Determine Mission Feasibility

Develop Trajectory Space (Conic)

Bound Trajectory Space Based on Constraints

Perform Trade Studies

Provide Data for Other Project Elements

Generate Precision Data (Integrated)

Take Part in Mission Operations

GRAVITY ASSIST -- A LIMITED SOLUTION

Gravity Assist is created by trajectory bending and results in momentum transfer between spacecraft and planet (bigger is better).

<u>Advantages</u>

Post flyby orbit can be shaped by selecting flyby altitude and latitude.

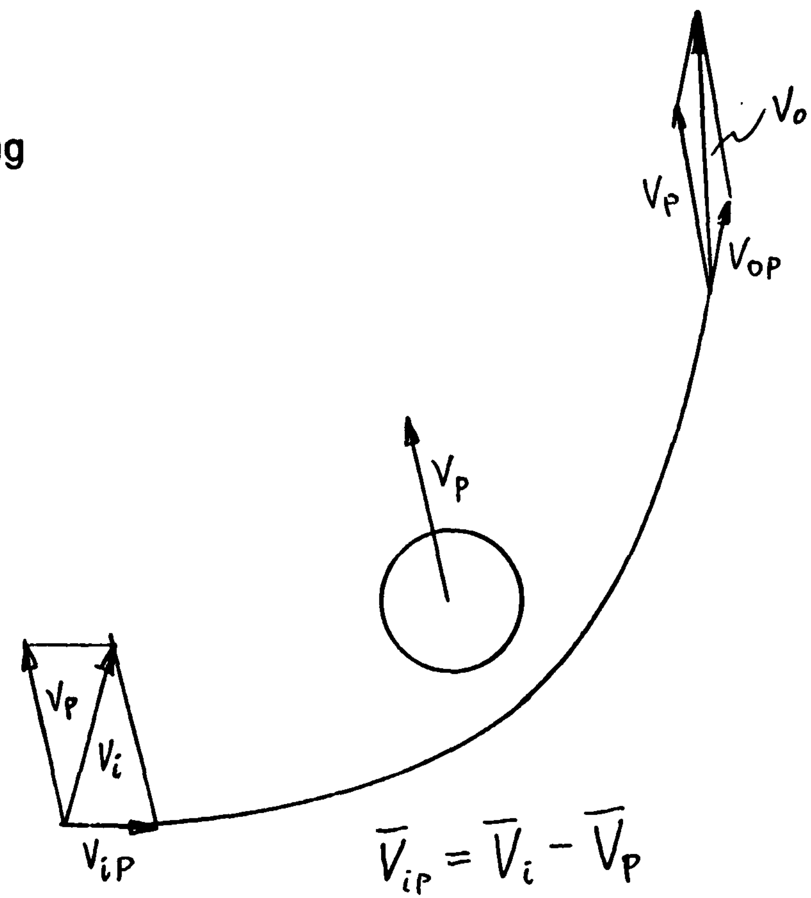
Precision navigation is required and available.

<u>Disadvantages</u>

Planet may not be massive enough for the required trajectory bending. (subsurface flyby)

Planet may not be in the right place at the right time (Grand Tour every 175 years).

Flyby geometry may conflict with the science requirements at the flyby planet.



$$\overline{V}_{iP} = \overline{V}_i - \overline{V}_P$$

$$V_{iP} = V_{oP}$$

$$V_o > V_i$$

MISSION DESIGN PROPULSION SYSTEM CONSIDERATIONS

Gravity assist provides added flexibility for one-way missions, but does not help with sample (or human) return.

Aerocapture at Mars is a serious need to accomplish Mars surface sample return, and the need will be even greater for manned Mars missions.

Low thrust provides great flexibility in interplanetary trajectory shaping, with little increase in transfer time. This is not the case for the Moon.

Chemical/gravity assist is a "hybrid" system and illustrates the great flexibility gained in combining two or more propulsion systems or techniques.

Low thrust and aerobreaking require in-flight test and development experience, which can be gained by flying high science and engineering payoff robotic missions.

A large power source in space (SP-100 and up) will be a generic need for many types of advanced propulsion systems.

A good deal of trajectory analysis has yet to be done - for hybrid systems to be used for Mars missions, and to better understand trajectory options for the Earth Moon system.

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ADVANCED PROPULSION for MANNED LUNAR and MARS MISSIONS

Presentation to the
Planetary Mission Analysis Meeting

Briefing at
NASA Lewis Research Center

Bryan Palaszewski
Jet Propulsion Laboratory
April 12–13, 1988

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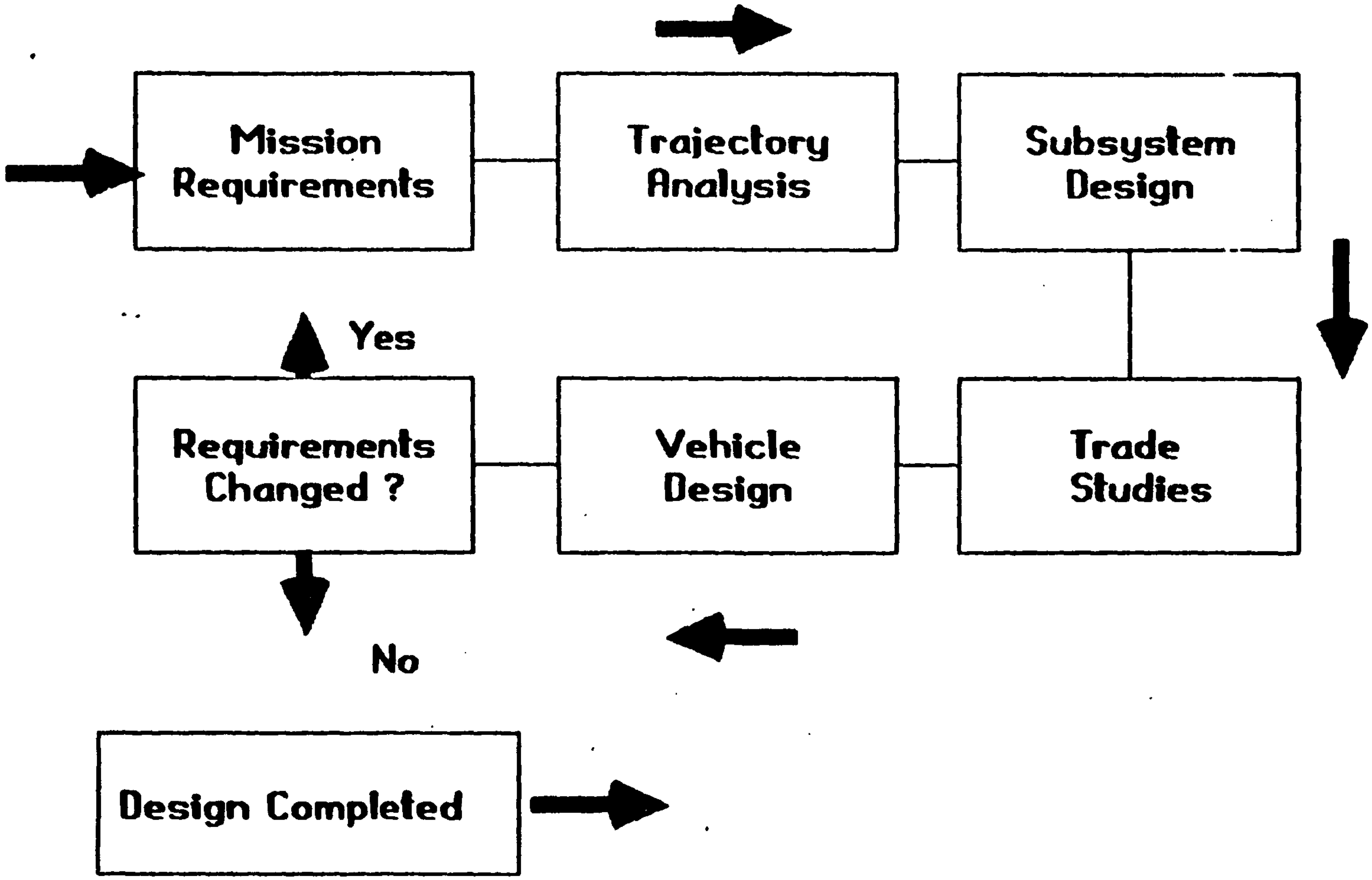
Introduction

- What Is Mission Analysis ?
- Propulsion System Parameters and System Masses
- Mission Analysis Results
- Observations

Mission Analysis Process



Mission Requirements,

Mission Requirements

- Mission Requirements Provide the Mission Description and the Top-Level Mission Goals
- Detailed Planning and Understanding of the Goals and Objectives are Required
- Results:
  - Mission Description
  - Science Objectives
  - Payload Masses
  - Life-Science Requirements

Lunar Mission Payloads

Element	Mass (kg)
Surface Exploration Probes	4000
L2 Relay Satellite	500
Lunar Cargo Delivery	35000, 22500
Lunar Base Set-Up	32000
Lunar Base Ops	32000, 19500

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Mars Mission Payloads

Element	Mass (kg)
Piloted Mission	
Habitation Modules	66000
Earth Return Capsule	5500
	71500
Cargo Mission	
Science Payload and Landers	80000
Earth Return Propellant and Tankage	100000
	180000

Mission Requirements Conclusions

- Orbital Traffic for Lunar and Mars Mission will Be High
- Payload Delivery Schedules Must Be Defined in Detail
- Science Objectives and Life–Science Impacts on the Crews Must Be Defined

Trajectory Analysis

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Trajectory Analysis

- Trajectory Analysis Provides the Flight Path of the Vehicle
- Detailed Computer Simulations are Required
- Results:
  - Trip Time
  - ΔV or Mass Ratios

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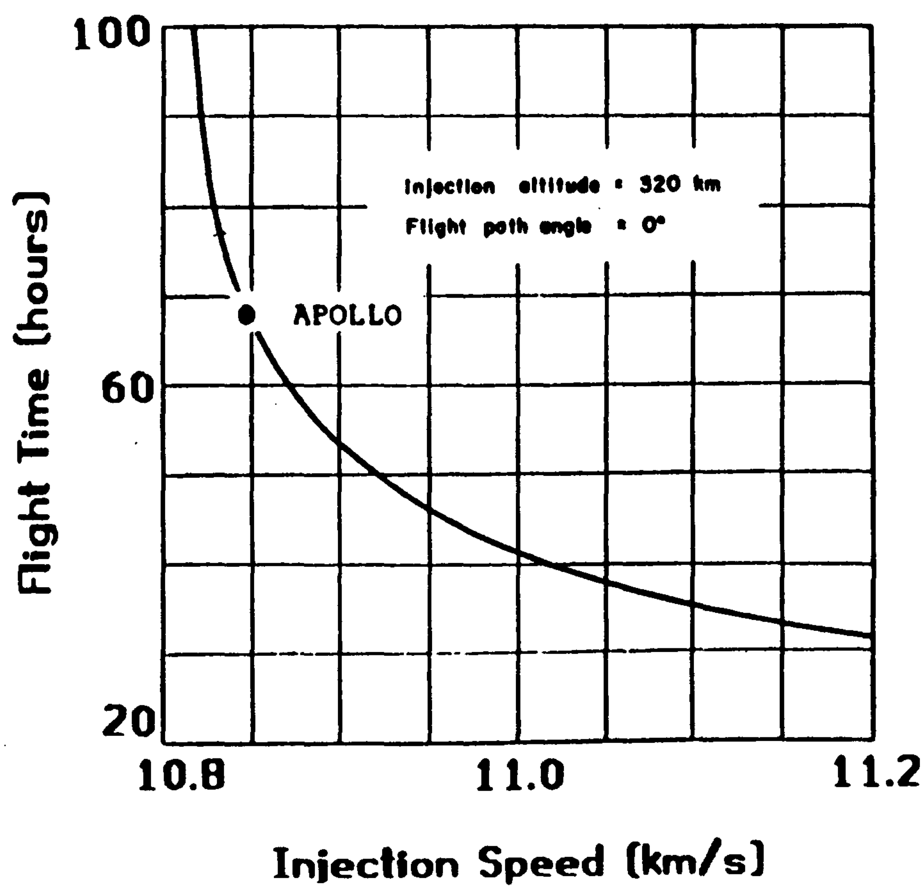
ADVANCED PROPULSION for MANNED LUNAR and MARS MISSIONS

Lunar Trajectories

- Orbit Transfer from 500-km LEO to 110-km LLO

- 10- to 12-day Transfer Time Allows Minimum Transfer Energy



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ΔV
What Does It Mean?

- ΔV Is the Velocity Change Imparted To A Spacecraft During a Rocket Engine Firing
- It Is Related to the Specific Impulse and Mass of the Spacecraft By

$$\Delta V = I_{sp}\, g \ln \left\{ \frac{\text{Initial Mass}}{\text{Final Mass}} \right\}$$

Final Mass = Initial Mass - Propellant Mass

ΔV and Gravity Losses

- There are Three Regimes of Maneuvers
  - Impulsive (Instantaneous or Very-High Thrust)
  - Non-Impulsive and High Thrust
  - Low Thrust

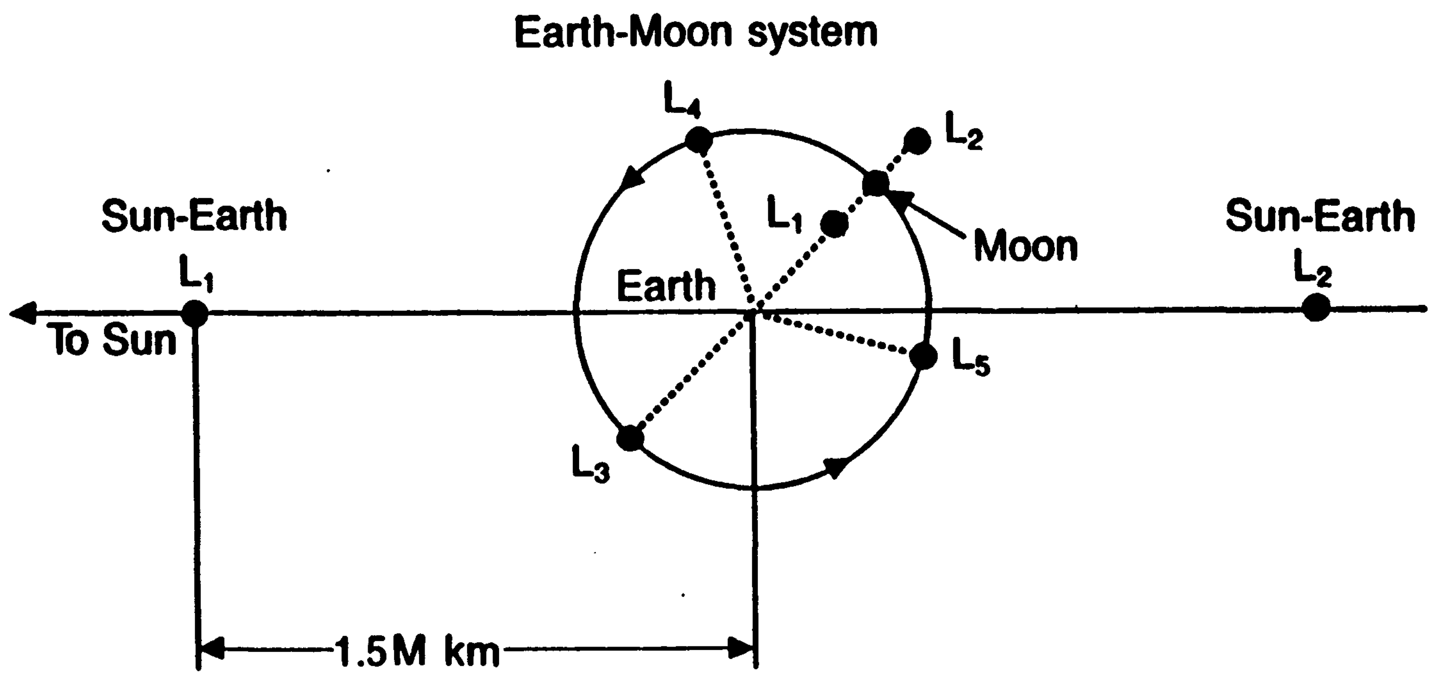
- Gravity Losses Occur When the Maneuver (Engine Firing) Is Not Impulsive

- An Added ΔV to Account for Gravity Losses Must be Included in Any Real-World Analysis of Rocket Propulsion

Lunar Mission ΔV (km/s)

Departure Point	Thrust Level: High	Thrust Level: Low
• LEO-LLO	4.05	8.0
LLO-LEO	1.15	8.0
• LLO-L1	0.85	1.4
L1-LLO	0.85	1.4
• LEO-L1	3.45	7.0
L1-LEO	0.55	7.0
• Earth's Surface-Orbit	9.7	n/a
• Moon's Surface-Orbit	2.0	n/a

EARTH VICINITY LIBRATION POINTS



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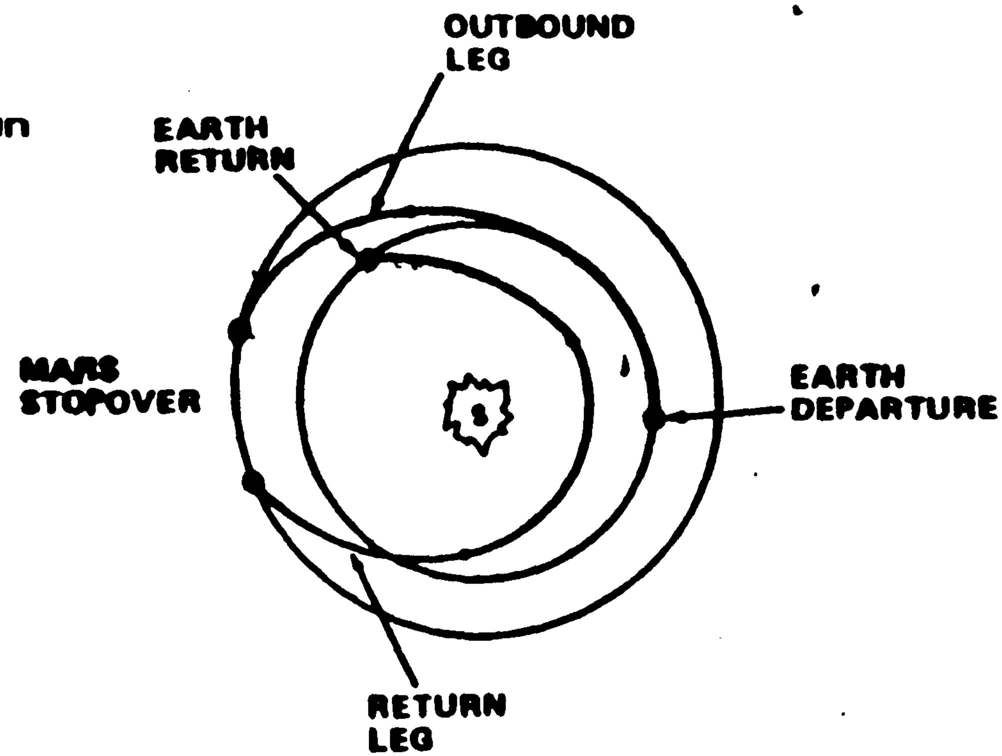
ADVANCED PROPULSION for
MANNED LUNAR and MARS MISSIONS

Fast Missions

- Opposition Launch
  Both Planets on the
  Same Side of the Sun
- Minimizes Trip Time
- Sprint Flight
  Allows Fast Flight
  to Mars

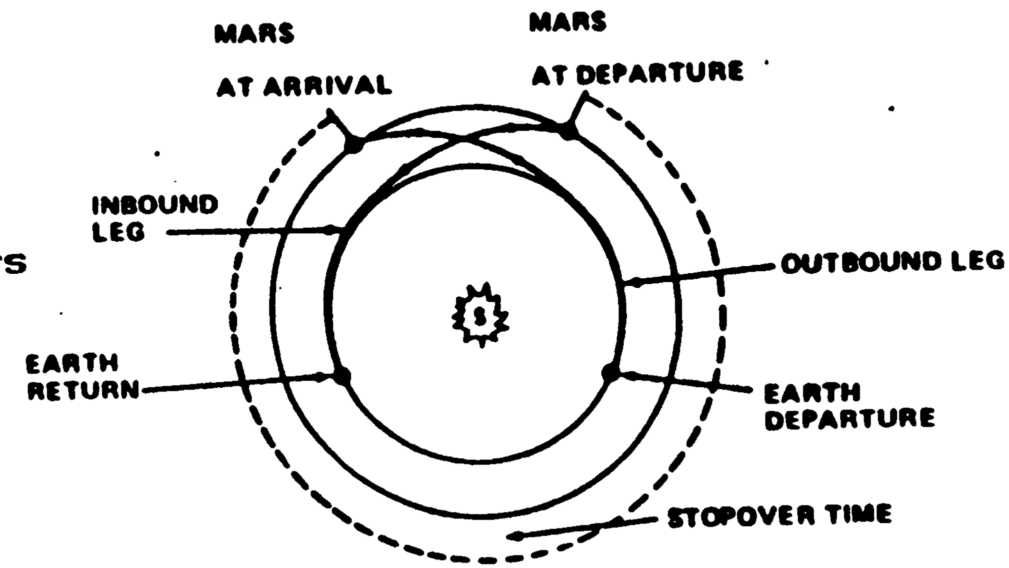


OPPOSITION CLASS
DIRECT STOPOVER MISSION MODE

"Slow" Missions

- Conjunction Launch
  Planets on Opposite
  Sides of the Sun

- Near-Hohmann Transfer
  Lowest Energy Transfers



CONJUNCTION CLASS STOPOVER MISSION

Mars Mission ΔV (km/s)

Departure Point	Thrust Level: High	Thrust Level: Low
• LEO-Mars	4-11	16-20
Mars-LEO	3.5-8	16-20
• LLO-LMO	3-10	10-14
LMO-LLO	3-10	10-14
• L1-LMO	1.1-8	9-13
LMO-L1	1.1-8	9-13
• Mars Surface-Orbit	4.0	n/a

Trajectory Analysis Conclusions

- Low-Energy Trajectories Enable Low Launch Masses in LEO But Require Long Trip Times

- High-Energy Trajectories Require Significantly Larger Propellant Masses in LEO and Enable Short Trip Times

Subsystem Design

Subsystem Design

- Subsystem Design Provides the Masses, Volumes and Other Requirements of the Vehicle
- Mass Models or Point Designs are Required
- Results:
    - Subsystem List
    - Subsystem Masses
    - Power Requirements
    - Thermal Requirements

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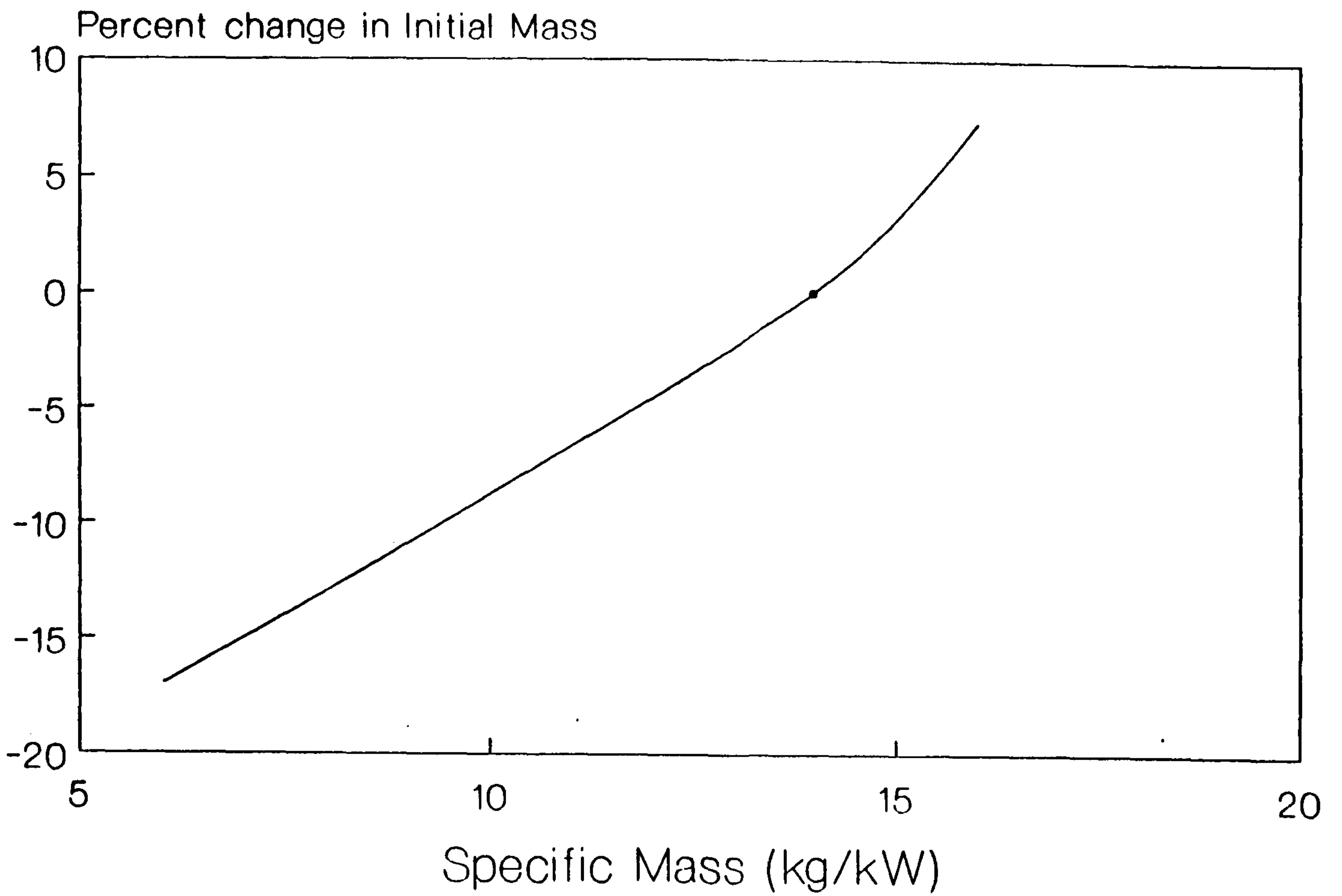
Effect of Specific Mass on Initial Mass

ASAO



EFFECT OF SPECIFIC MASS

- DEFINITION
  The ratio of reactor mass to reactor power output
- Specific mass is a measure of the effectiveness of the power generation system (Inverse of specific power)
- Low values imply better overall system performance

OPTIMIZATION PHILOSOPHY

- Fully optimize round-trip mission
- Allow launch day and heliocentric trip time to vary in the optimization scheme
- All other independent variables are costates or steering directions
- Use analytic approximation for tangential spirals in and out from planets

PERFORMANCE MEASURE

$$\frac{\text{(INITIAL MASS − BASELINE INITIAL MASS)}}{\text{BASELINE INITIAL MASS}}$$

negative ⇒ LESS initial mass

- Vary ONE mission/vehicle parameter and reoptimize trajectory to compute impact on initial mass in LEO
- Hold EVERYTHING else FIXED
- Must consider effects as MISSION specific although they may generalize

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Propulsion System Parameters

Propulsion System	Isp (lbf-s/lbm)	Efficiency
Oxygen/Hydrogen	480	95%(Combustion)
Ion	3000-10000	50-85%(Pj/Pe)
MPD	3000-8000	50%(Pj/Pe)
Arcjet	1500	50%(Pj/Pe)
Nuclear Thermal	900	84%(Pj/Pt)

Propulsion System Technologies

- Power: Reactor Mass — 5–30 kg/kWe
  - Solar Array Mass — 7–10 kg/kWe
  - PPU Mass — 0.5–5 kg/kWe

- Propulsion System:
  - Specific Impulse
  - Electrical Efficiency
  - Mass-Scaling Equations and Point Designs

- Other OTV Subsystems:
  - Mass and Power Models:
  - Point Designs and
  - Scaling Equations

Figure of Merit: Launch Mass

- The Most-Costly Part of Space Exploration Is Launch Into Earth Orbit
- An Important Figure of Merit Is The Total Launch Mass
- Reducing The Total Launch Mass Makes Space Exploration Less Costly and More Easily Achievable

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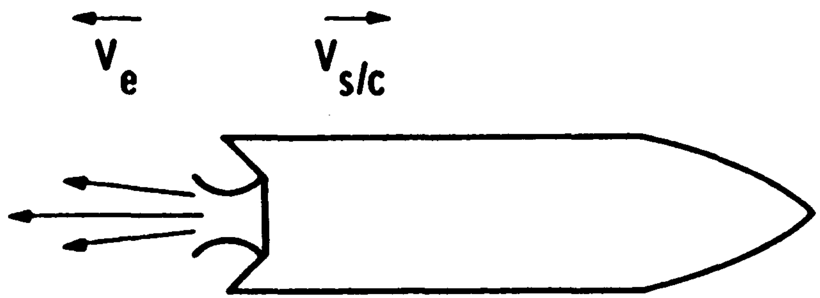
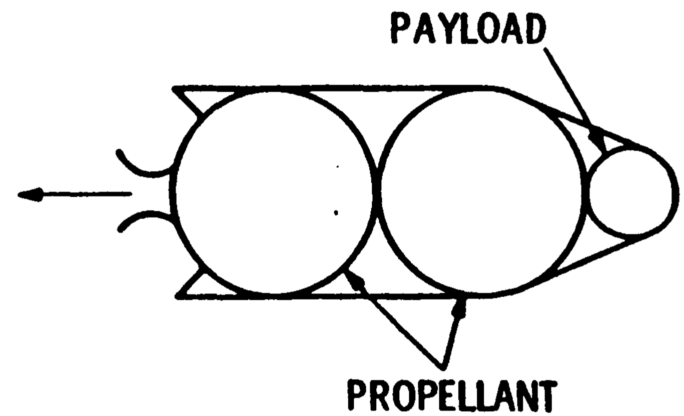
HIGH EXHAUST VELOCITY MEANS LESS PROPELLANT AND MORE PAYLOAD



CHEMICAL PROPULSION

I_{sp} = 300-500 sec



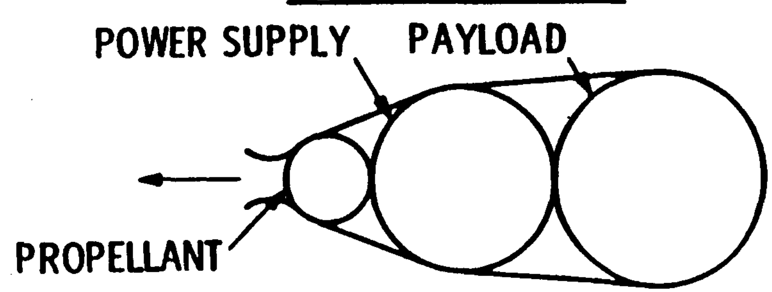
ROCKET EQUATION

$$\frac{M_f}{M_0} = \exp\left(\frac{-\Delta V_{s/c}}{V_e}\right)$$

$$I_{sp} = \frac{V_e}{g_0}$$

ELECTRIC PROPULSION

I_{sp} = 500-10,000 sec



Subsystem Design Conclusions

- Many Parameters Must Be Modeled for Accurate Mass Analyses
- Power System and the Propulsion Module Are the Most-Massive Components

Trade Studies

Trade Studies

- Trade Studies Determine the "Optimum" Operating Points of the Vehicle

- Results:
  - Minimum Trip Time
  - Minimum Launch Mass
  - Maximum Payload
  - Minimum Servicing and Operations

OTV Needs A High Power Level for a Short Trip Time

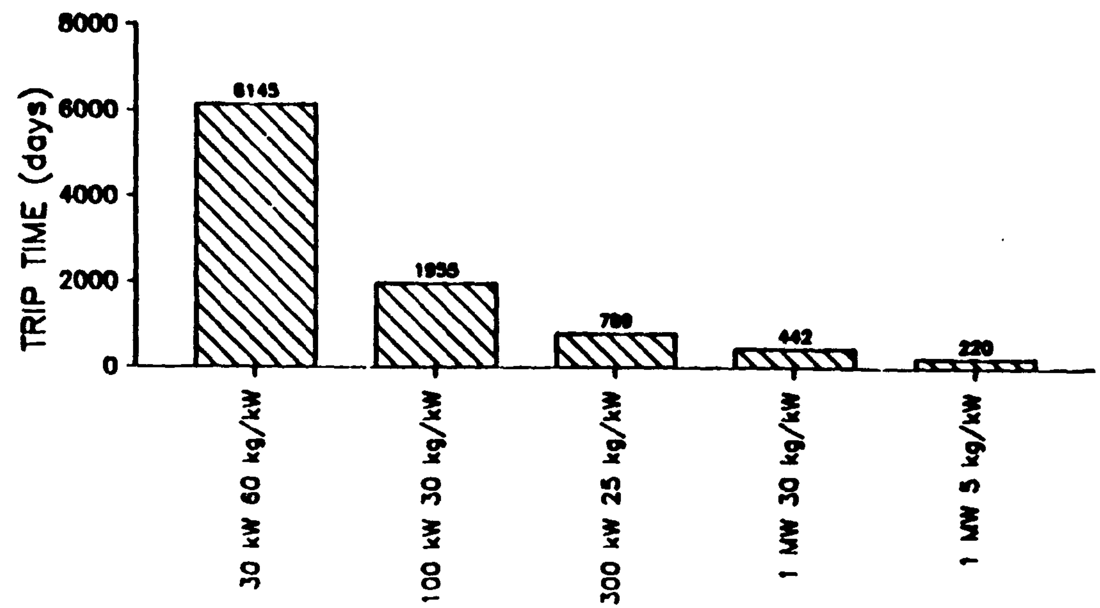
- A 100-kW Power Level OTV Needs 9 Times Longer Trip Time Than a 1-MW Power Level

- A Megawatt Power Level is Required for Short Trip Time

- Trip Time Calculations:
  Isp = 5000 lbf-s/lbm
  Efficiency = 0.70

Xe ION PROPULSION

EFFECT OF POWER LEVEL ON TRIP TIME



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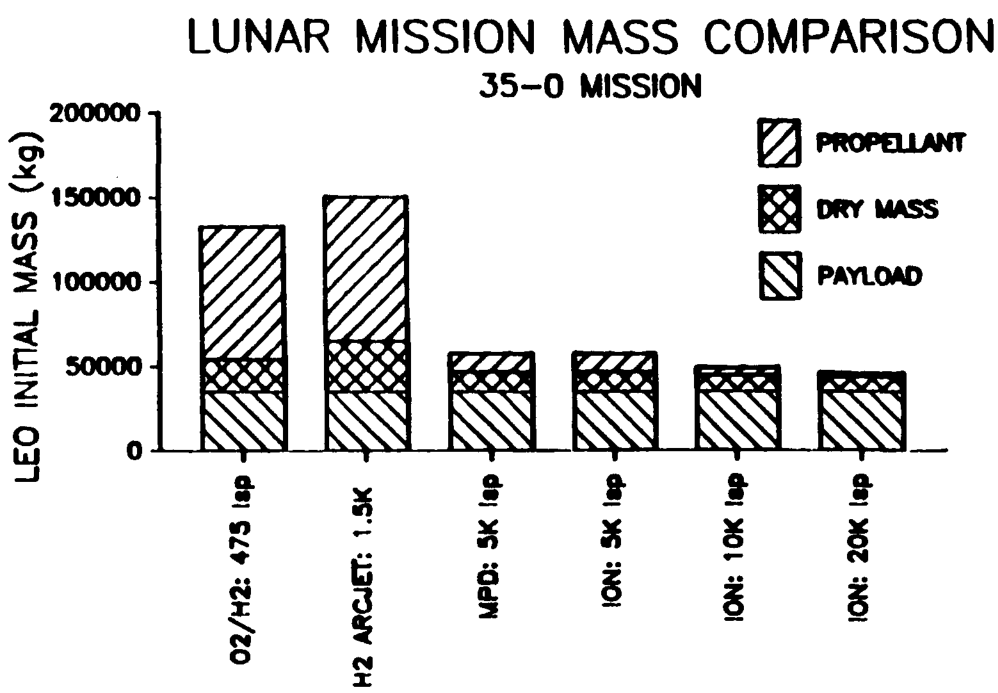
ADVANCED PROPULSION for
MANNED LUNAR and MARS MISSIONS

Ion and MPD Propulsion Significantly Reduce LEO Launch Mass

- Delivery to Lunar Orbit: 35000 kg, 0 kg Returned to LEO
- Ion and MPD Provide Significant Mass Savings Over O2 / H2
- Arcjet Requires a Higher Mass Than O2 / H2



PROPULSION TECHNOLOGY

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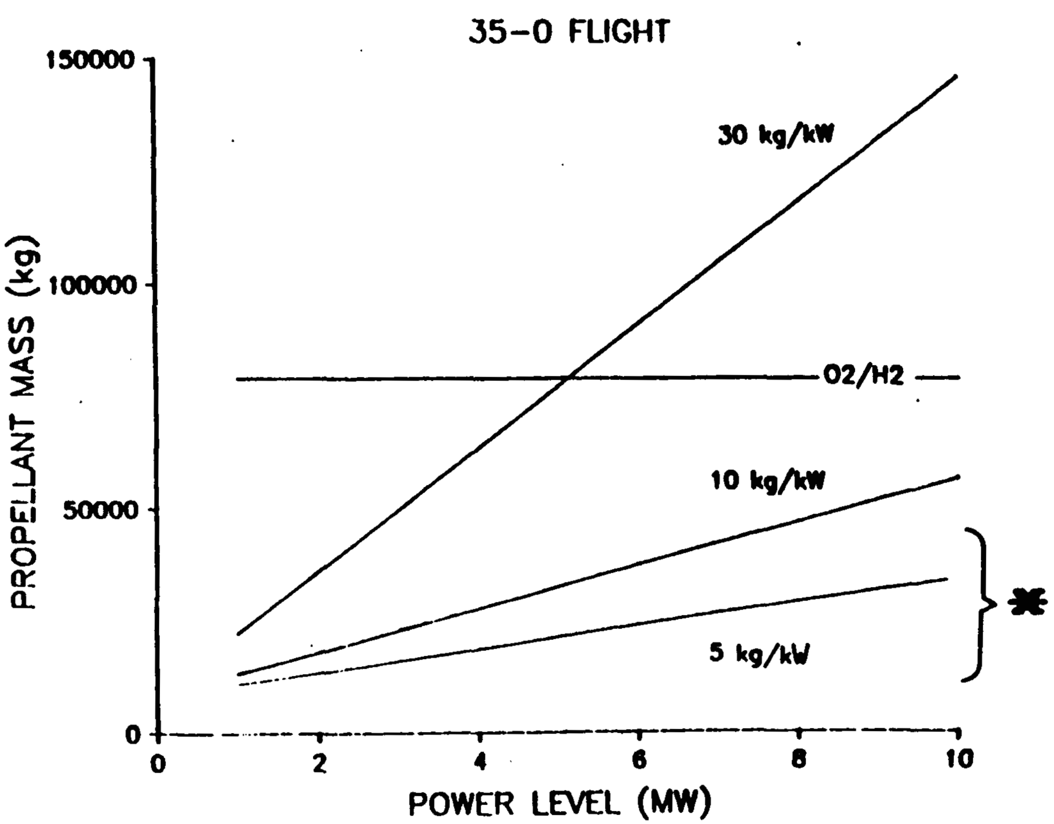
ADVANCED PROPULSION for
MANNED LUNAR and MARS MISSIONS

Lunar OTV Propellant Is Reduced With a Low Reactor Specific Mass

- Low Specific Mass Significantly Reduces the Total Propellant Load

- Xe Ion Propulsion
  5000 lbf-s/lbm Isp
  Efficiency = 0.70



✱ Desired Operating Point

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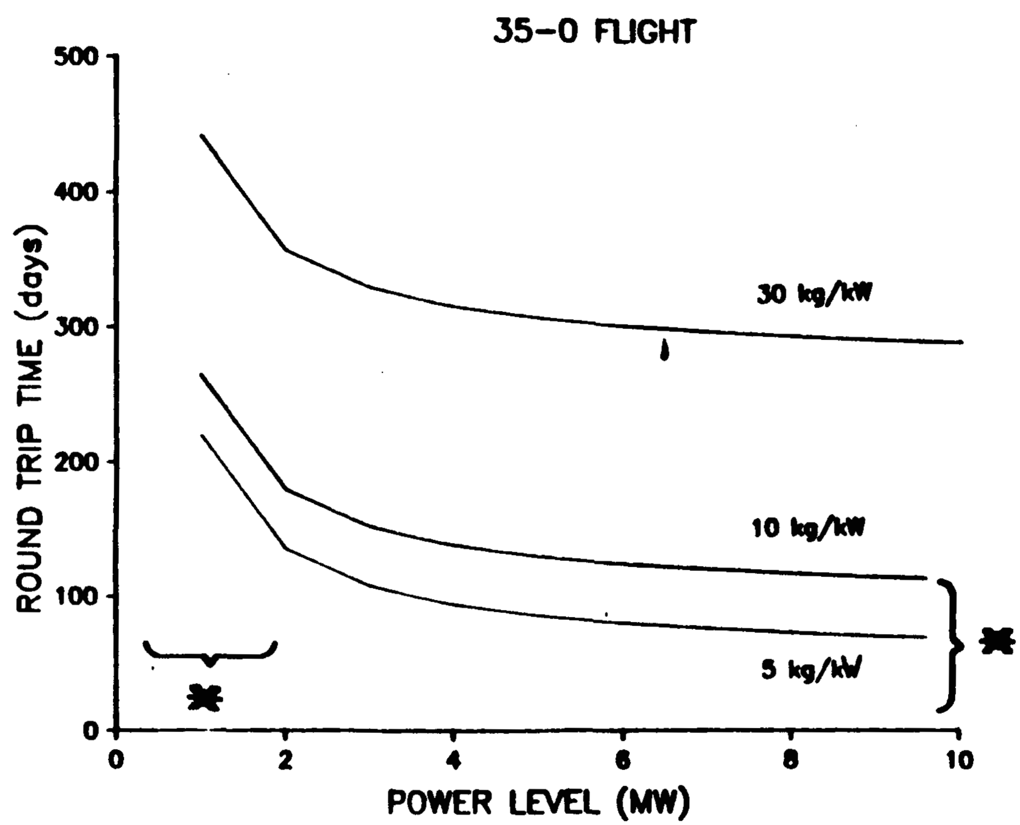
ADVANCED PROPULSION for
MANNED LUNAR and MARS MISSIONS

A Short Lunar OTV Trip Time Requires a Low Reactor Specific Mass

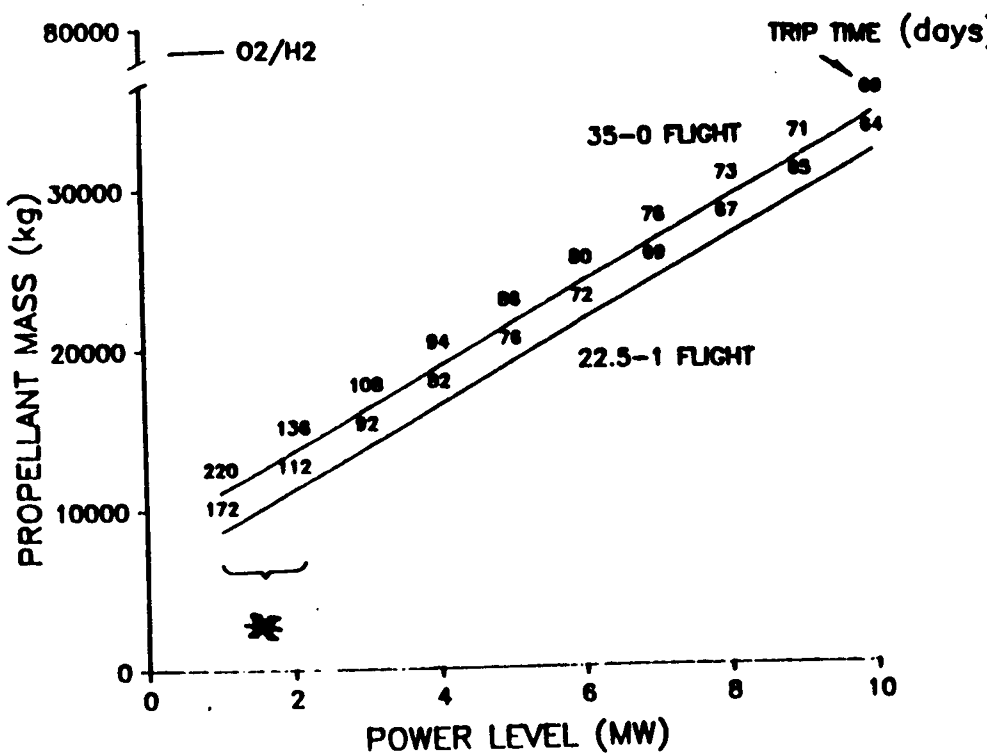
- A Megawatt Power Level at 5 kg/kW Enables Fast Lunar Missions
- Xe Ion Propulsion 5000 lbf-s/lbm Isp Efficiency = 0.70



Lunar Cargo Delivery Propellant Mass Savings Is Greatest at a 1-MW Power Level

- Xe Ion Propulsion
  5000 lbf-s/lbm Isp
  Efficiency = 0.70
- Power System
  Specific = 5 kg/kW

LUNAR CARGO DELIVERY



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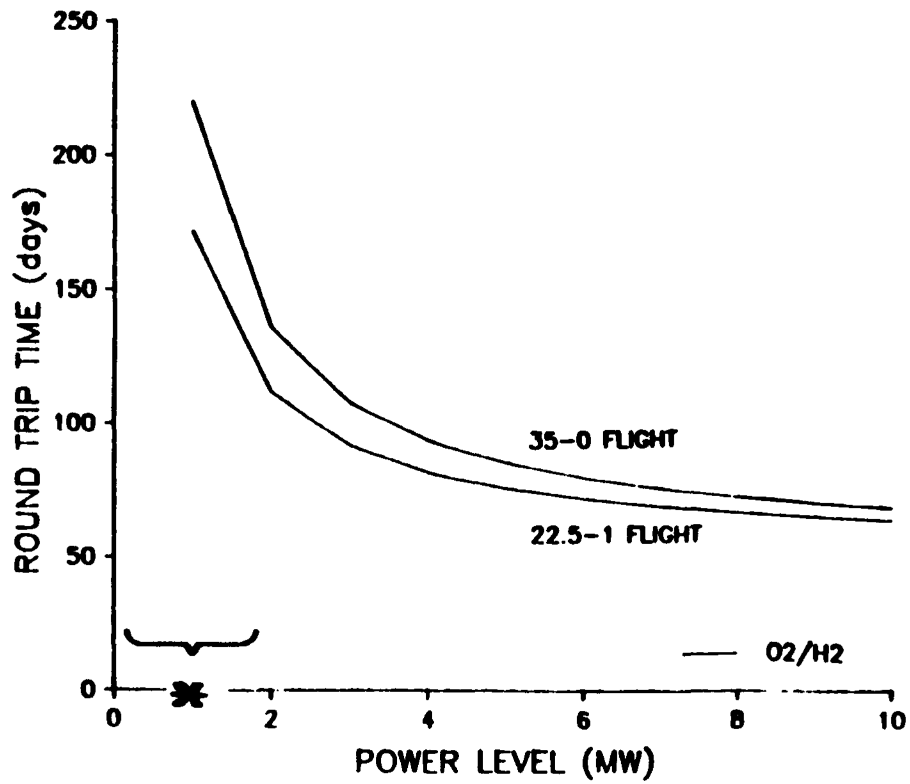
ADVANCED PROPULSION for MANNED LUNAR and MARS MISSIONS

Lunar Cargo Delivery Mission Trip Time Is 200 days With 1-MW Power Level

- Xe Ion Propulsion
  5000 lbf-s/lbm Isp
  Efficiency = 0.70
- Power System
  Specific = 5 kg/kW



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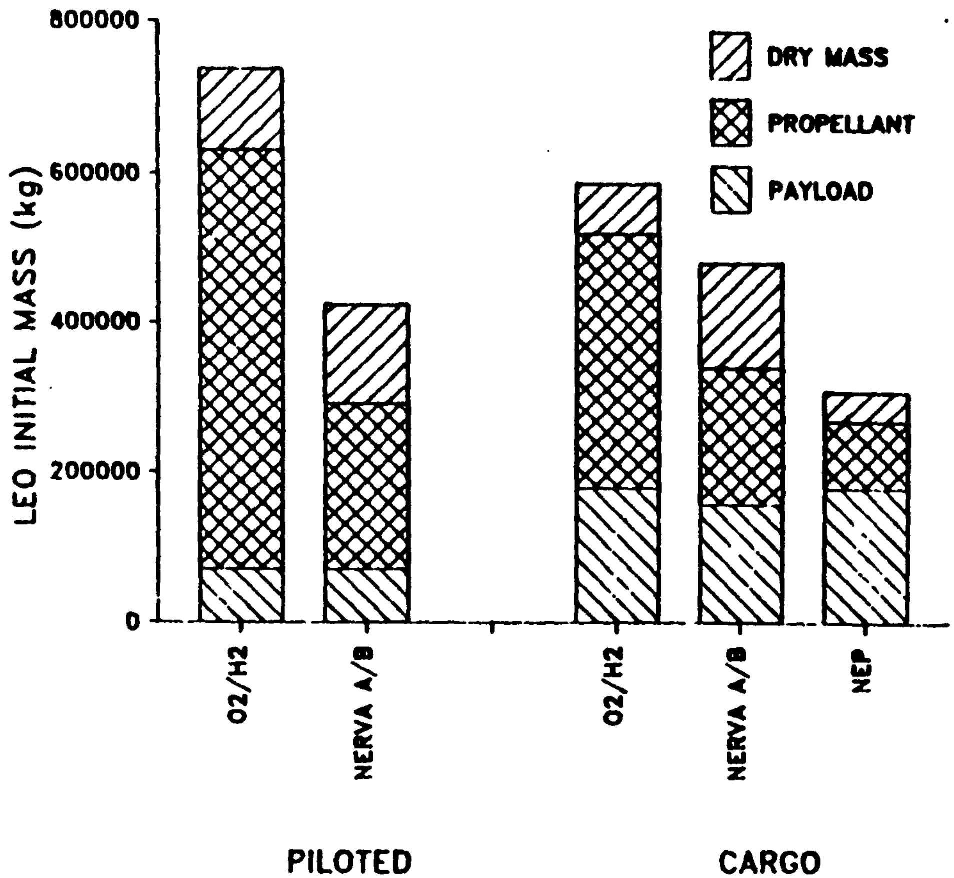
ADVANCED PROPULSION for
MANNED LUNAR and MARS MISSIONS

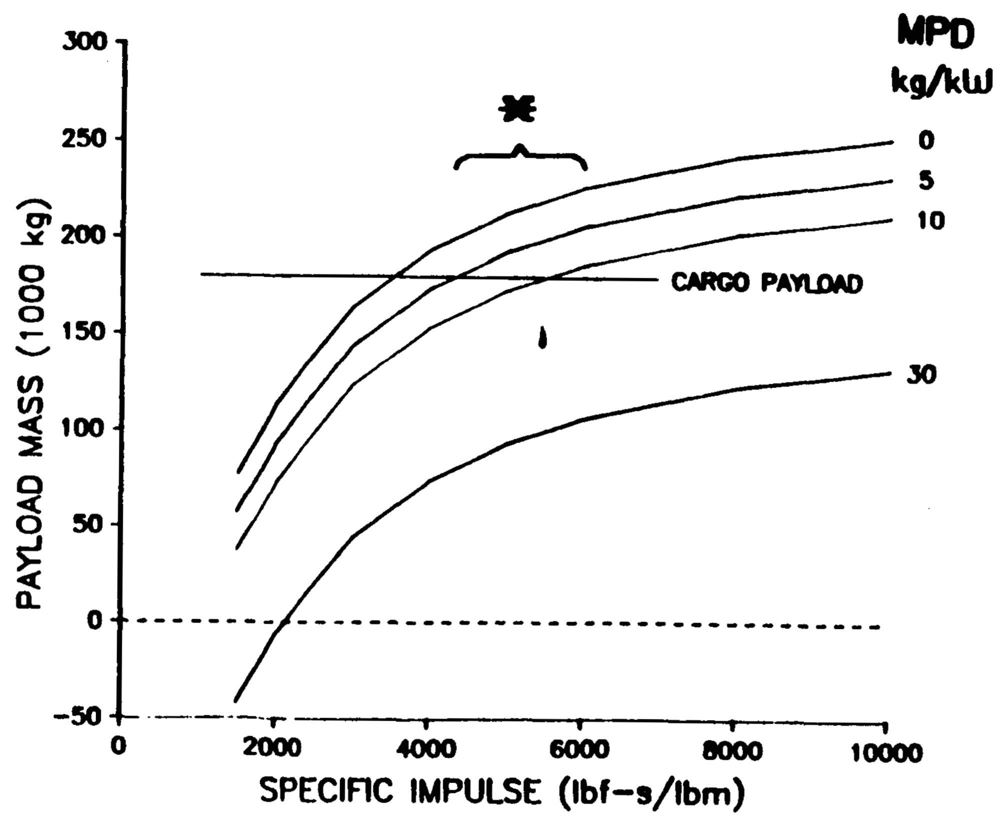
NEP and Nuclear Thermal Allow Significant Sprint Mission Mass Savings

- Nuclear Thermal Significantly Reduces the Piloted Sprint Mission Launch Mass
- Ion and MPD Provide Significant Cargo Mission Mass Savings Over 02 / H2



A Low Power-and-Propulsion System Specific Mass Significantly Increases the Mars Cargo Vehicle's Effective Payload

- Initial Mass: 300,000 kg
  4-MW Power Level
- Isp Below 5000 lbf-s/lbm Does Not Deliver A Significant Payload Fraction
- A Propulsion Specific Mass of Less Than 10 kg/kW Delivers Large Effective Payloads



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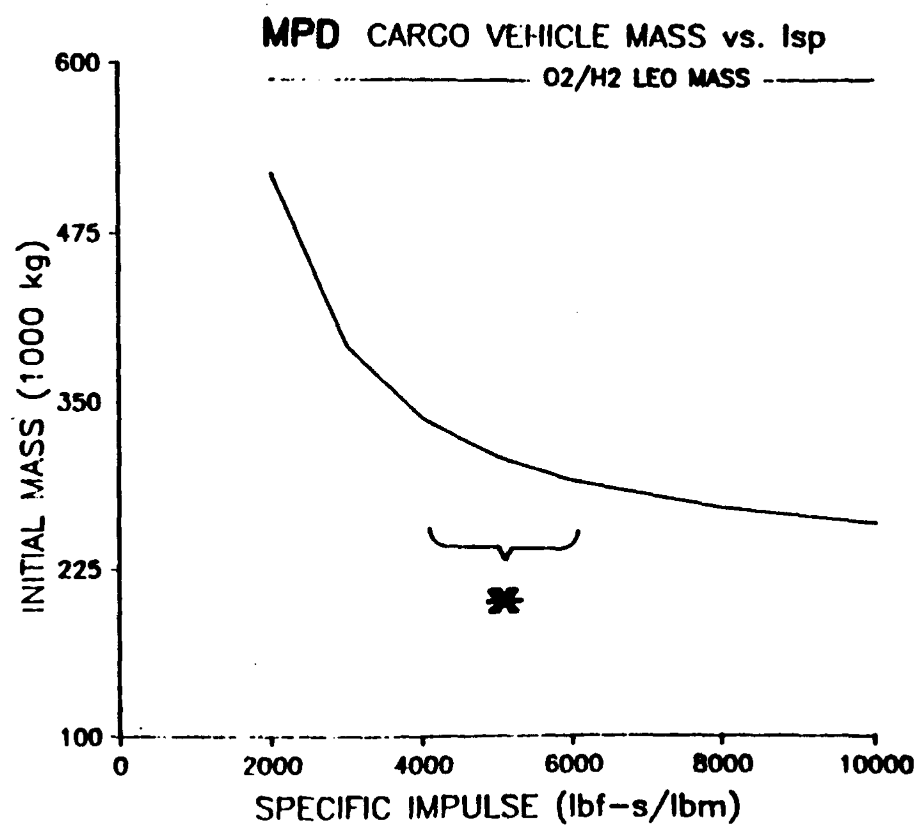
ADVANCED PROPULSION for
MANNED LUNAR and MARS MISSIONS

A 5000-lbf-s/lbm Isp Enables a Near-Minimum Mars Cargo Vehicle Initial Mass

- Payload Mass: 180000 kg
  4-MW Power Level
  Propulsion/Power Mass: 10 kg/kW
- Isp Below 2000 lbf-s/lbm Provides No Significant Mass Savings
- Isp of 10000 lbf-s/lbm Gives the Lowest Mass but a Significantly Longer Trip Time



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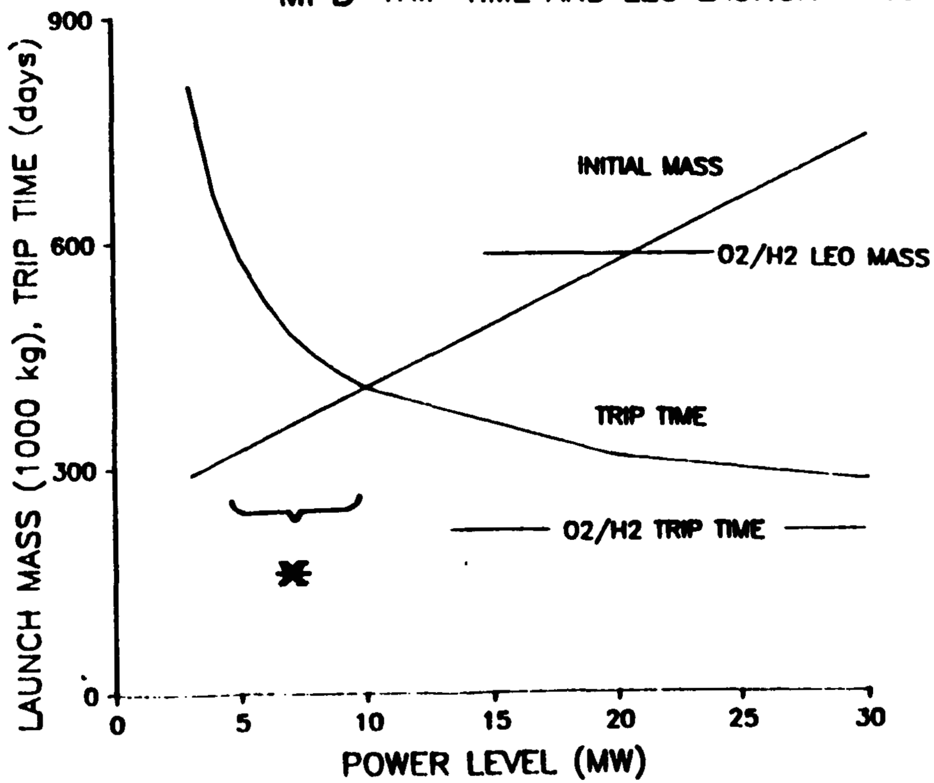
ADVANCED PROPULSION for
MANNED LUNAR and MARS MISSIONS

Mars Cargo Vehicle Power Level Must Be Below 10 MW To Enable A Mass Savings Over O2 / H2

- Payload Mass: 180000 kg
  Propulsion and Power Mass: 10 kg/kW
  Isp = 5000 lbf-s/lbm

- Power Level of 4-10 MW Give Short Trip Time and Mass Savings Over O2 / H2

MPD TRIP TIME AND LEO LAUNCH MASS



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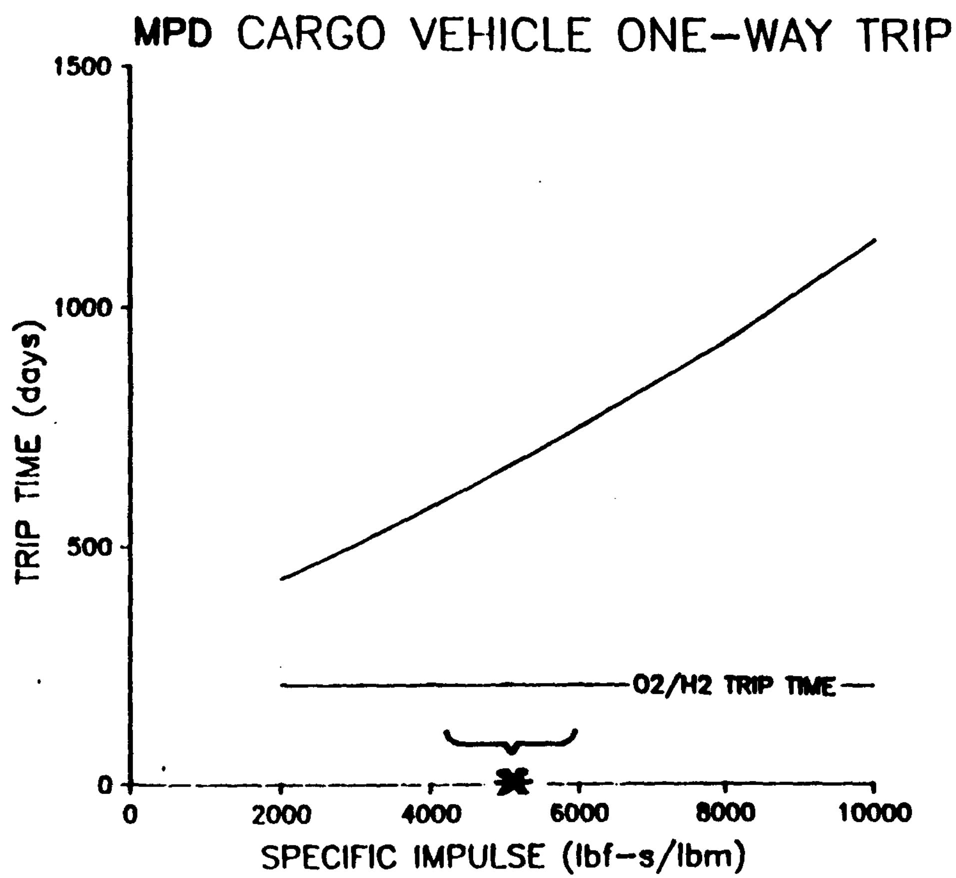
Mars Cargo Vehicle Trip Time Is Short For 5000 lbf-s/lbm Isp

- Payload Mass: 180000 kg
  4-MW Power Level
  Propulsion/Power Mass:
  10 kg/kW

- Isp of 5000 lbf-s/lbm
  Gives Trip Time of 700 days

- LEO Departure - 500 km
  LMO Arrival - 1000 km



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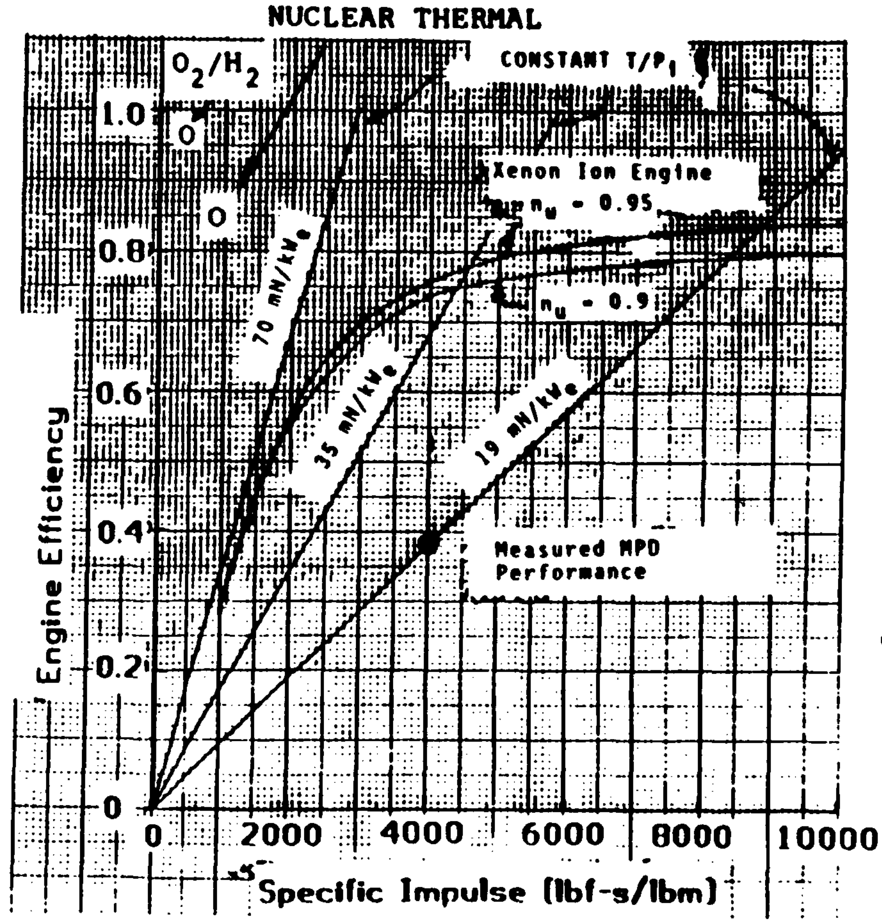
NUCLEAR—ELECTRIC PROPULSION

Propulsion System Comparison

- MPD PROPULSION ENABLES SIMILAR MISSION PERFORMANCE TO ION
- UNRESOLVED SYSTEM COMPLEXITY AND QUALIFICATION ISSUES DO NOT PRECLUDE ION OR MPD
- QUALIFICATION OF MULTI—MEGAWATT NEP IS COMPLEX

  MPD THRUSTER LIFE = 2000 hr
  ABLE TO PROCESS MWs

  ION THRUSTER LIFE = 15000 hr
  MAY PROCESS 100 kW — 500 kW
- HIGHER- EFFICIENCY ION REDUCES
  RADIATOR MASS
  DRY PROPULSION MASS



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ADVANCED PROPULSION for
MANNED LUNAR and MARS MISSIONS

The Number of Electric Propulsion Thrusters Is Dependent on the Power Level

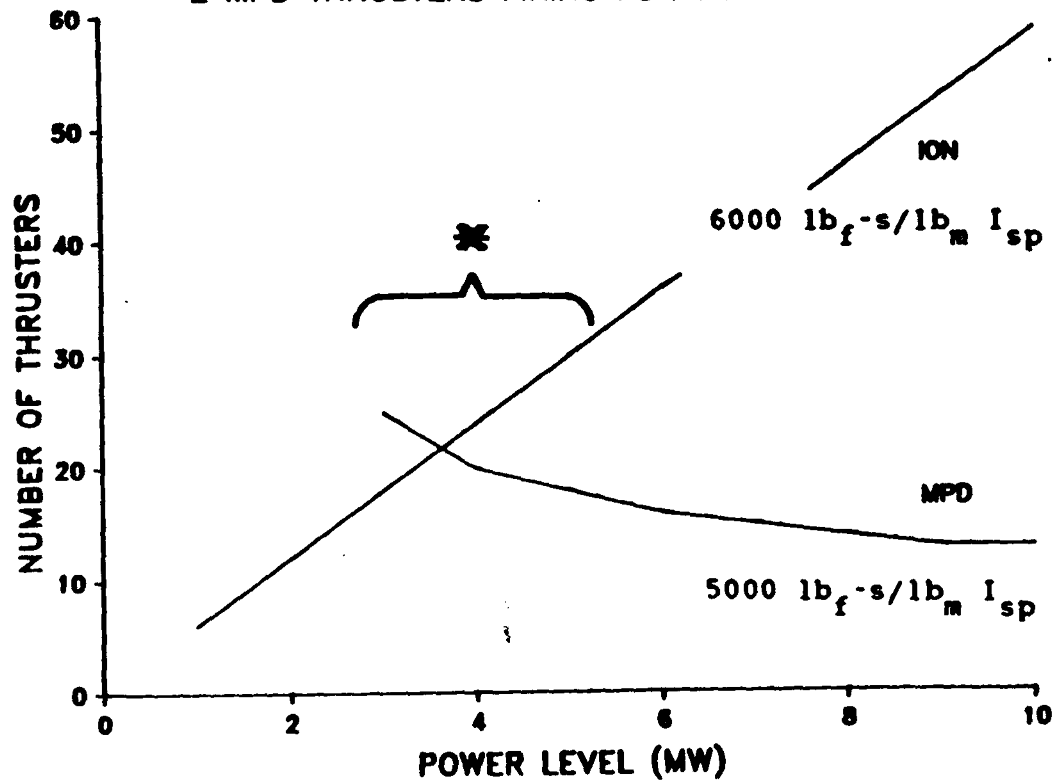
- **Ion Propulsion:**

  Power Per Thruster Is Limited By Accelerator Grid Accuracy

- **MPD Propulsion:**

  May Process Many MW Per Thruster Reducing the Number of Thrusters

214 kW per ION THRUSTER
2 MPD THRUSTERS FIRING FOR THRUST VECTORING



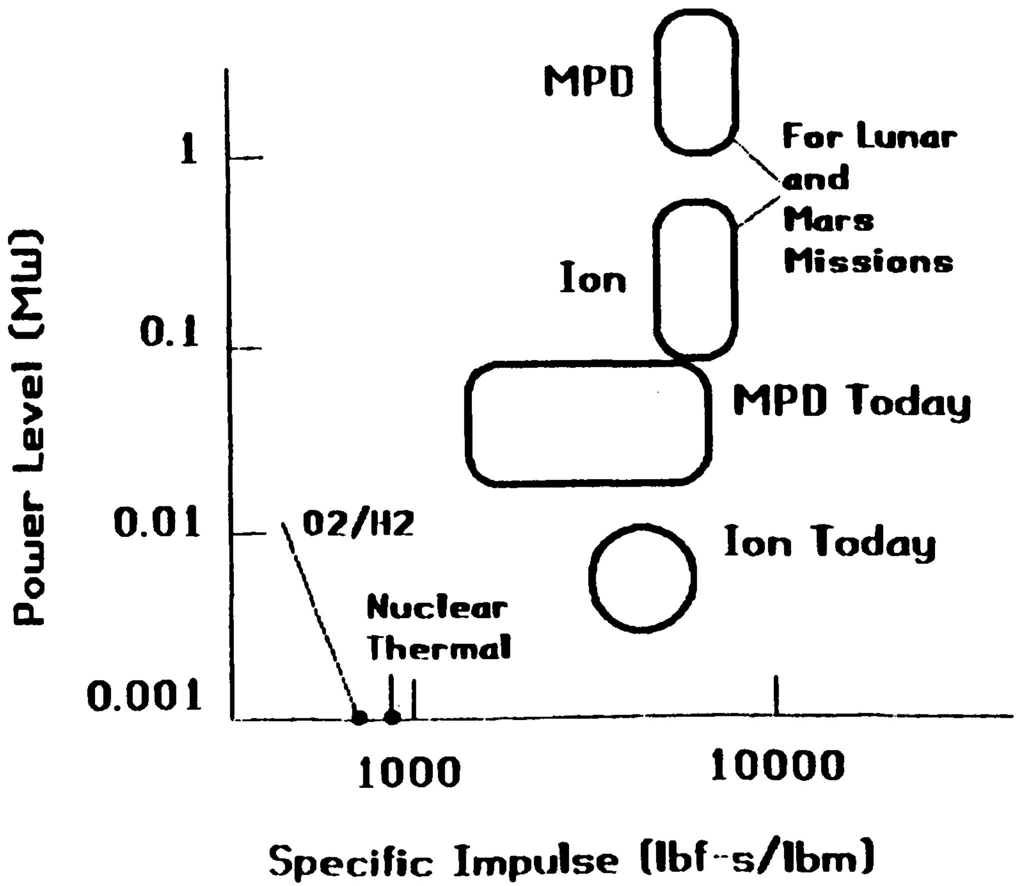
Electric Propulsion Technology Has Not Been Demonstrated at MW Power Levels

- Ion Propulsion:
  Steady-State
  10 kW per Thruster
  Demonstrated

  Short Firing of a
  130-kW Thruster
  at 10000 lbf-s/lbm

- MPD Propulsion:
  Steady-State
  72 kW per Thruster
  Demonstrated

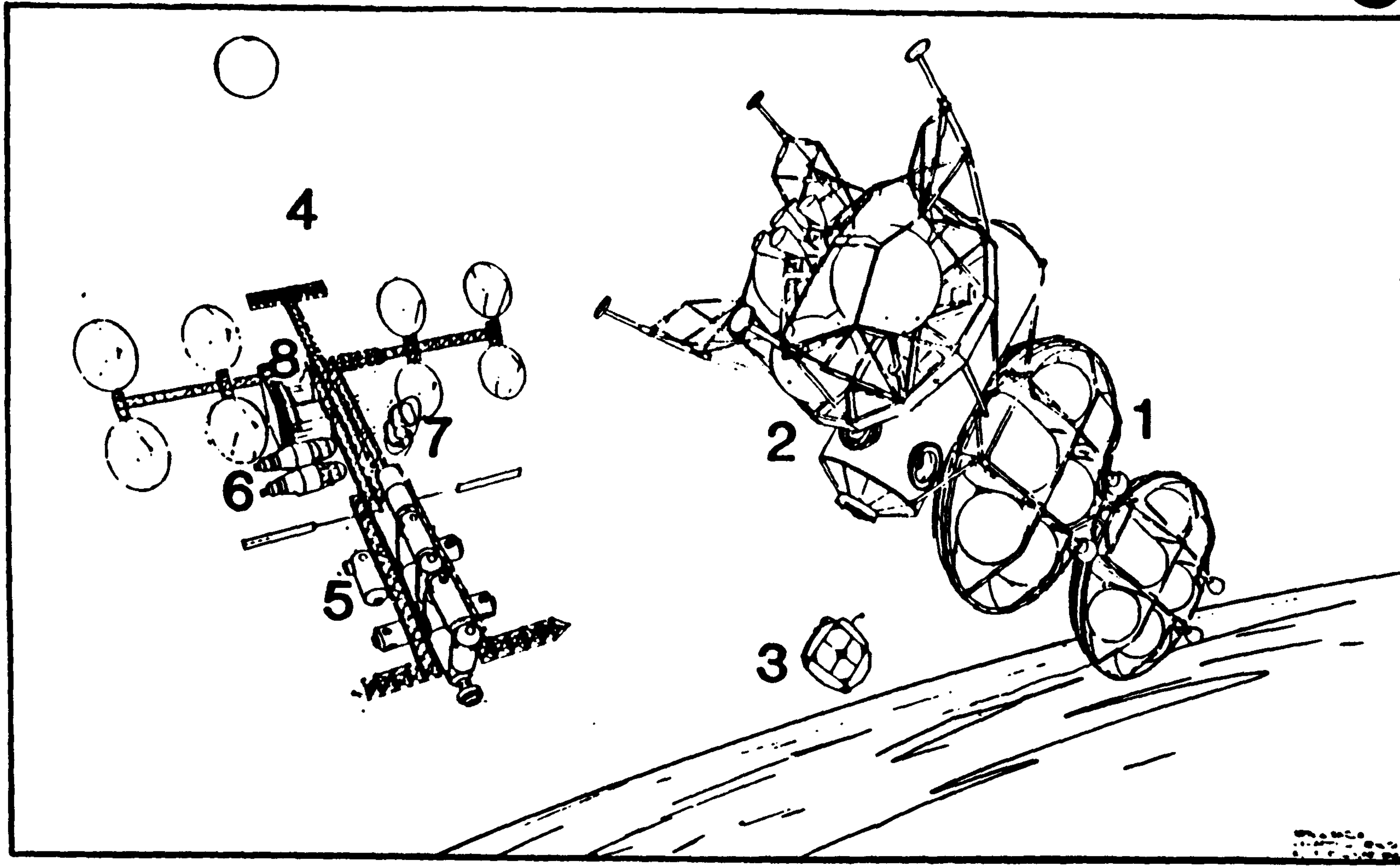
  Pulse-Mode:
  MW Thruster



Vehicle Design

Vehicle Design

- Vehicle Design Provides the "Optimum" Vehicle
- Detailed Studies are Required
- Results in Optimal
    - Vehicle Initial Masses
    - Specific Impulse
    - Power Level



AOTV Stack Departing Space Station

1. Stacked AOTVs
2. E-Lander with Common Module
3. OMV
4. Growth Space Station
5. Quarantine Module
6. OSM (Propellant Storage Modules)
7. OTV Stacking Facility
8. AOTV Hangar

Lunar Mission OTVs

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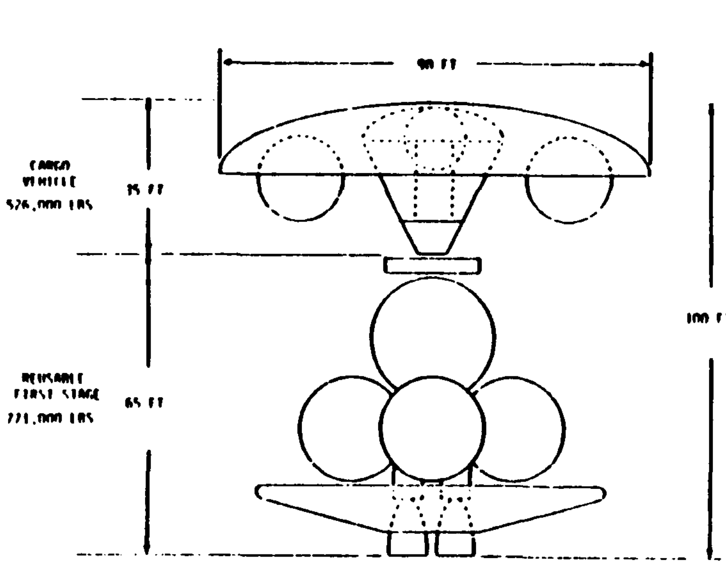
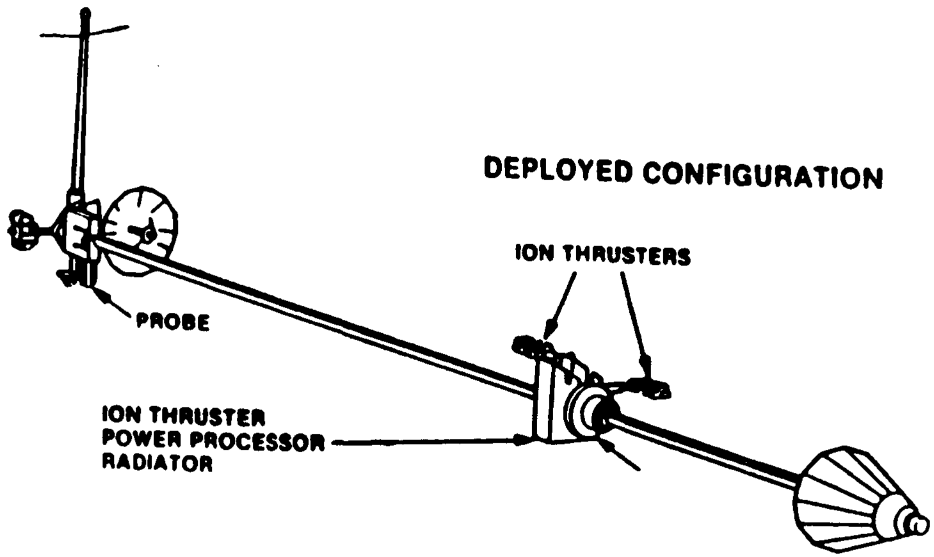
ADVANCED PROPULSION for MANNED LUNAR and MARS MISSIONS

Mars Mission Cargo Vehicles

CARGO VEHICLE STACK AT LAUNCH



DEPLOYED CONFIGURATION



o O_2/H_2 PROPULSION

o NUCLEAR-ELECTRIC PROPULSION

Vehicle Design and Trade Study Conclusions

- O_2 / H_2 OTVs with Aerobraking and a 475 lbf–s/lbm Isp are Planned for the Lunar Base Missions

- Ion and MPD OTVs with a 5000 lbf–s/lbm Isp are a Near–Optimal Design

- Power System Specific Masses Below 10 kg/kW are Required for Short Trip Time and Mass Savings Over O_2 / H_2

- Power Levels of 1–10 MW Provide a Mass Savings Over O_2 / H_2

Lunar Mission Observations

- Both Ion and MPD Enable
  Significant Propellant Mass Savings
  Over Chemical Propulsion

- An Isp of 5000 lbf-s/lbm Provides
  A Short Trip Time
  and Significant Propellant Mass Savings

- A Power Level of 1 MW Provides
  the Maximum Propellant Mass Savings

- Arcjet Propulsion Provides
  No Propellant Mass Savings
  Over Chemical Propulsion

- NEP Fleet Size Must Be Considered
  In Resupply Planning

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ADVANCED PROPULSION for
MANNED LUNAR and MARS MISSIONS

Mars Observations

- Ion and MPD Provide Significant Performance Benefit Over Chemical Propulsion

- An NEP Specific Impulse of 5000 lbf-s/lbm and a Power-and-Propulsion System Specific Mass of 10 kg/kW at a 4-10 MW Power Level is Needed For The Cargo Vehicle

- Arcjets Provide No Mass Savings Over Chemical Propulsion

- Nuclear Thermal Allows Comparable Mass Savings to NEP

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ADVANCED PROPULSION for MANNED LUNAR and MARS MISSIONS

Global Observations

- NEP is Promising But
  A MW-Class Electric Propulsion System
  and MW Space Nuclear Power
  Have Not Been Demonstrated

- If NEP is Used for the Lunar or Mars Missios,
  Much Work Must Be Done

- Nuclear-Thermal Propulsion is Promising:
  More Analysis is Warranted
    - Shielding for Manned Missions
    - Ground Testing
    - Nuclear-Thermal Aerobraking

- Hybrid Nuclear-Thermal
  and Nuclear-Electric Propulsion is Promising

Nomenclature

- Initial Mass - Mass In Earth Orbit at Launch (kg)
- Isp - Specific Impulse (lbf-s/lbm)
- kW - kilowatt
- kg - kilogram
- LEO - Low Earth Orbit
- LLO - Low Lunar Orbit
- LMO - Low Mars Orbit
- L1, L2, etc. - Libration Point
- MPD - Magneto-Plasma-Dynamic
- NEP - Nuclear-Electric Propulsion
- Payload Mass - Non-Propulsion and Non-Vehicle Related Mass (kg)
- Pj - Exhaust Jet Power (kW)
- Pe - Electrical Input Power (kW)
- Pth - Thermal Power (kW)
- Specific Mass - Mass-Scaling Factor (kg/kW)
- Xe - Xenon Propellant

- ΔV - Free-Space Velocity Change (km/s)

- 35-0 or 22.5-1 Mission or Flight - Lunar Payload Delivery Mission: 35000 kg Delivered to LLO, 0 kg Returned to LEO, etc.

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Lewis Research Center
Space Flight Systems Directorate

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ADVANCED SPACE PROPULSION WORKSHOP

APRIL 12, 1988 - ROOM 215
APRIL 13, 1988 - ROOM 225

ADMINISTRATION BUILDING

LEWIS RESEARCH CENTER

CLEVELAND, OHIO

ADVANCED SPACE ANALYSIS OFFICE

WEEK'S ACTIVITIES

- APRIL 12 AND 13, 1988 - ADVANCED SPACE PROPULSION WORKSHOP
- APRIL 14, 1988 - OFFICE OF EXPLORATION PROGRAM REVIEW
- APRIL 15, 1988 - POWER TECHNOLOGY WORKSHOP

NASA

OFFICE OF EXPLORATION - GOALS

- Provide recommendations and alternatives for a Presidential decision on a focused program of human exploration of the solar system by 1992
- Develop program opportunities and alternatives to expand exploration, human presence and activity into the solar system

OFFICE OF EXPLORATION - OBJECTIVES

- Provide an Agency focus for human exploration of space strategies to:
  - Define the Pathfinder technologies necessary to enable and support a range of future human exploration strategies
  - Identify precursor and prerequisite requirements and coordinate implementation with HQ program offices
  - Understand the science and applications potential for exploration opportunities
  - Ensure that the appropriate expertise throughout the Agency and the country is applied to the study and development of exploration plans.

PURPOSE

- DETERMINE THE STATE-OF-THE-ART OF MISSION ANALYSIS SUITABLE FOR THE LUNAR AND MARS MISSIONS
  - ASSESS THE ADEQUACY OF EXISTING STUDIES FOR DETERMINING APPLICABILITY OF CANDIDATE PROPULSION SYSTEMS
  - ASSESS THE EASE OF COMPARING RESULTS
- REVIEW
  - ADVANCED PROPULSION SYSTEMS
  - LUNAR AND MARS BASE BUILD-UP SCENARIOS

OBJECTIVE

PROVIDE AN ASSESSMENT OF STRENGTHS, WEAKNESSES, AND HOLES IN EXISTING STUDIES

- IDENTIFY WHERE FURTHER STUDIES ARE NEEDED
- CONSIDER HOW TO COORDINATE STUDIES TO GET RESULTS THAT ARE COMPARABLE

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NEXT STEP

ASAO

LeRC WILL PROVIDE A SUMMARY OF THESE PROCEEDINGS INCLUDING YOUR:

- IDENTIFICATION OF FUTURE MISSION ANALYSIS NEEDS
- RECOMMENDATIONS FOR FUTURE MEETINGS

SOME IMPORTANT PARAMETERS FOR A LOW THRUST ROUND TRIP TRAJECTORY TO MARS

JOHN P. RIEHL

LEWIS RESEARCH CENTER

APRIL 12-13, 1988

ADVANCED SPACE PROPULSION WORKSHOP

BASELINE VEHICLE
FROM A SCALED UP JPL SP-100 MISSION

- **INITIAL MASS AT DEPARTURE**
  566,028 kg. in LEO

- **DROP MASS**
  6041 kg. at Mars

- **RETURN PAYLOAD MASS**
  166,687 kg. to LEO

- **POWER SYSTEM**
  4 Mw Nuclear reactor
  Specific Mass 13.96 kg/Kw

- **PROPULSION SYSTEM**
  Isp = 4600 sec.
  Efficiency = .84
  Tankage fraction = 14.7% of propellant

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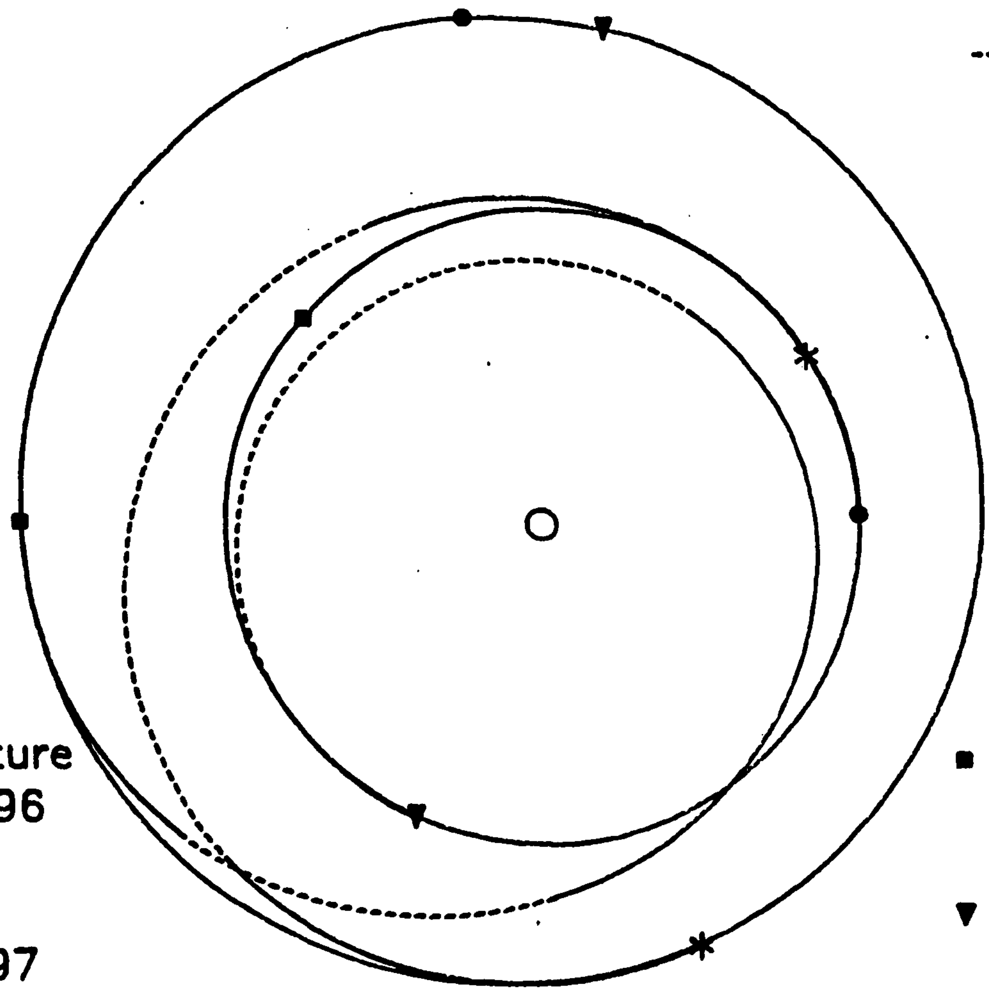
Baseline Trajectory

ASAO



ADVANCED SPACE ANALYSIS OFFICE

LOW THRUST TRAJECTORY ANALYSIS APPROACH

- Task--Discuss the quantitative impact of vehicle and mission parameters on performance

- Method--For a Mars round trip mission, perform a sensitivity analysis of initial mass in LEO to the following:

- Power Level	- Trip Time
- Specific Mass	- Stay Time
- Isp	- Drop Mass
- Efficiency	- LEO Altitude

Mass in LEO can imply the number of launches to start the mission

BASELINE MISSION
FROM A SCALED UP JPL SP-100 MISSION

- **LAUNCH**
  From 500 km. circular orbit
  September 25, 1996
  Spiral about Earth until escape reached

- **ARRIVE AT MARS**
  October 27, 1997 in 500 km. circular
  orbit after spiralling in

- **STAY TIME**
  285 Earth days at Mars
  Spiral escape from Mars

- **ARRIVE AT EARTH**
  May 28, 2000 in 500 km. circular orbit
  after spiralling in

EFFECT OF SPECIFIC IMPULSE

- Isp a measure of propulsion system efficiency
  = exhaust velocity/g
  = thrust/mass flow rate

- Higher Isp suggests better performance but there is an optimum Isp for each mission. Ergo, higher is not always better

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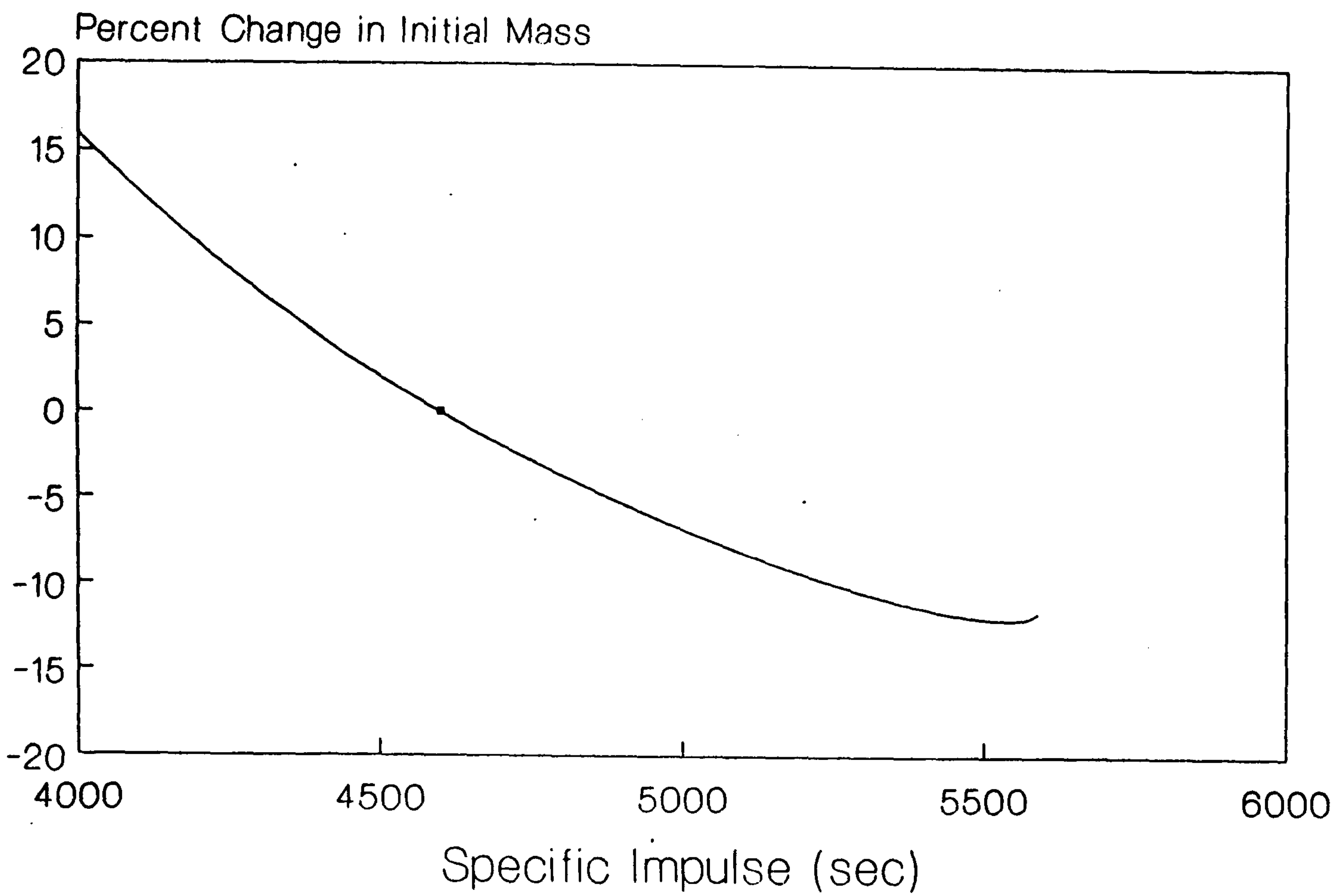
Space Flight Systems Directorate

Effect of Specific Impulse on Initial Mass

ASAO



ADVANCED SPACE ANALYSIS OFFICE

EFFECT OF INITIAL/FINAL ALTITUDE

- DEFINITION
  The altitude in circular orbit about the Earth where the mission starts and ends

- This study is not concerned with how the initial mass gets to this altitude

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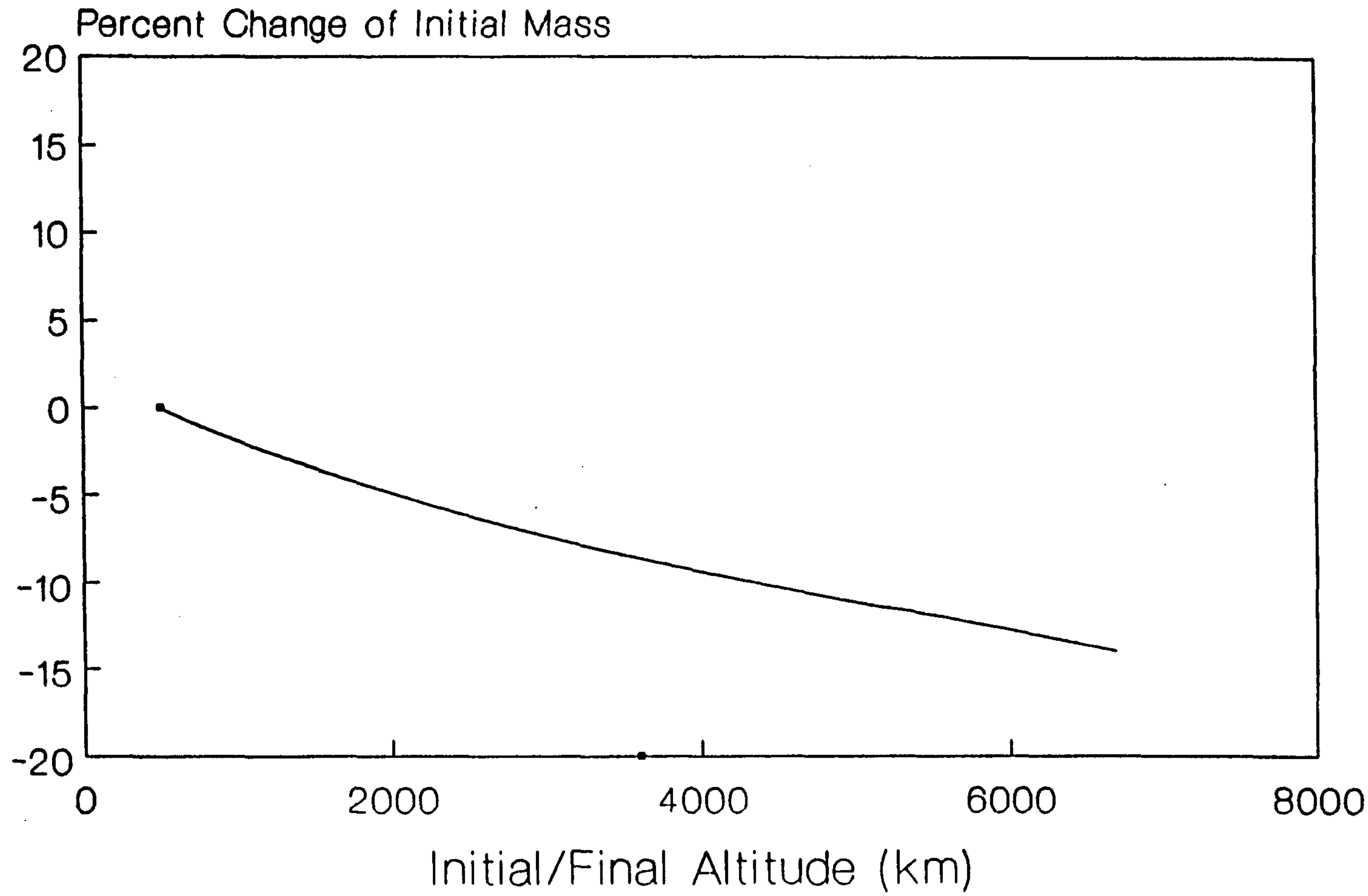
Effect of Initial/Final Altitude on Initial Mass



ADVANCED SPACE ANALYSIS OFFICE

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EFFECT OF MARTIAN ALTITUDE

ASAO

- **DEFINITION**
  The circular altitude about Mars at which the mission arrives and departs

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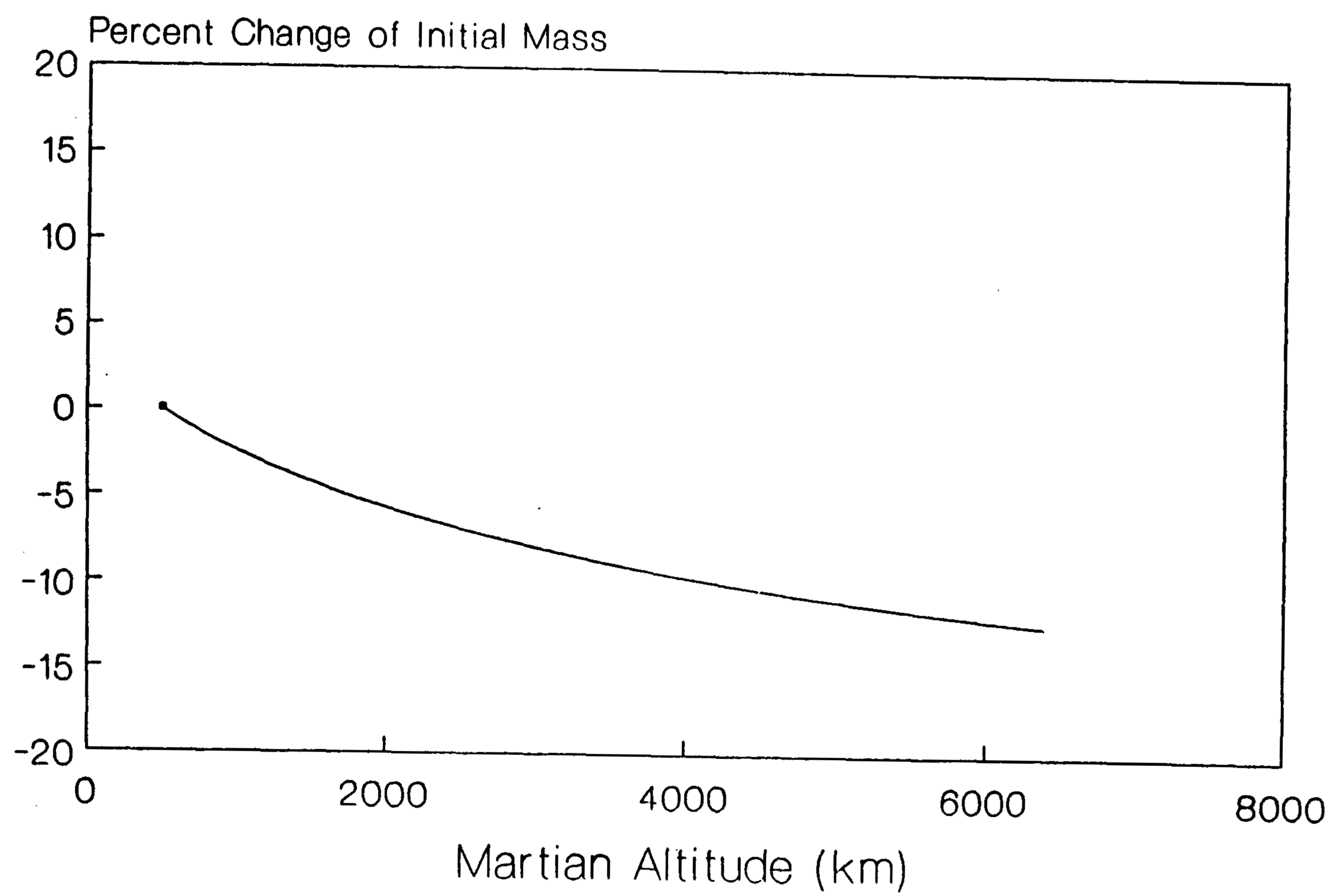
Effect of Martian Altitude on Initial Mass

ASAO



EFFECT OF DROP MASS

- DEFINITION
  The amount of mass left at Mars at conclusion of Martian portion
- To drop off more, it is necessary to depart with more in order to return with the same payload mass

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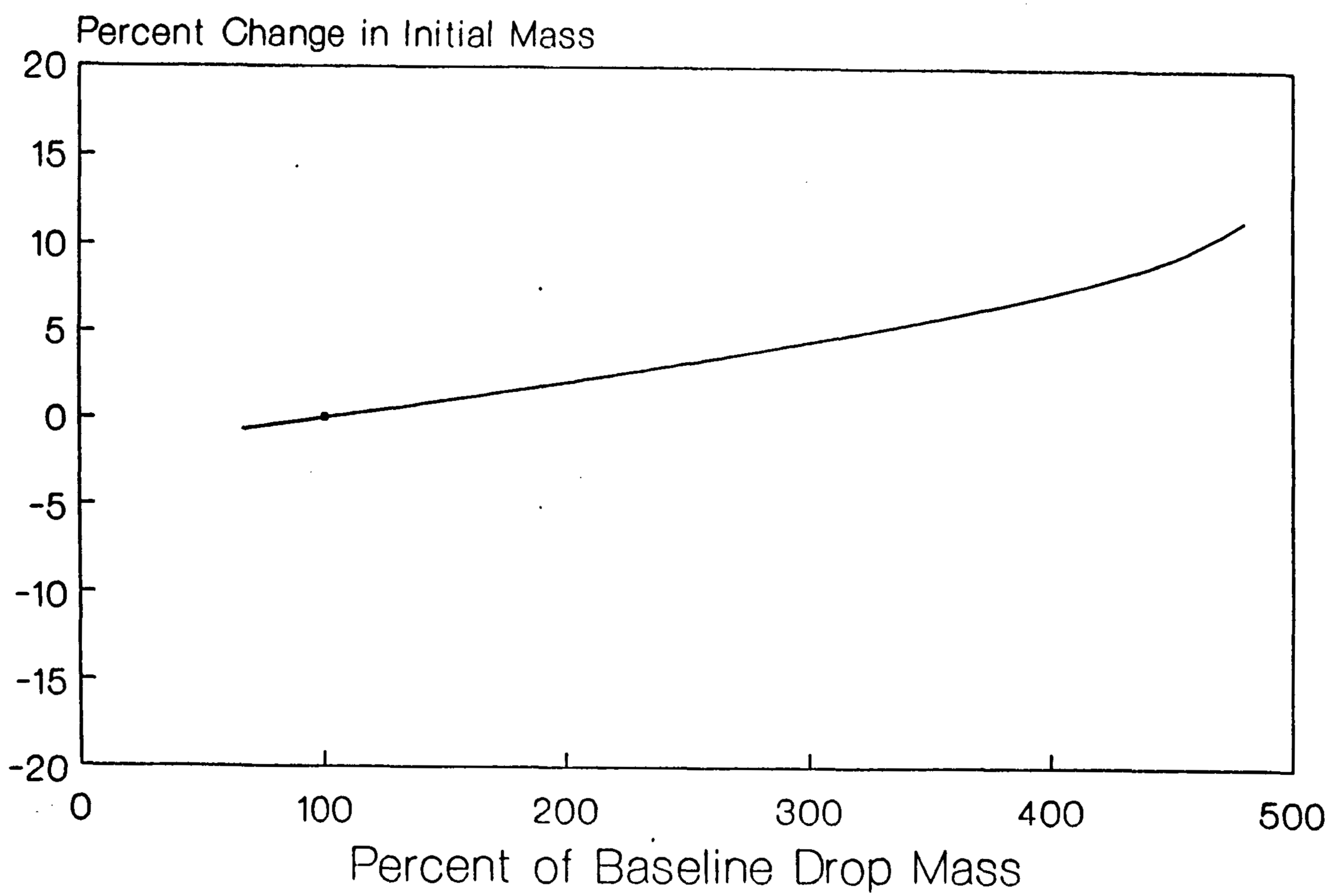
Space Flight Systems Directorate

Effect of Drop Mass on Initial Mass

ASAO



ADVANCED SPACE ANALYSIS OFFICE

EFFECT OF STAY TIME

- DEFINITION
  The amount of the total trip time actually spent at Mars excluding spiral in and out times

- Changing stay time subtly changes the geometry of the problem

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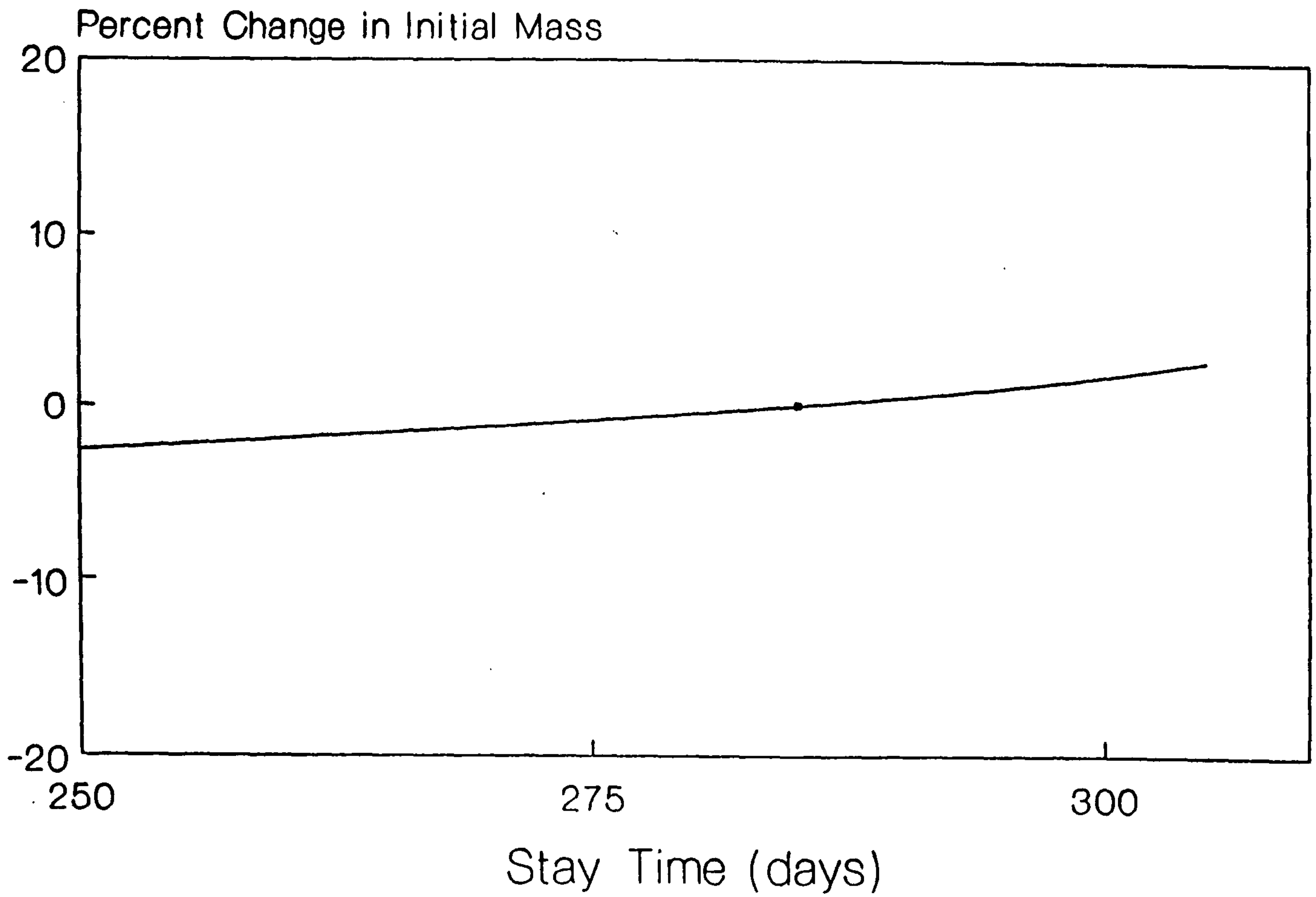
Effect of Stay Time at Mars on Initial Mass

ASAO



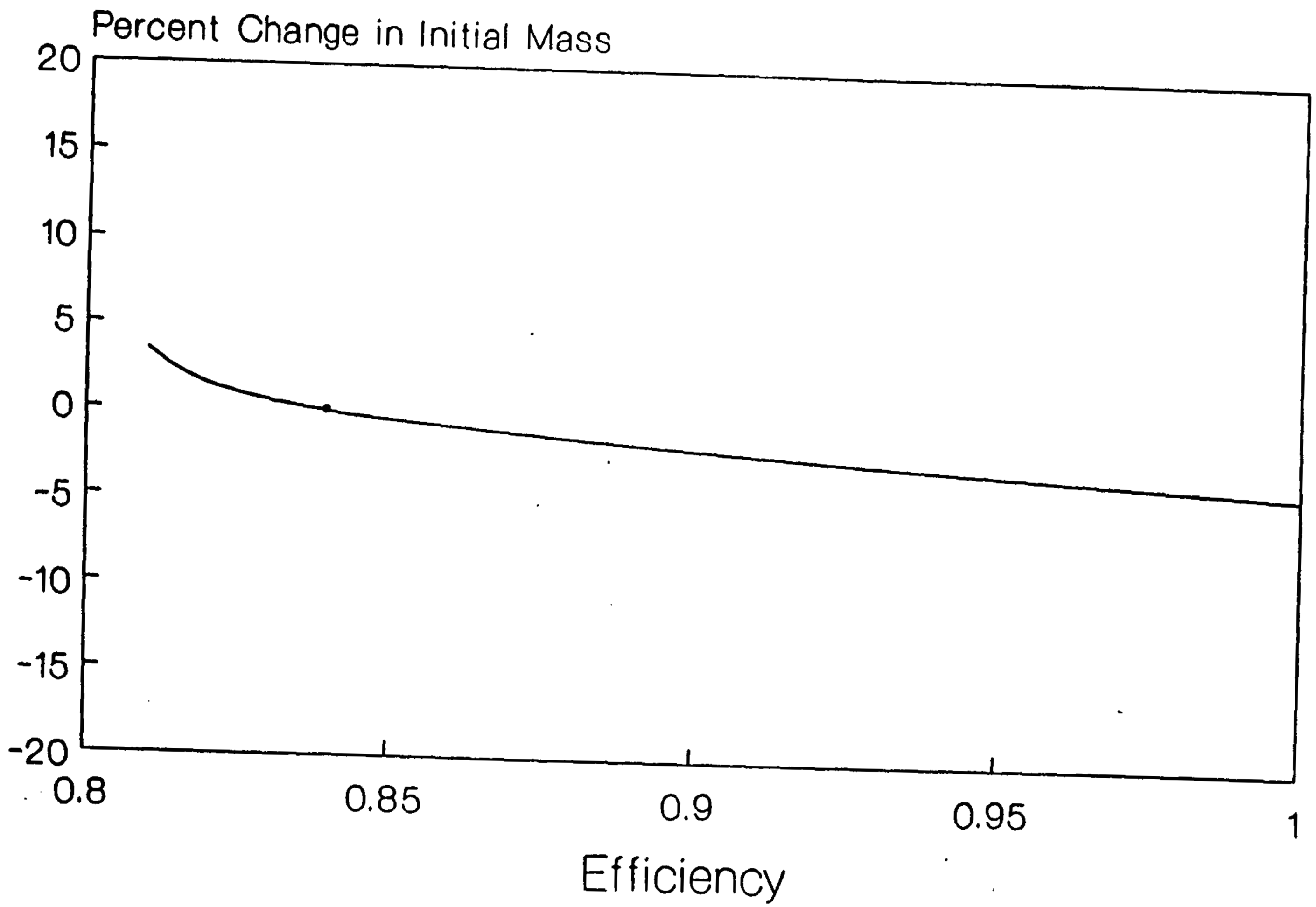
EFFECT OF THRUSTER EFFICIENCY

- DEFINITION

$$\text{eta} = \frac{\text{thruster beam power}}{\text{power input into thruster}}$$

- eta's less than .81 would not converge.
  Preliminary results suggest eta = .5 would incur significant initial mass increase

Effect of Efficiency on Initial Mass



EFFECT OF POWER

- DEFINITION
  Power output of the power system. It is usually a constant. It is assumed that any power conversion loses occur prior to the availability of the power to the thruster subsystem

- Power affects the initial mass of the reactor system and the amount of thrust available to the vehicle because of:
  1) reactor mass = specific mass* power
  2) thrust = 2* power/(exhaust velocity)

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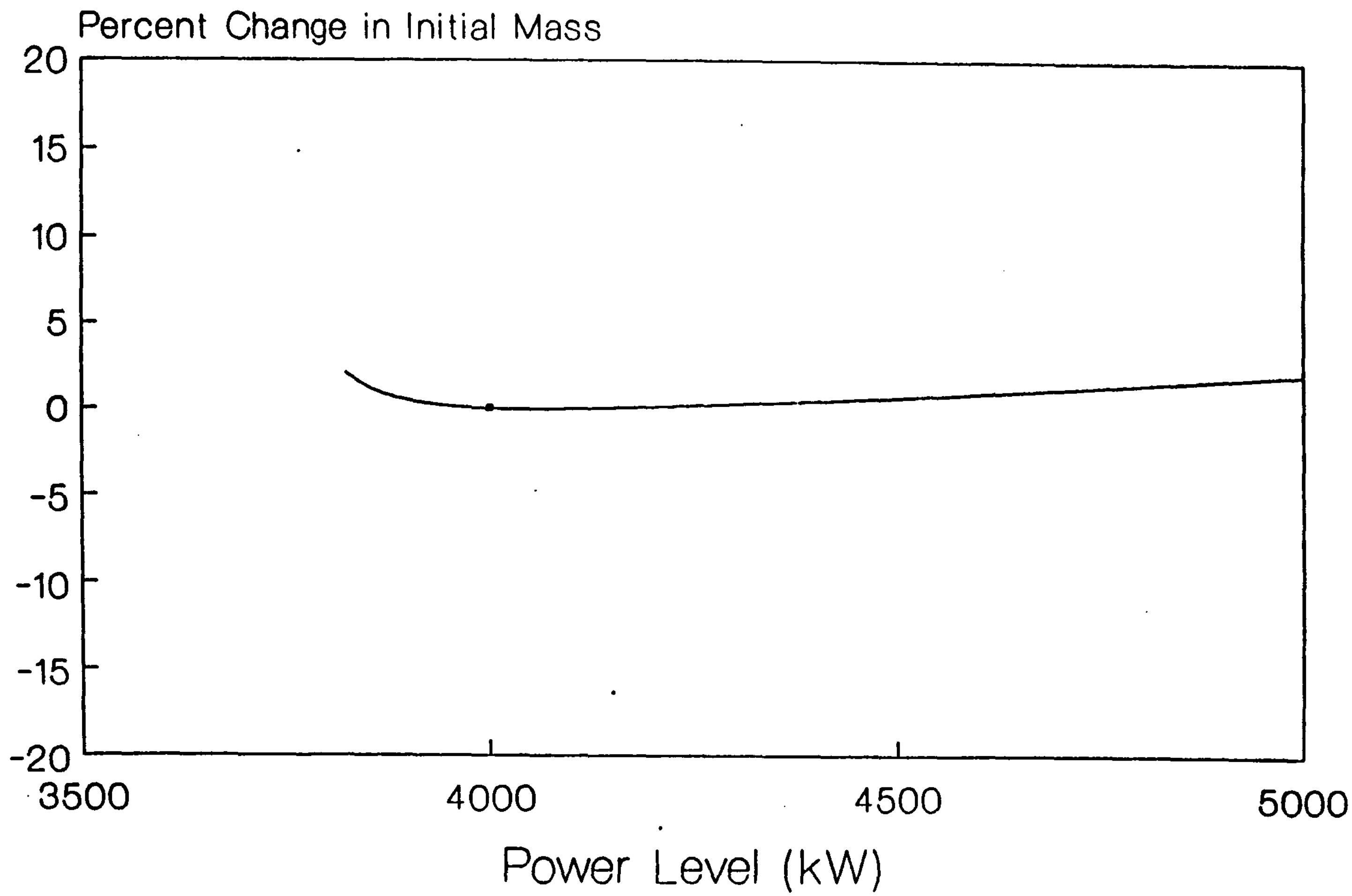
Effect of Power on Initial Mass

ASAO



ADVANCED SPACE ANALYSIS OFFICE

EFFECT OF TOTAL TRIP TIME

- **DEFINITION**
  Total Trip time is the number of days from Earth departure to Earth arrival including spiral times

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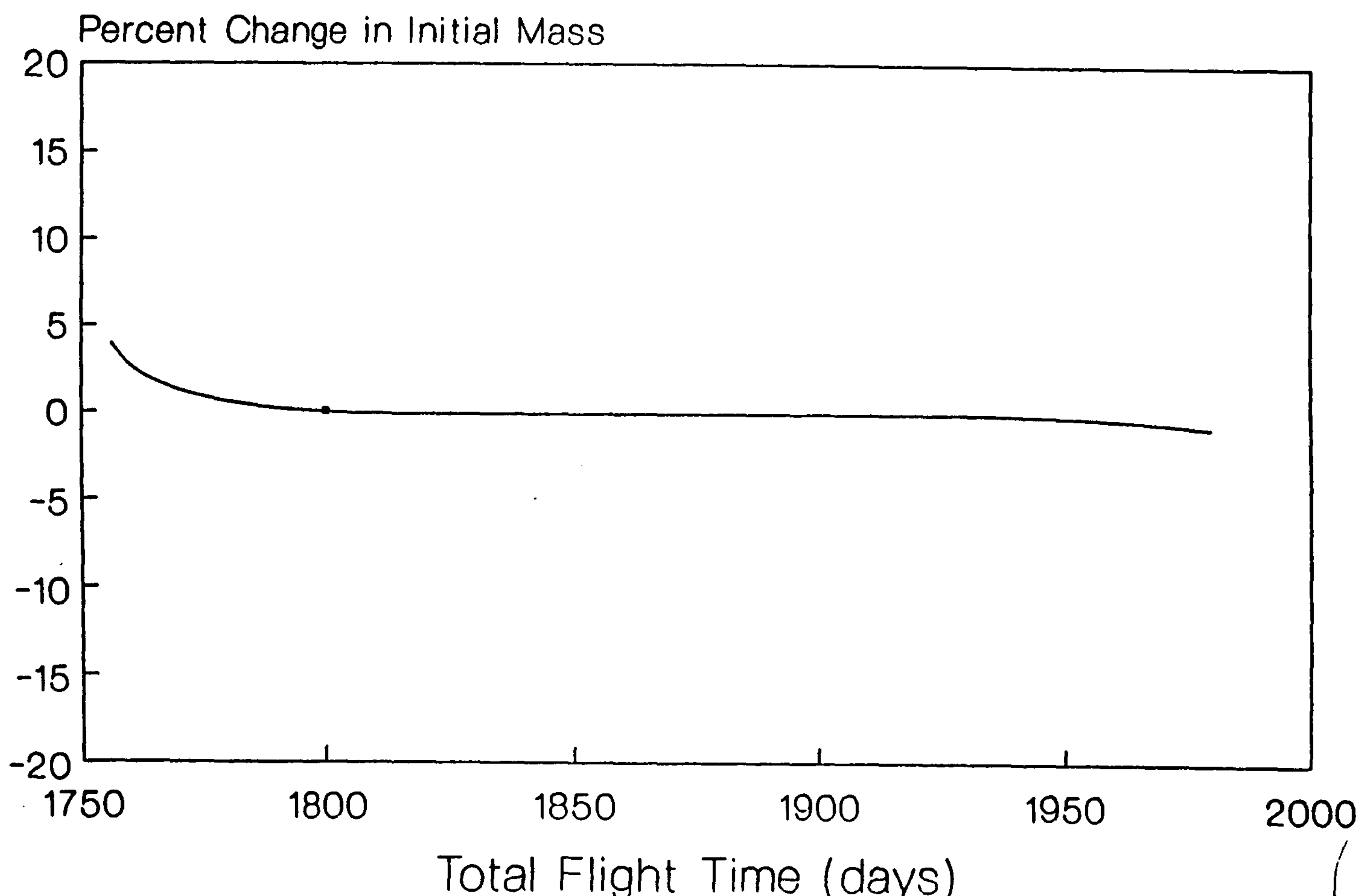
Effect of Total Trip Time on Initial Mass

ASAO



ADVANCED SPACE ANALYSIS OFFICE

OBSERVATIONS FOR THIS MISSION

- The data suggests higher Isp and lower specific mass yield significantly less initial mass in LEO
- The data suggests drop mass and both LEO and Martian altitude have some impact on reducing initial mass
- The data suggests trip time, stay time, thruster efficiency, and power level have little effect on initial mass

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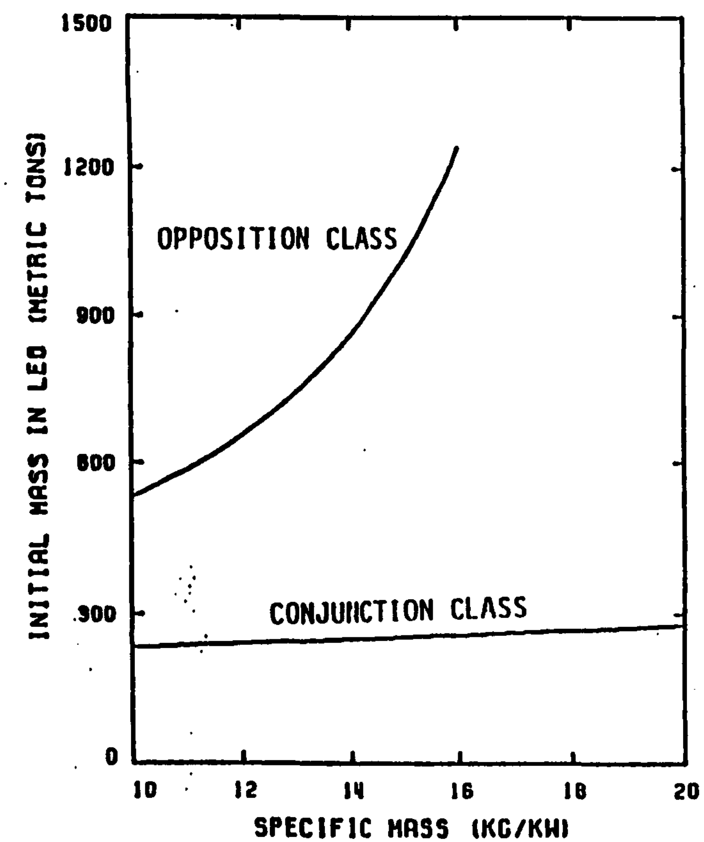
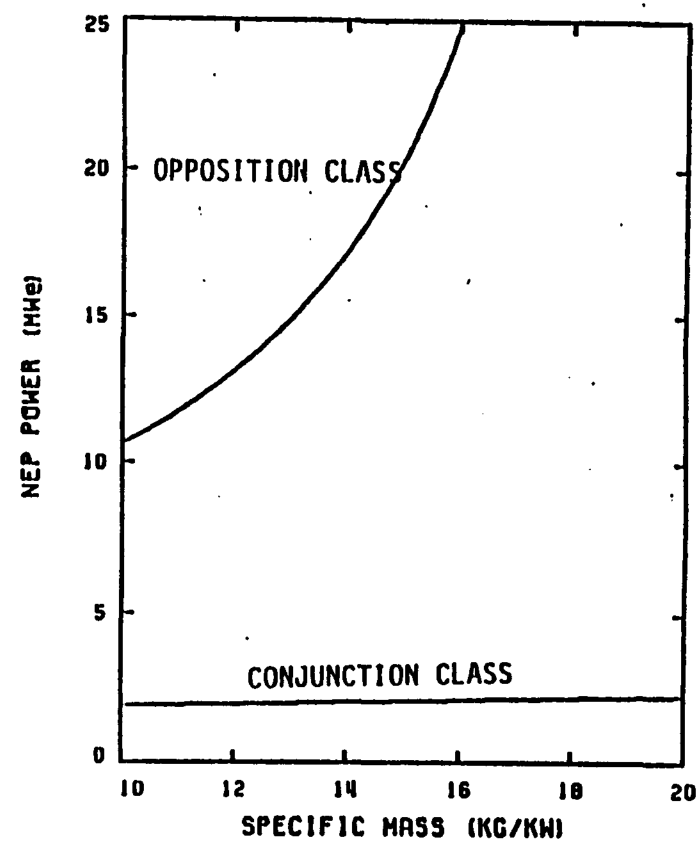
Performance Sensitivity to NEP Specific Mass

ALAN FRIEDLANDER
SAIC SPACE SCIENCES

Ion Thrusters, Isp = 6000 sec





CONJUNCTION WITH 7 MONTH STAYTIME AT MARS
OPPOSITION WITH 2 MONTH STAYTIME AT MARS, NO AEROBRAKING

ADVANCED SPACE ANALYSIS OFFICE

PROPULSION BY SOLAR SAIL

CAPABILITY

- 1 KM^2 AL-MYLAR SAIL GIVES 8-1O N PHOTON FORCE AT 1 A.U.
- IF SAIL WEIGHS 5 GM/M^2, TOTAL MASS IS 5000 KG
- IF PAYLOAD MASS IS 5000 KG, TOTAL MASS IS 10,000 KG
- THEN, ACCELERATION = F/M = 10/10,000 = 10^{-3} M/S^2
- SPECIFIC IMPULSE IS INFINITE (NO PROPELLANT)
- TACKING WILL CHANGE FORCE DIRECTION (BUT STILL OUTWARD)

EXAMPLE MISSION

- 250 KG MERCURY SAMPLE RETURN IN 500 DAYS

TECHNOLOGY DEVELOPMENT/PROBLEMS

- STRUCTURAL DESIGN
- CONTROL/NAVIGATION
- VIBRATION
- STATIC CHARGE
- TEMPERATURE
- DEPLOYMENT
- SAIL LIFETIME
- LONG DEVELOPMENT TIME (NEED SPACE DEMOS)
- DECREASED THRUST BEYOND 1 A.U.
- COST

PAUL PENZO

LUNAR AND MARS EXPLORATION

VEHICLE SIZING USING ALL PROPULSION COMPARED WITH AEROBRAKING

MAIN ENGINE THRUST LEVEL SENSITIVITY

PTS 824-0533

MSFC/A. YOUNG
M. HERRMANN
J. HAYS
N. BROWN
R. CHAMPION
G. HAJOS

1-961-9-3T

MARS EXPLORATIONS
STUDY ASSUMPTIONS

TIME PERIOD OF CONSIDERATION: 2003 OPPOSITION (AUG 2002 LAUNCH)

PLANET DEPARTURE AND CAPTURE ORBIT PARAMETERS

EARTH DEPARTURE	CIRCULAR ORBIT ALTITUDE = 270 N.MI.
MARS CAPTURE	24 HR ELLIPTIC ORBIT PERIAPSIS ALTITUDE = 270 N.MI.
MARS ESCAPE	24 HR ELLIPTIC ORBIT PERIAPSIS ALTITUDE = 270 N.MI.
EARTH CAPTURE	24 HR ELLIPTIC ORBIT PERIAPSIS ALTITUDE = 270 N.MI.

HELIOCENTRIC PROFILE

OUTBOUND VENUS SWINGBY MODE
VENUS MINIMUM CLOSEST APPROACH EQUAL 0.1 PLANET RADII (330 N. MI.)

INTERPLANETARY SPACE VEHICLE

			OPPOSITION MISSION (LBS)
SPACECRAFT:	MISSION MODULE WEIGHT	=	135,000
	MARS EXCURSION MODULE WEIGHT	=	133,000 (gross wt
	PROBES WEIGHT	=	25,000
	EARTH RETURN CAPSULE	=	15,000
			308,000

PROPULSION STAGES	FIRST STAGE	SECOND STAGE	THIRD STAGE	
MASS FRACTION (λ)	S. EQ.	S. EQ.	S. EQ.	
Isp (SEC)	480	480	480	LOX/LH_2 CHEMICAL
Isp (SEC)	836	836	836	NUCLEAR THERMAL H_2

J. A. Montgomery

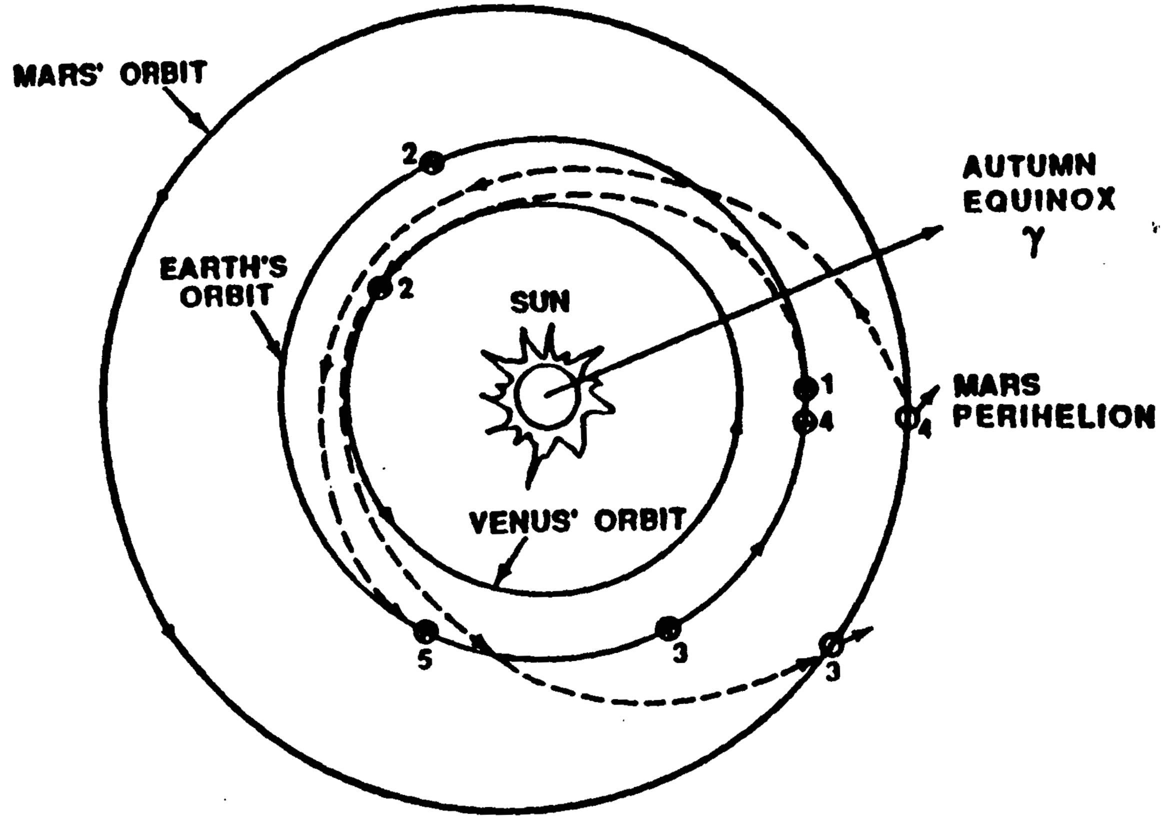
ORIGINAL PAGE IS
OF POOR QUALITY

-962-8-3

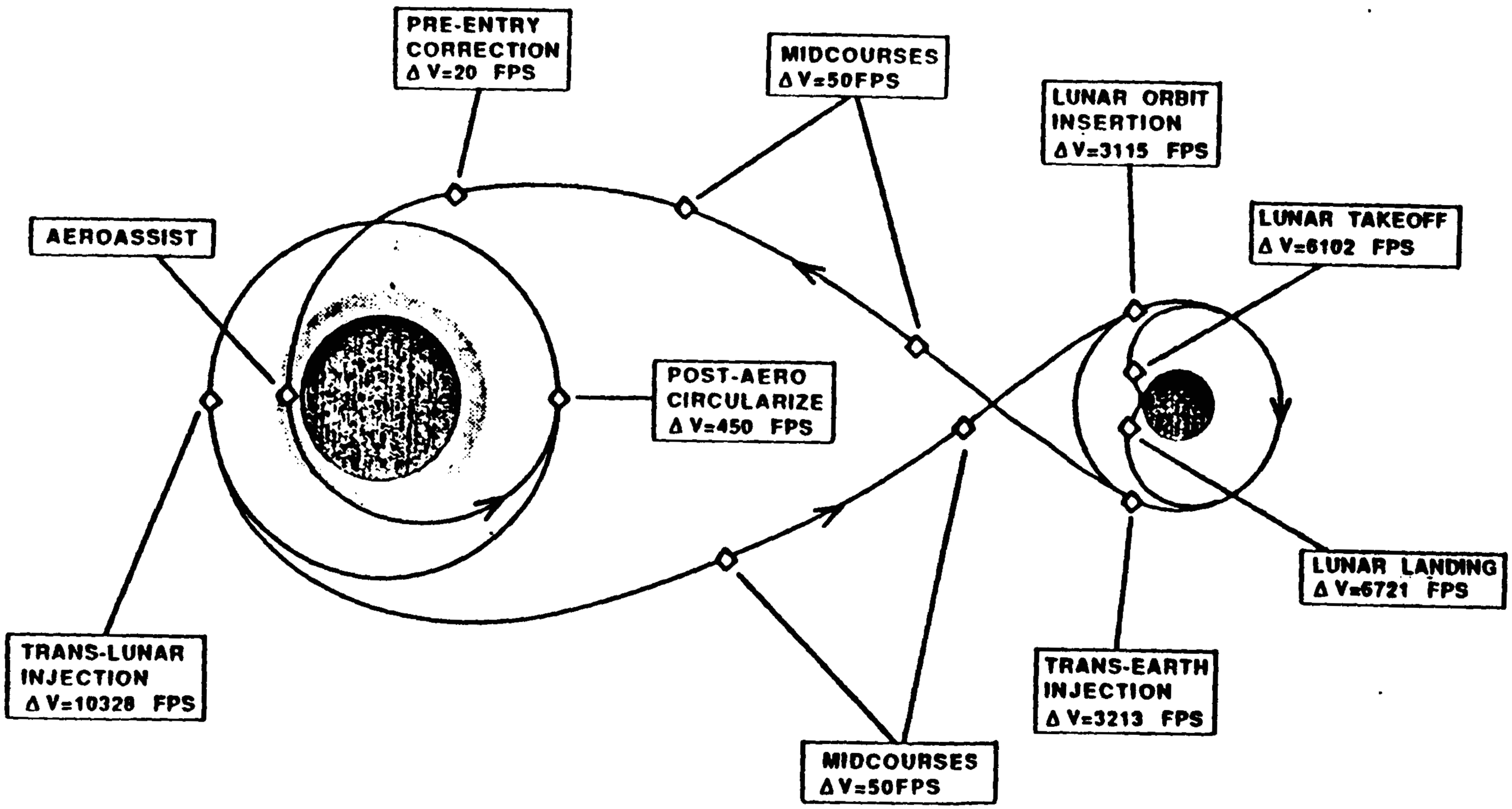
MARS EXPLORATION
MISSION PROFILE FOR 2003 OPPOSITION

OUTBOUND VENUS SWINGBY 2003 OPPOSITION

1. EARTH DEPARTURE, AUG. 29, 2002
2. VENUS PASSAGE, DEC. 26, 2002
3. MARS ARRIVAL, JUN. 25, 2003
4. MARS DEPARTURE, AUG. 24, 2003
5. EARTH ARRIVAL, APR. 28, 2004



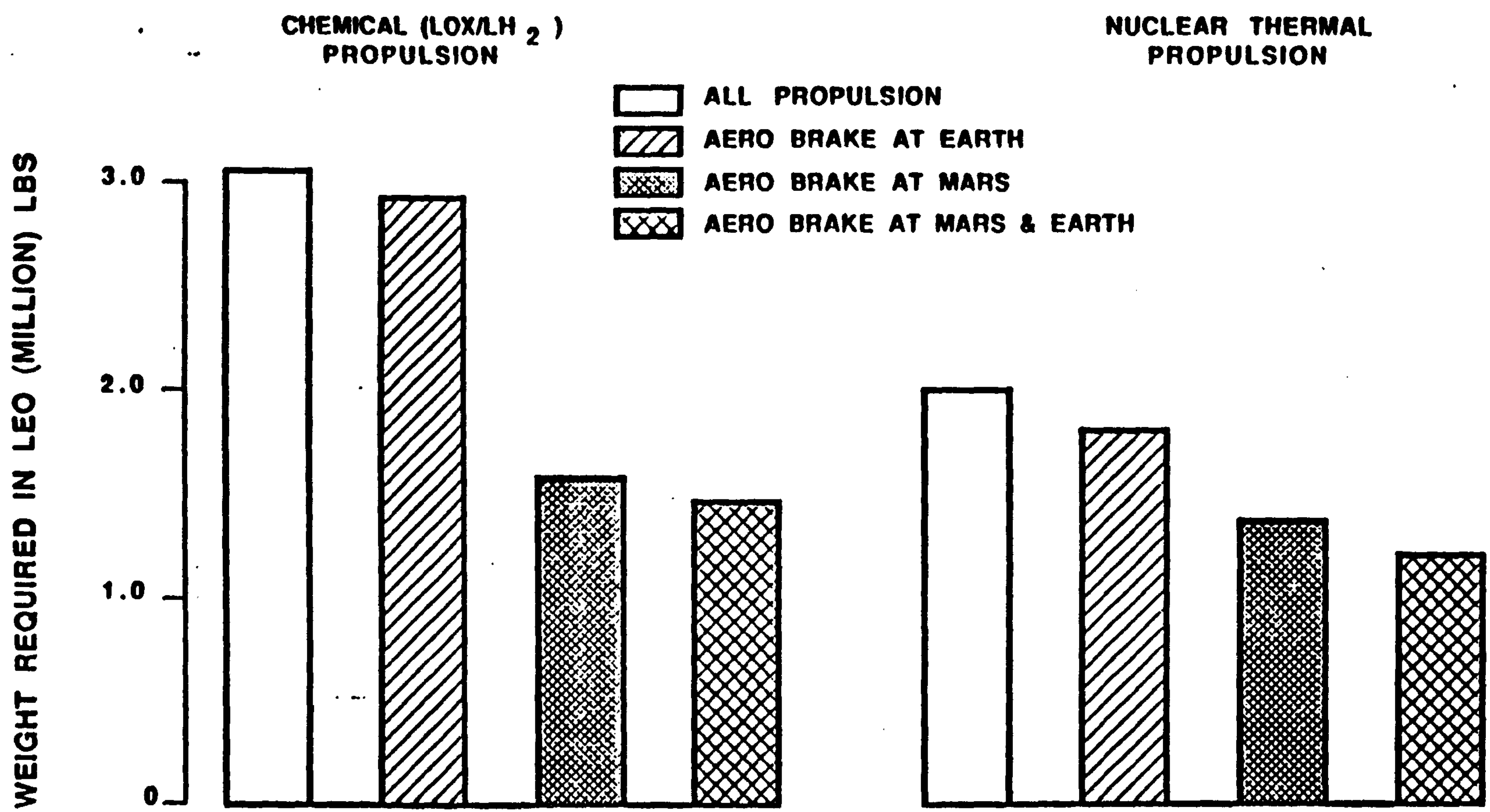
LUNAR PROFILE – LUNAR ORBIT



Martin Marietta?

1-939-8-3D

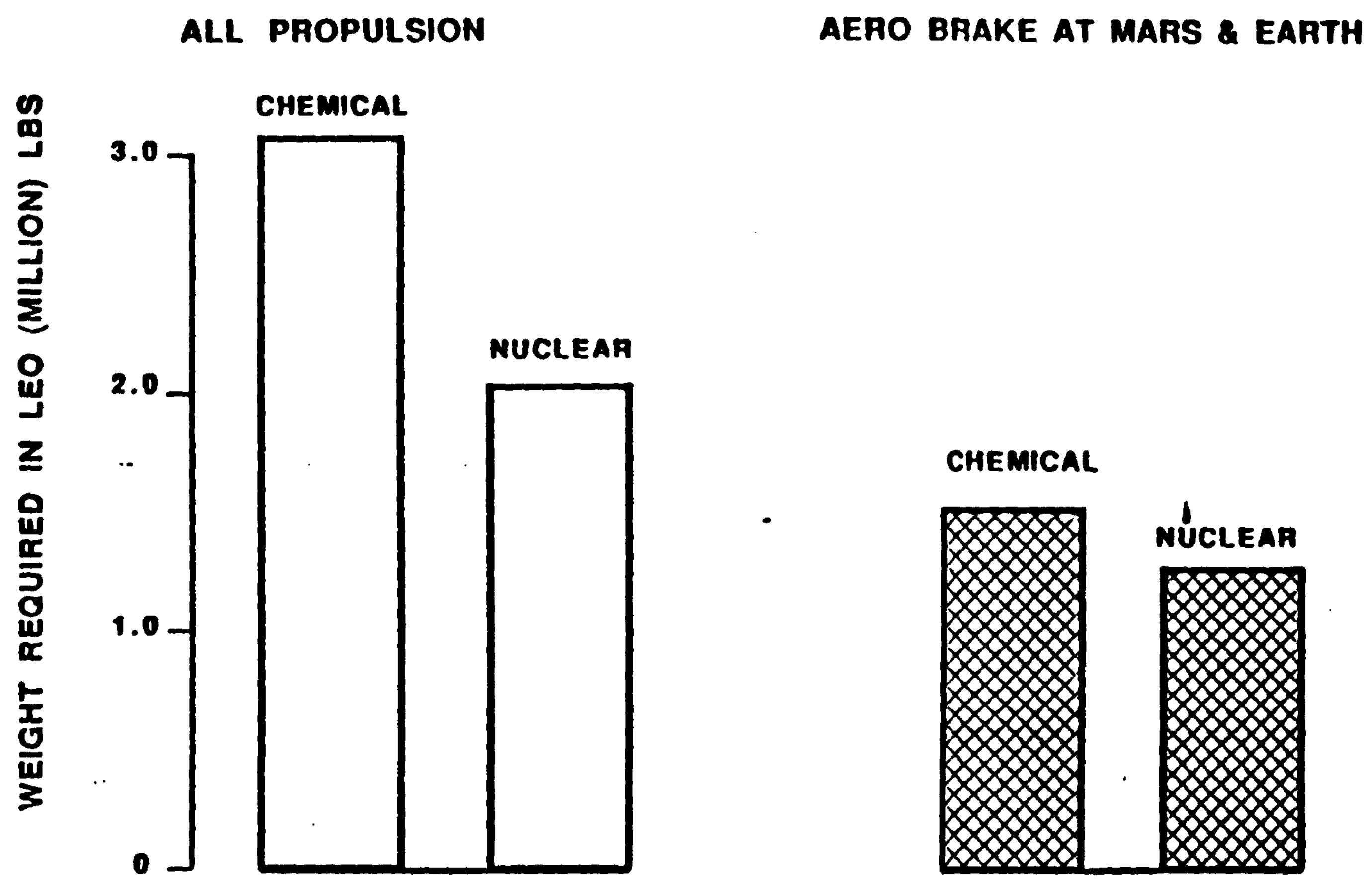
MARS EXPLORATION
ALL PROPULSION MANEUVERS COMPARED TO AERO BRAKE
2003 OPPOSITION MISSION
WITH VENUS OUTBOUND SWING BY
608 DAY MISSION TIME



MSFC/A. YOUNG

1-940-8-3D

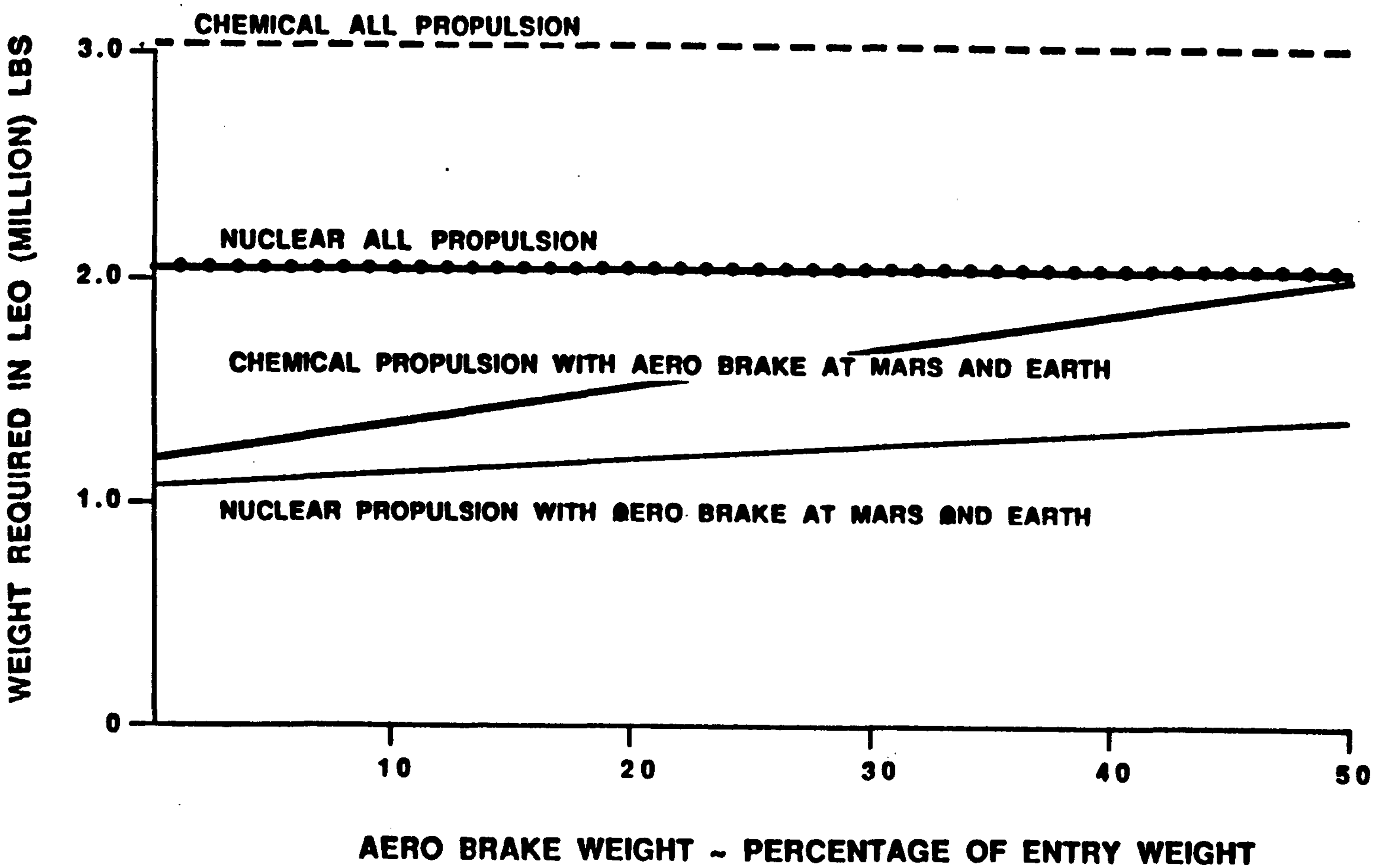
MARS EXPLORATION
ALL PROPULSION MANEUVERS COMPARED TO AERO BRAKE
2003 OPPOSITION MISSION
WITH VENUS OUTBOUND SWING BY



MSFC/A. YOUNG

1-938-8-3D

MARS EXPLORATION
WEIGHT REQUIRED IN LEO
2003 OUTBOUND VENUS SWINGBY OPPORTUNITY

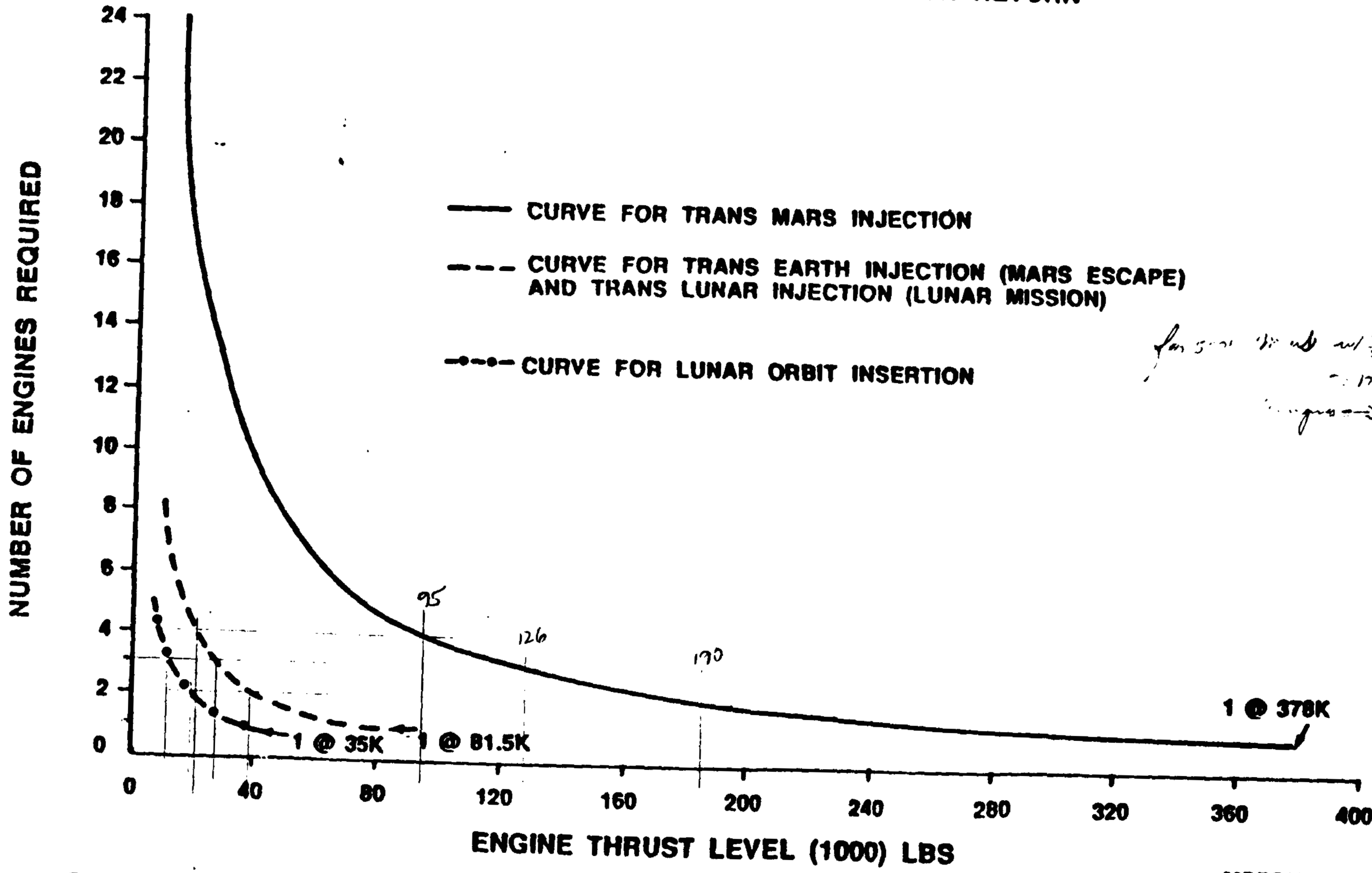


MSFC/A. YOUNG

1-956-88

LUNAR AND MARS EXPLORATION LOX/LH_2 PROPULSION NUMBER OF ENGINES REQUIRED INITIAL THRUST TO WEIGHT RATIO IS 0.25

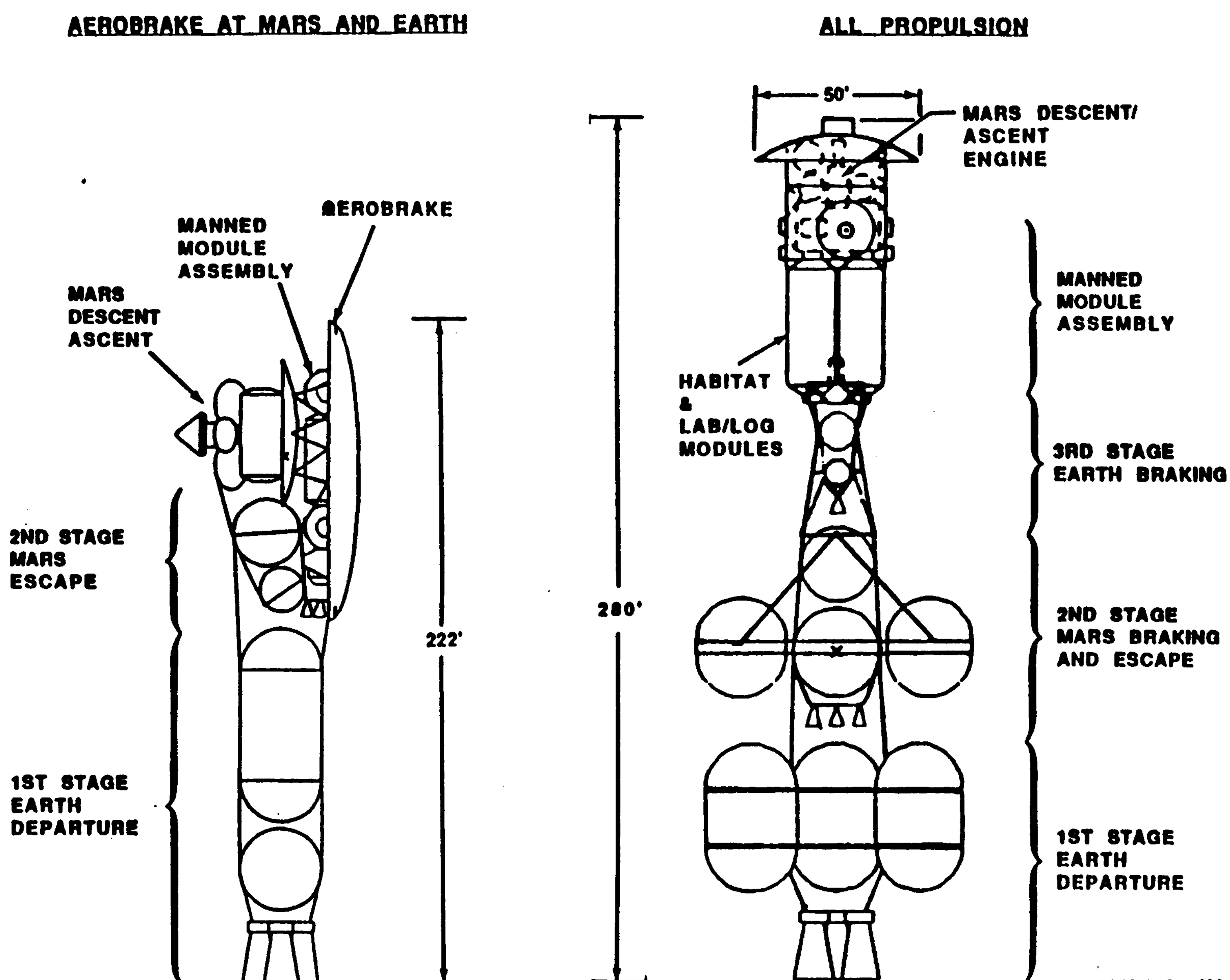
MARS MISSIONS PROFILE WITH OUTBOUND VENUS SWINGBY AEROBRAKE AT MARS AND EARTH RETURN LUNAR PROFILE AEROBRAKE AT EARTH RETURN



MSFC/A. YOUNG

1-1016-8

CONFIGURATIONS AT EARTH ORBIT DEPARTURE

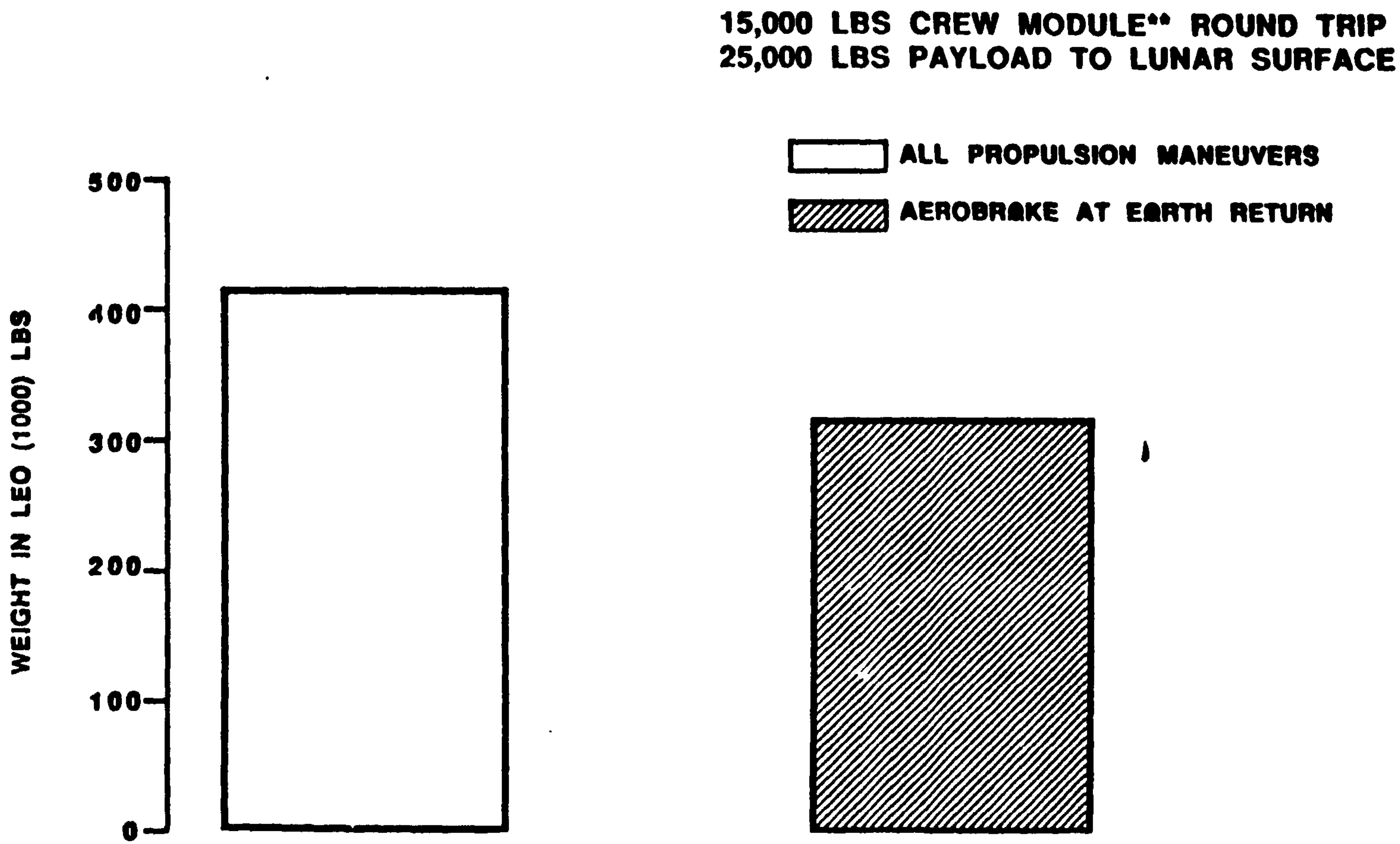


MSFC/G. HAJOS

1-941-8-3D

LUNAR EXPLORATION

ALL PROPULSION MANEUVERS COMPARED TO AERO BRAKE
TWO STAGE CONFIGURATION
CHEMICAL (LOX/LH_2) PROPULSION*

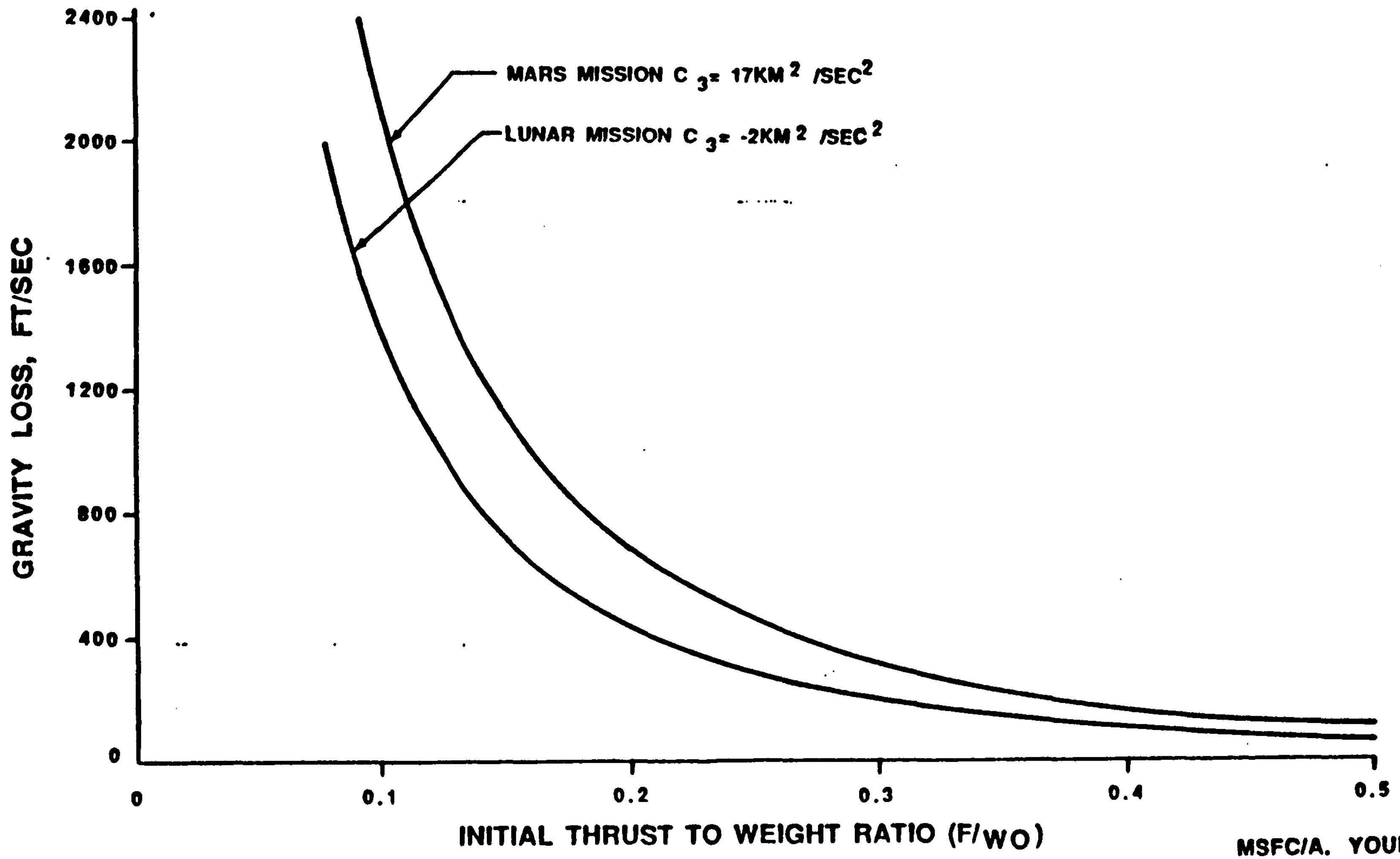


*ONE STAGE DESCENT/ASCENT STAGE POSITIONED IN LUNAR ORBIT
**CREW MODULE GOES TO LUNAR SURFACE AND RETURNS TO LEO

MSFC/A. YOUNG

1-954-88

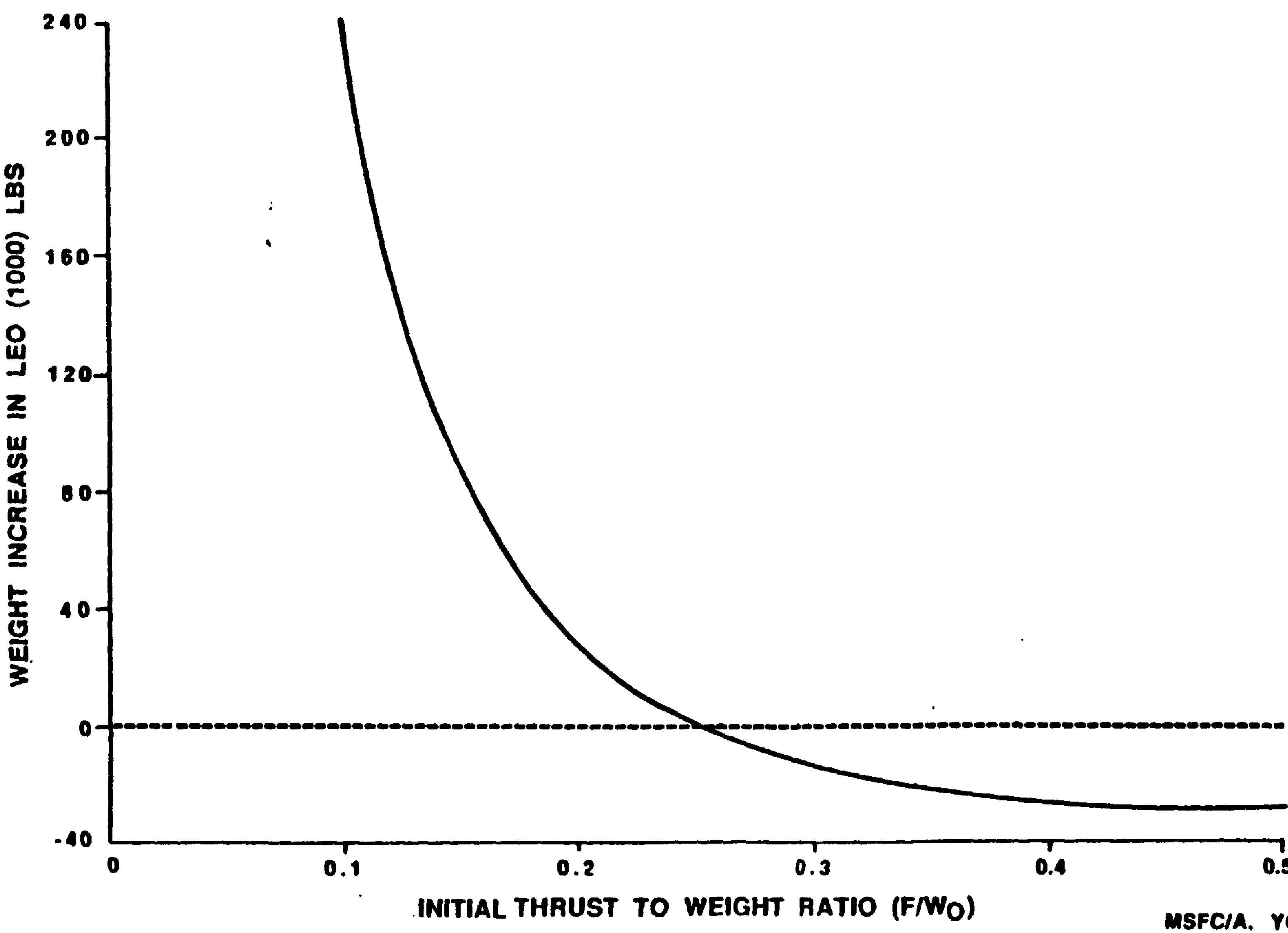
LUNAR AND MARS EXPLORATION
SINGLE CRYOGENIC PROPULSION STAGE
GRAVITY LOSS FOR EARTH ESCAPE (SINGLE BURN)



MSFC/A. YOUNG

1-955-88

MARS EXPLORATION
WEIGHT INCREASE IN LEO ABOVE THAT OF $F/W_O = 0.25$
OUTBOUND VENUS SWINGBY 2003 OPPOSITION
CHEMICAL (LOX/LH_2 /PROPULSION WITH AEROBRAKE AT MARS AND EARTH



MSFC/A. YOUNG

LUNAR AND MARS EXPLORATION
VEHICLE SIZING AND MAIN ENGINE THRUST LEVEL SENSITIVITY

SUMMARY REMARKS

- AEROBRAKING AT MARS AND EARTH CAN REDUCE THE WEIGHT REQUIRED IN LEO BY 50% COMPARED TO CHEMICAL ALL PROPULSION
- AEROBRAKING WITH CHEMICAL PROPULSION REQUIRES LESS WEIGHT IN LEO THAN NUCLEAR THERMAL ALL PROPULSION
- AEROBRAKING AT EARTH FOR LUNAR MISSION REQUIRES 25% LESS WEIGHT IN LEO COMPARED TO CHEMICAL ALL PROPULSION
- AN EFFECTIVE THRUST TO WEIGHT RATIO FOR MAIN ENGINES IS $\approx$ 0.25
- THRUST LEVEL RANGE FOR TRANS-MARS INJECTION IS 75,000 TO 125,000 LBS
- THRUST LEVEL RANGE FOR TRANS-EARTH INJECTION (MARS ESCAPE) AND TRANS-LUNAR INJECTION IS 20,000 TO 50,000 LBS

MSFC/A. YOUNG

1-957-8-3T

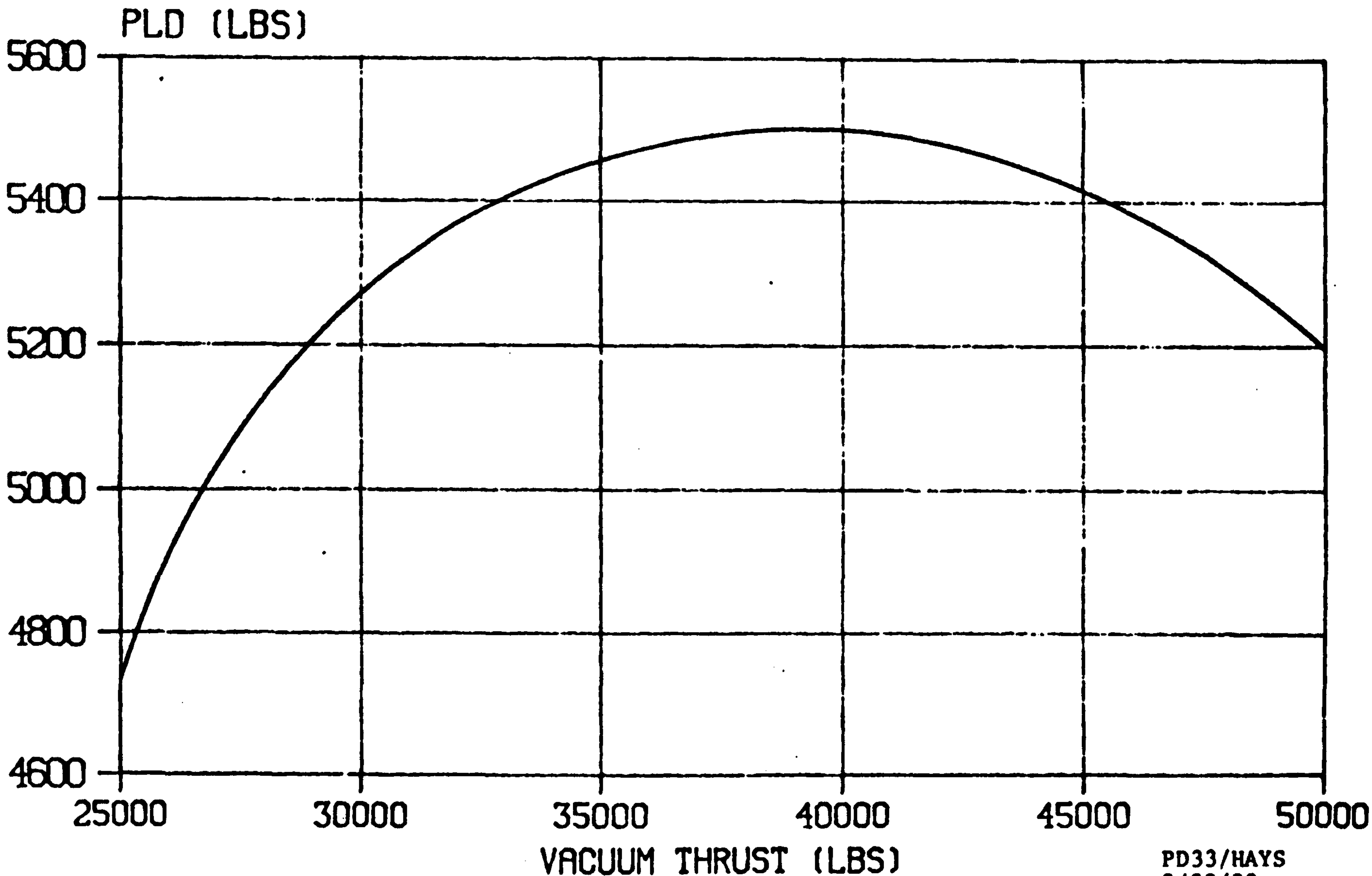
LUNAR AND MARS EXPLORATION
ASCENT AND DESCENT PROPULSION

- EFFECT OF THRUST LEVEL
- EFFECT OF Isp

MSFC/A. YOUNG

MARS ASCENT--GLOW=49,062 LBS
270 X 18122 NMI/74.73 DEG

PLD VS FVAC

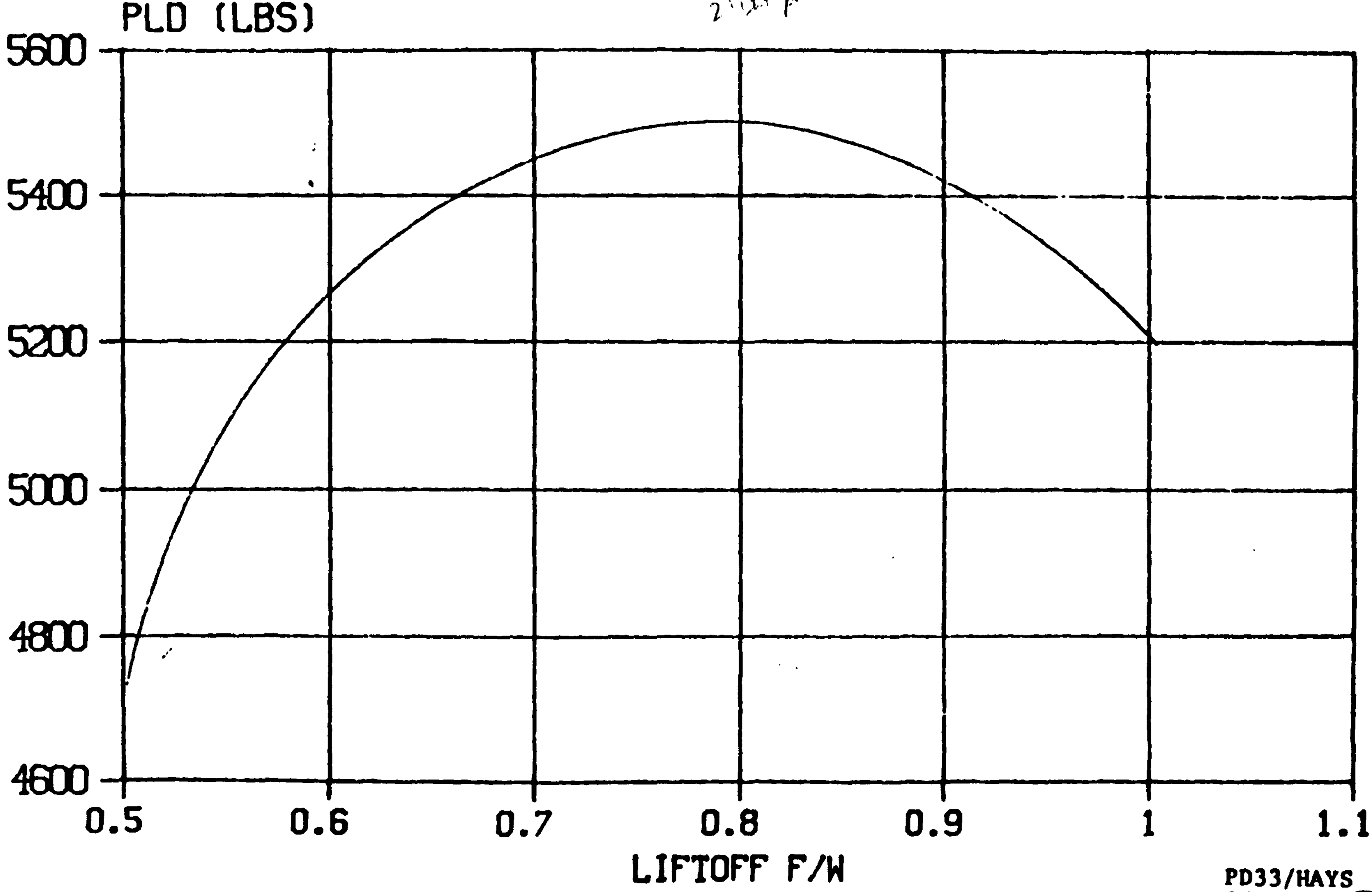


PD33/HAYS
2/29/88

MARS ASCENT--GLOW=49,062 LBS
270 X 18122 NMI/74.73 DEG

mars synchronous

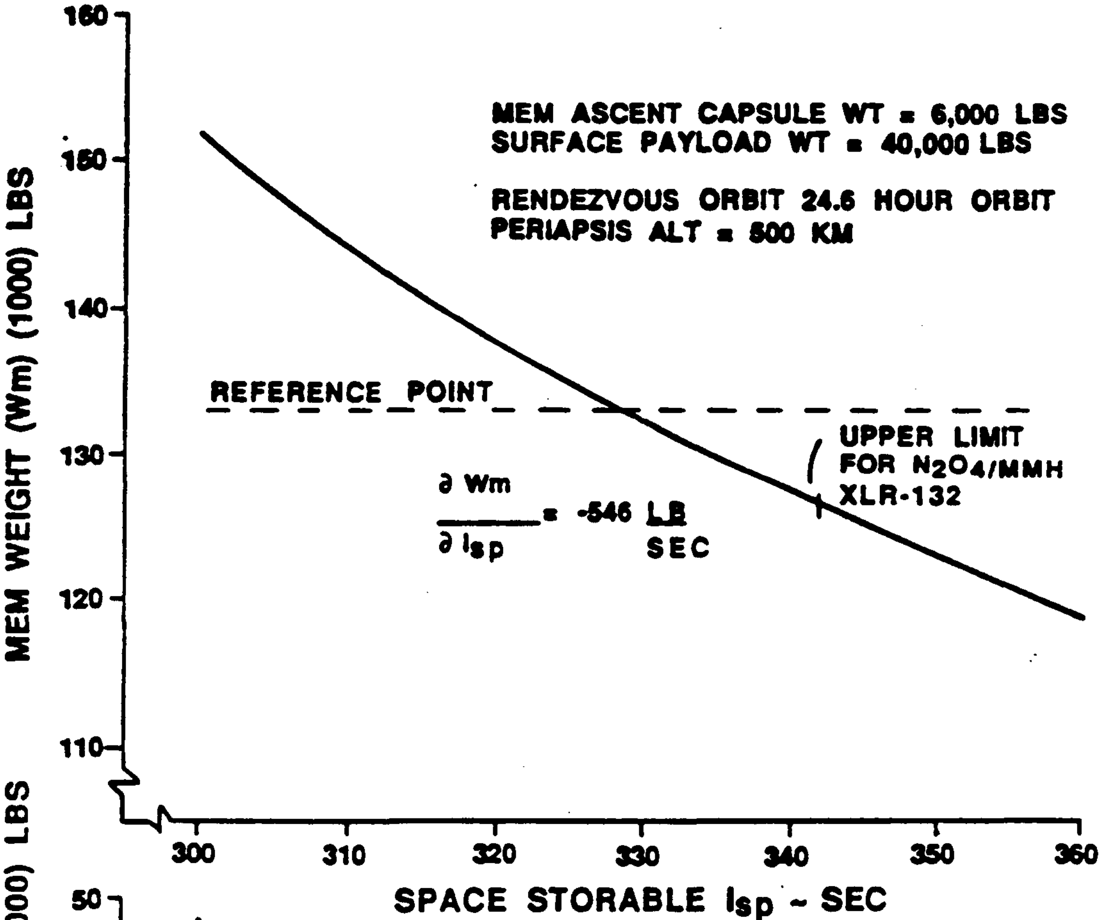
PLD VS F/W

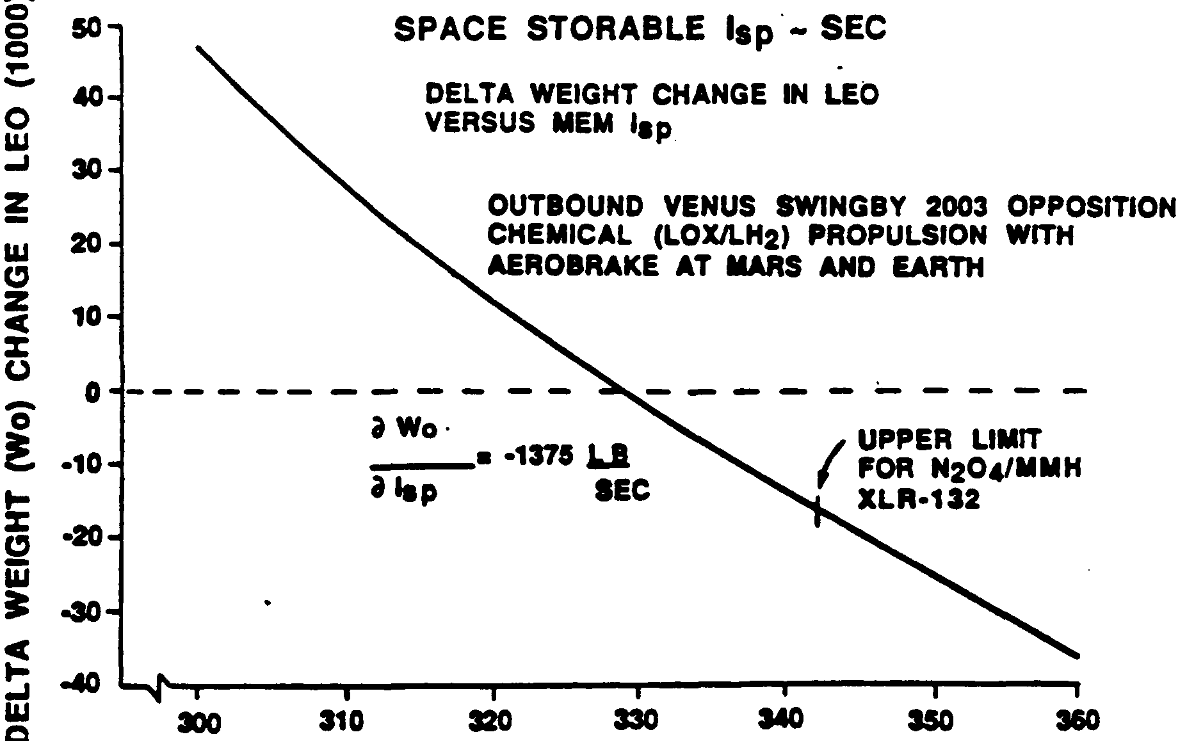


PD33/HAYS
2/29/88

1-958-8

MARS EXPLORATION
MEM WEIGHT VERSUS I_{sp}





MSFC/A. YOUNG

1-1161-8-5D

MARS EXPLORATION
LOX/LH_2 BOIL OFF ON MARS SURFACE

BOIL OFF	MARS SURFACE TIME	
	45 DAYS	500 DAYS
LH_2	518 LBS	5760 LBS
LOX	286 LBS	3168 LBS
TOTAL	804 LBS	8928 LBS

NOTE: NEED ≈ 30,000LBS LOX/LH_2 @ 6:1 MIXTURE RATIO FOR ASCENT STAGE

MEM WEIGHT FOR N_2O_4/MMH IS 133,000 LBS
MEM WEIGHT FOR LOX/LH_2 IS 112,000 LBS

MSFC/N. BROWN

1-1189-8-4T

MARS EXPLORATION
ASCENT AND DESCENT PROPULSION
EFFECT OF THRUST LEVEL AND Isp

SUMMARY REMARKS

- OPTIMUM TOTAL THRUST FOR MARS ASCENT STAGE IS 40,000 LBS TO RETURN A CREW MODULE TO A HIGHLY ELLIPTICAL ORBIT
- THE MOST EFFECTIVE INITIAL THRUST TO WEIGHT RATIO IS 0.80
- FOR THE MEM WEIGHT, THE PARTIAL WEIGHT TO A SECOND OF Isp IS -546 LB/SEC FOR STORABLE PROPELLANT
- THE PARTIAL LEO WEIGHT TO A SECOND OF Isp IS - 1375 LB/SEC FOR THE ABOVE MEM DESCENT AND ASCENT STAGE
- PRELIMINARY RESULTS INDICATE THAT LOX/LH_2 PROPELLANT WOULD BE AN EFFECTIVE PROPELLANT FOR THE MEM DESCENT AND ASCENT STAGE USED IN STAY TIME ON SURFACE OF MARS OF ~ 45 DAYS
- STORABLE PROPELLANT MAY BE A MORE EFFECTIVE PROPELLANT FOR CONJUNCTION CLASS MISSION WHERE STAY TIME ON MARS SURFACE IS UP TO 500 DAYS

MSFC/A. YOUNG

A COMPARISON OF LOW-THRUST AND IMPULSIVE TRAJECTORY CHARACTERISTICS AND PERFORMANCE WITH APPLICATION TO LUNAR AND MARS EXPLORATION

PRESENTATION TO THE

ADVANCED SPACE PROPULSION WORKSHOP

BY

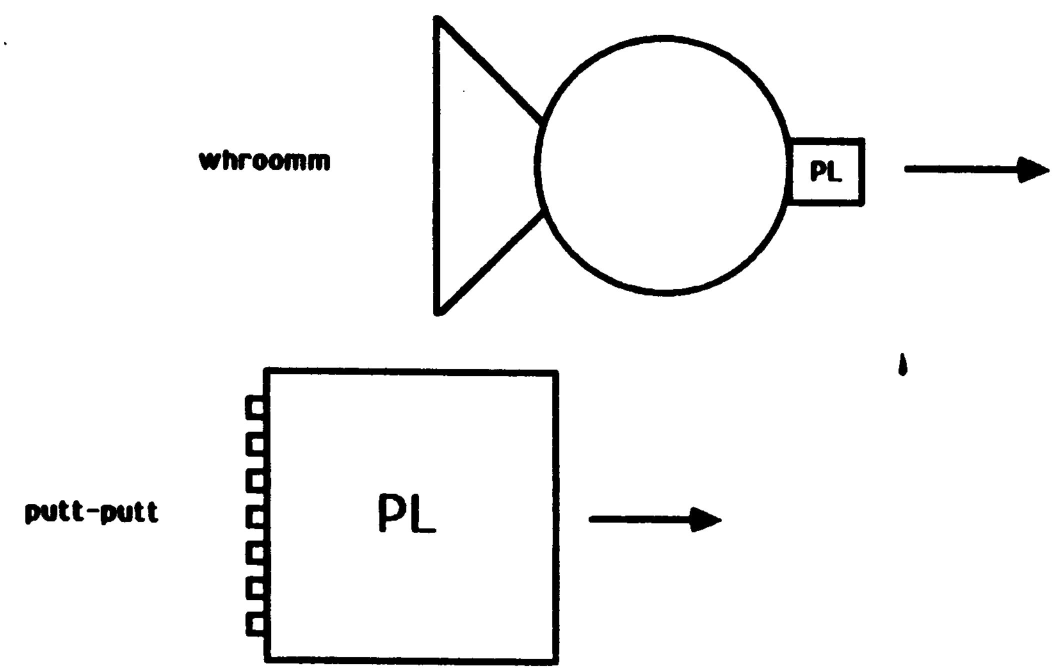
ALAN FRIEDLANDER
SCIENCE APPLICATIONS INTERNATIONAL CORPORATION

AT

NASA LEWIS RESEARCH CENTER

APRIL 13, 1988

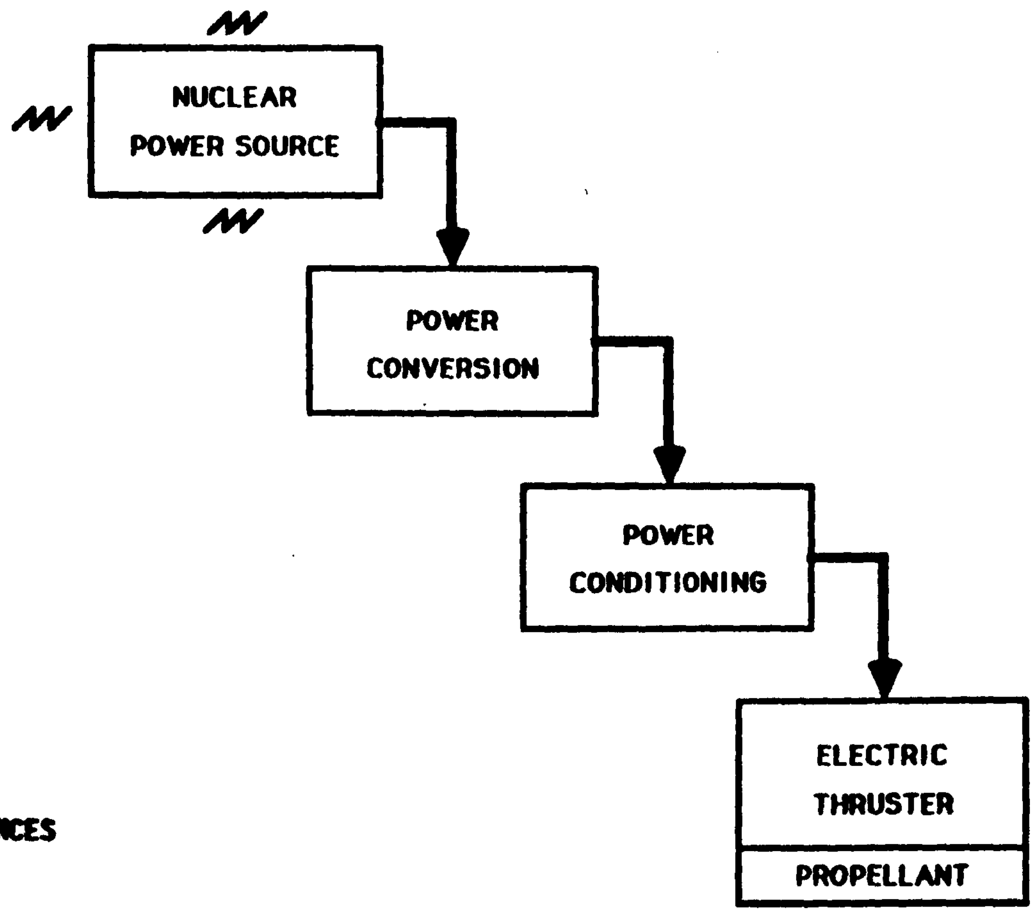
There was a young trucker from PEP
Who was ridiculed for being a "schlepp"
His neighbors were fast
With their chemical blast
But he delivered more with his NEP.



A SLIGHT OVERSTATEMENT !

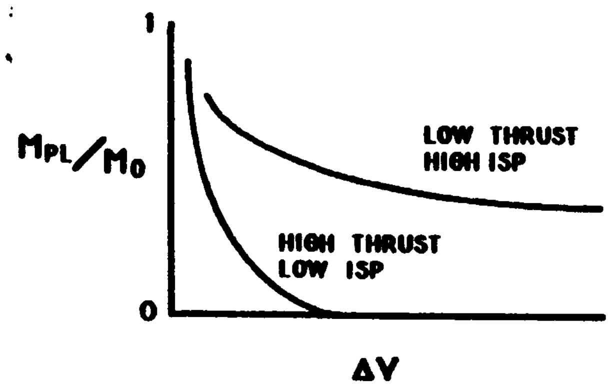
SAIC SPACE SCIENCES

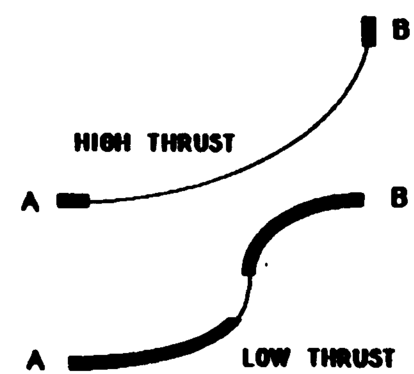
Our hapless trucker's neighbors were jealous
They accused him of not being too zealous
His ion plasma solution
Produced Mercury pollution
And his irradiated payload was a menace.



SAIC SPACE SCIENCES

Not guilty, if you please, my explanation
Why I do much better with low acceleration
Specific impulse efficiency
Trajectory shaping resiliency
And lightweight shielding against radiation.





Allow me to give you further reason
Why I'm not guilty of this low treason
Large payloads and high power
For Mars exploration tomorrow
And my propellant is Xenon this season !

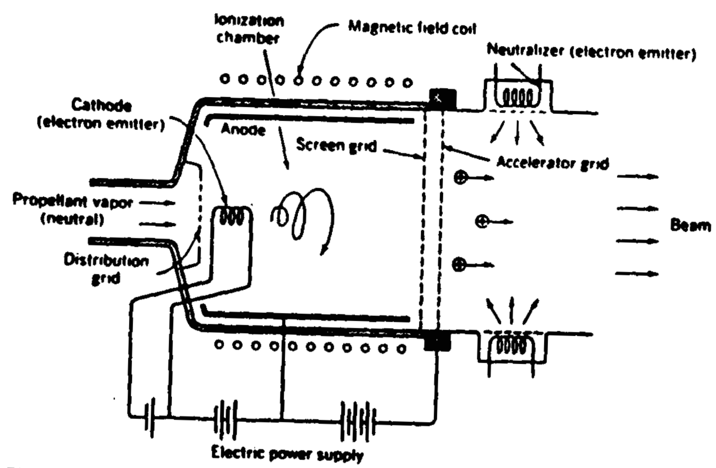
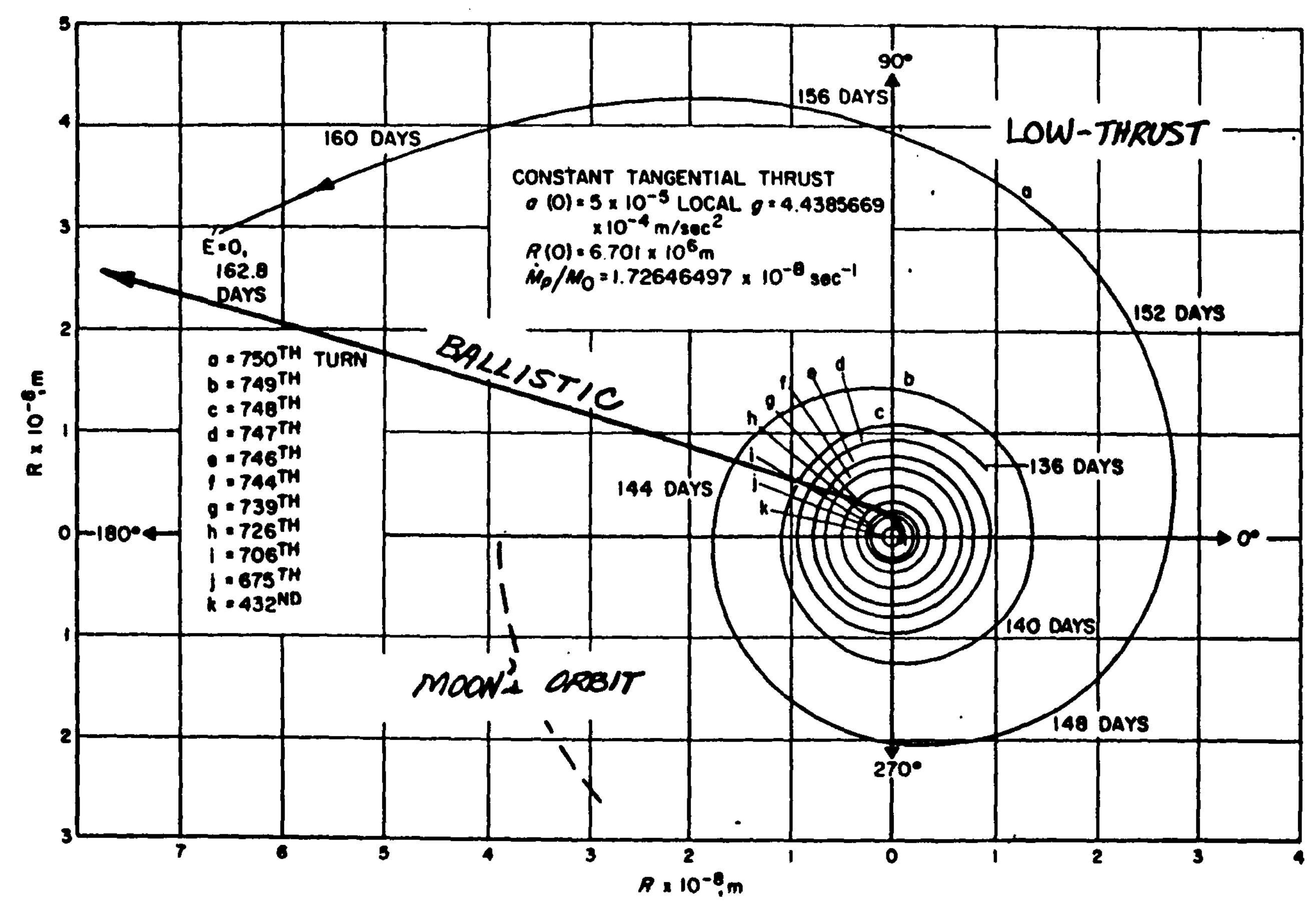


Fig. 13-5. Simplified schematic diagram of electron bombardment ion thruster.

KEY ATTRIBUTES IN COMPARING LOW-THRUST AND HIGH-THRUST TRAJECTORIES

- FLIGHT PATH SHAPING
- LAUNCH OPPORTUNITY DEPENDENCY
- TRANSFER TIME
- ΔV
- SPECIFIC IMPULSE
- MASS RATIOS (SYSTEM, PROPELLANT, PAYLOAD)



COMPARISON OF LOW-THRUST AND BALLISTIC EARTH ESCAPE TRAJECTORIES

SPACE SCIENCES

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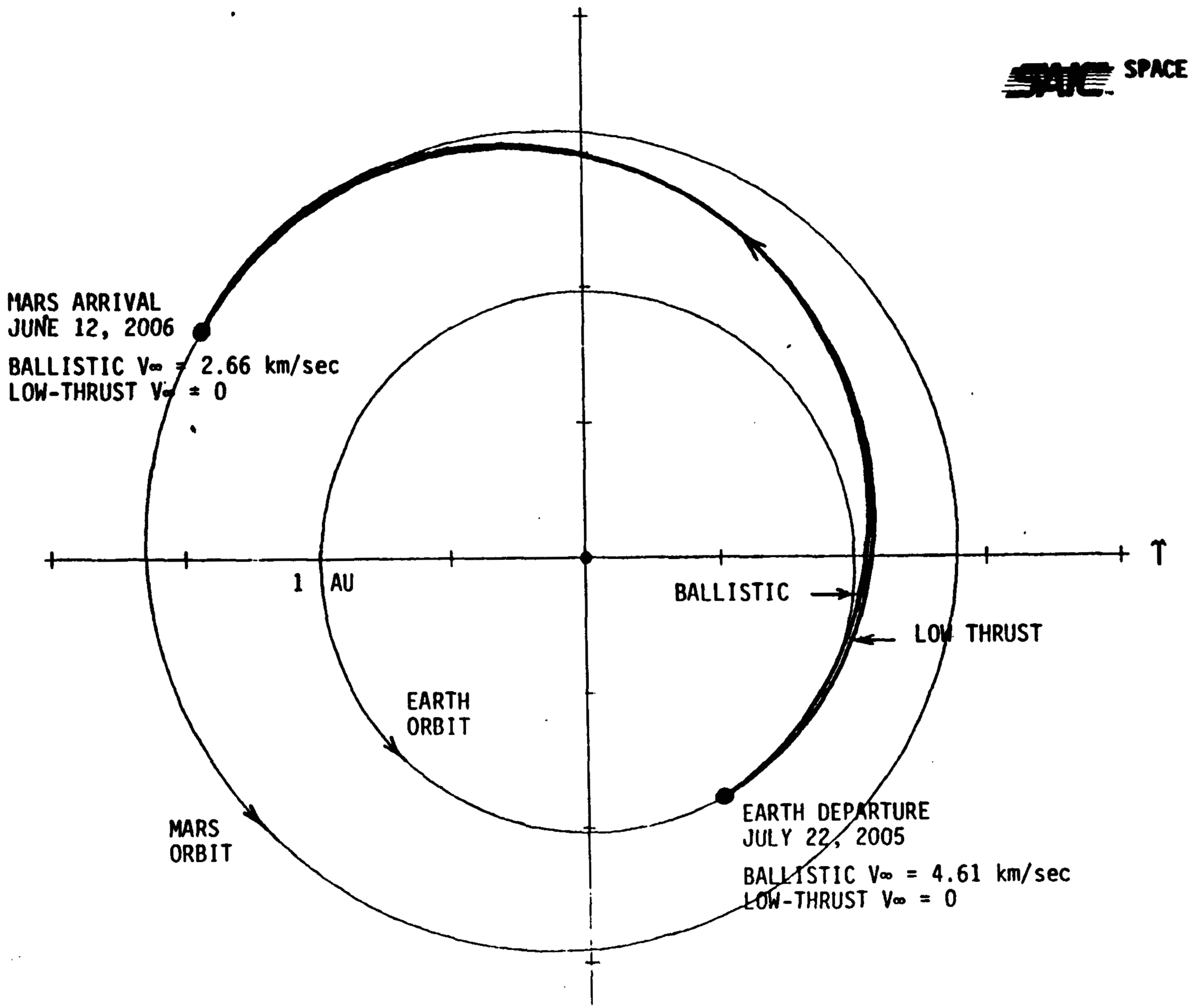
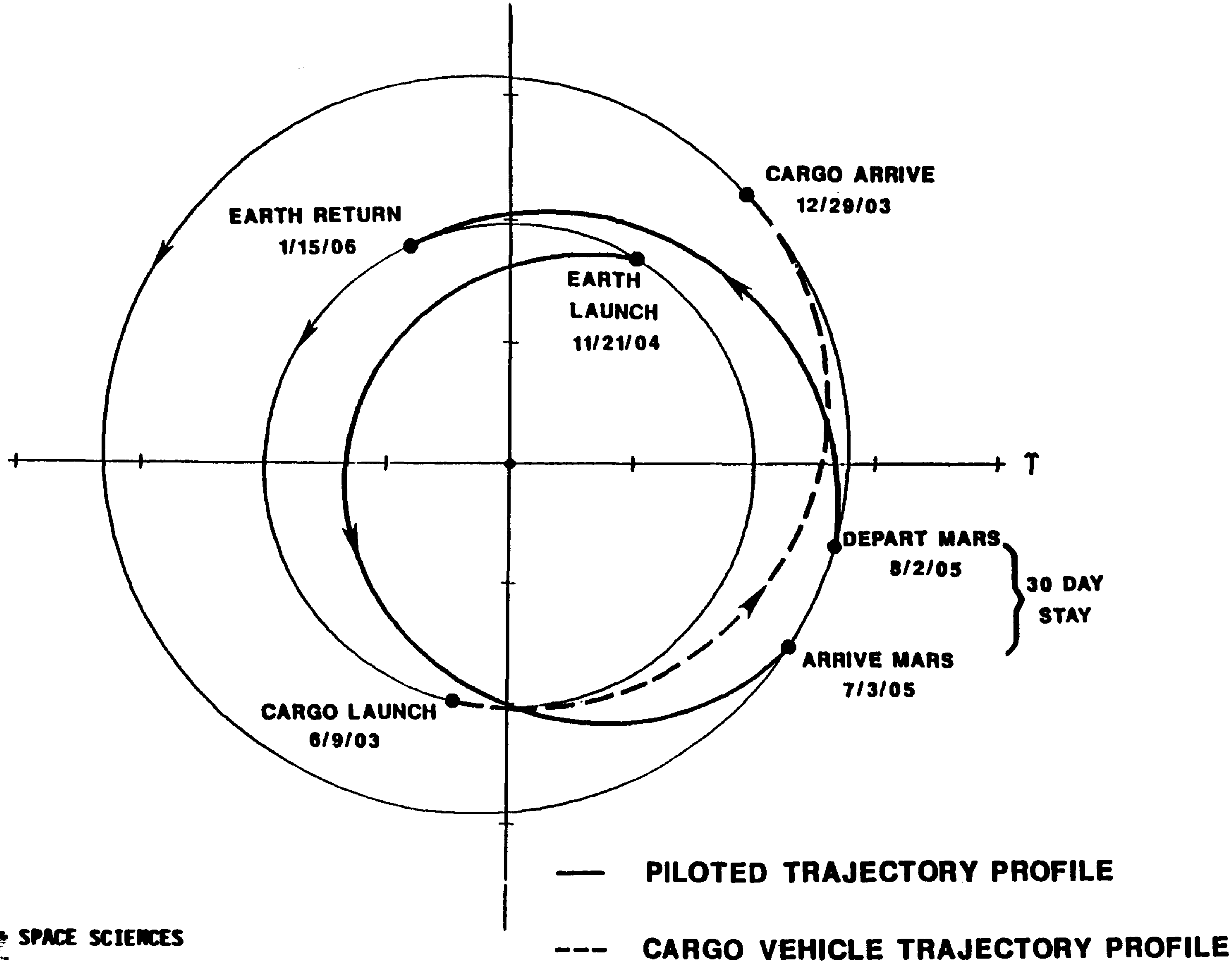


COMPARISON OF LOW-THRUST AND BALLISTIC EARTH-MARS HELIOCENTRIC TRAJECTORIES



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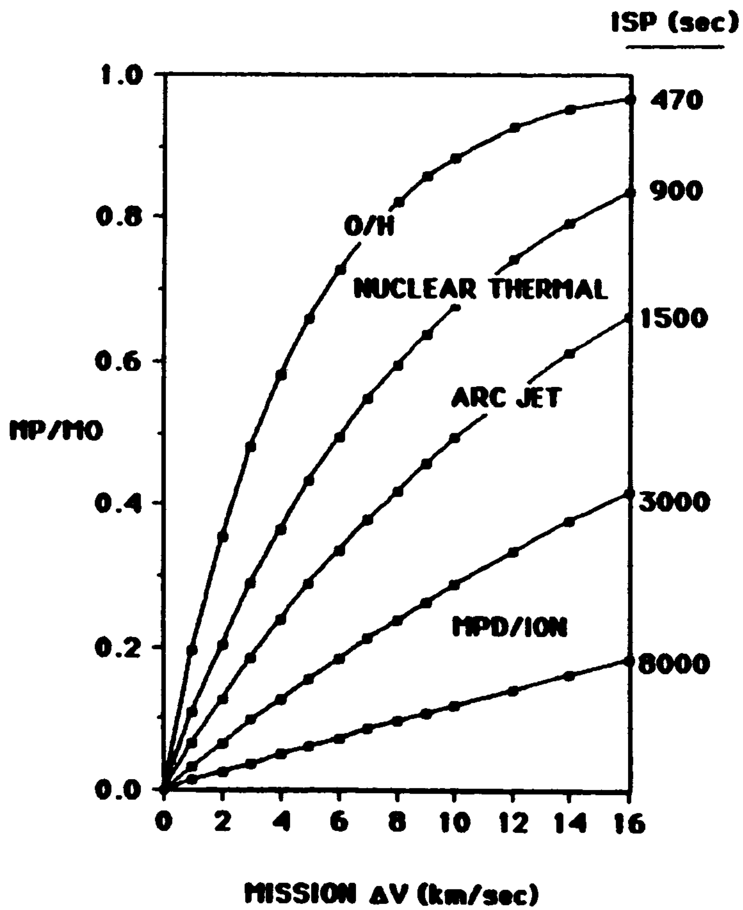
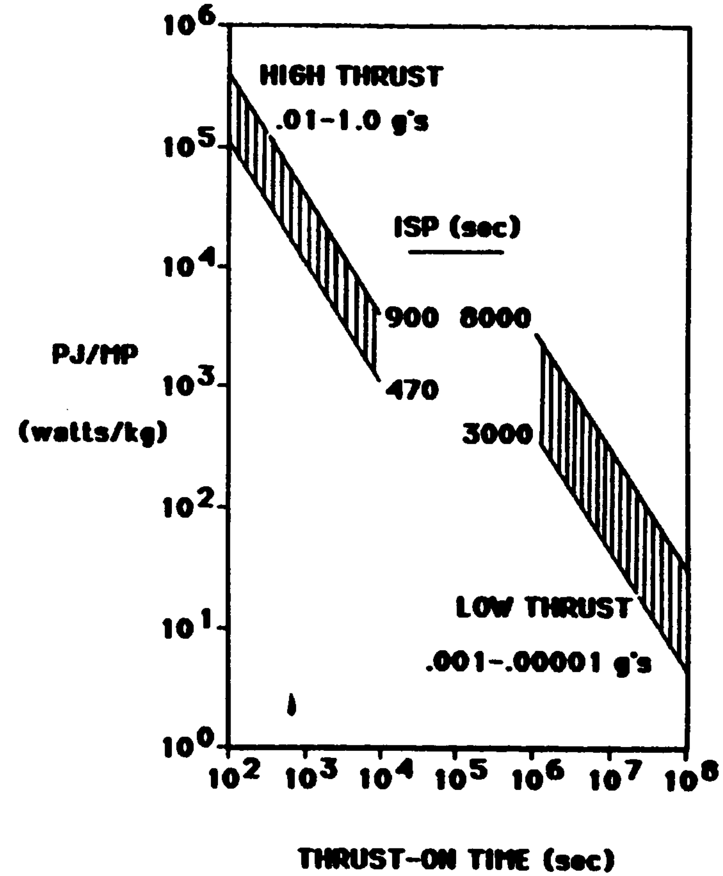
Piloted Mars Mission - Split Option Trajectory



PROPELLANT FRACTION CHARACTERISTICS



THRUST POWER CHARACTERISTICS

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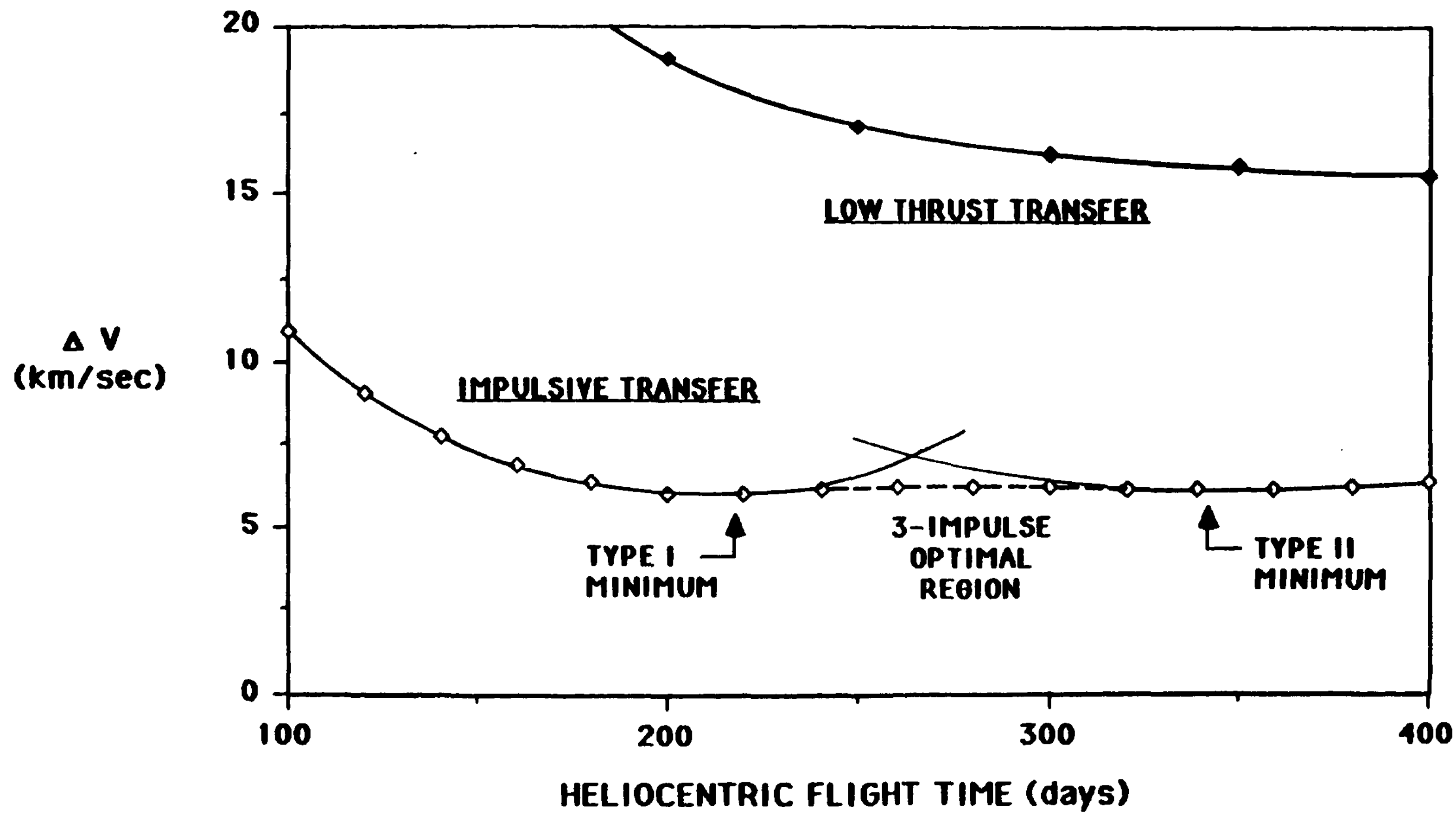
CHARACTERISTICS OF OPTIMAL EARTH-TO-MARS TRANSFERS
500 KM CIRCULAR ORBIT TERMINALS



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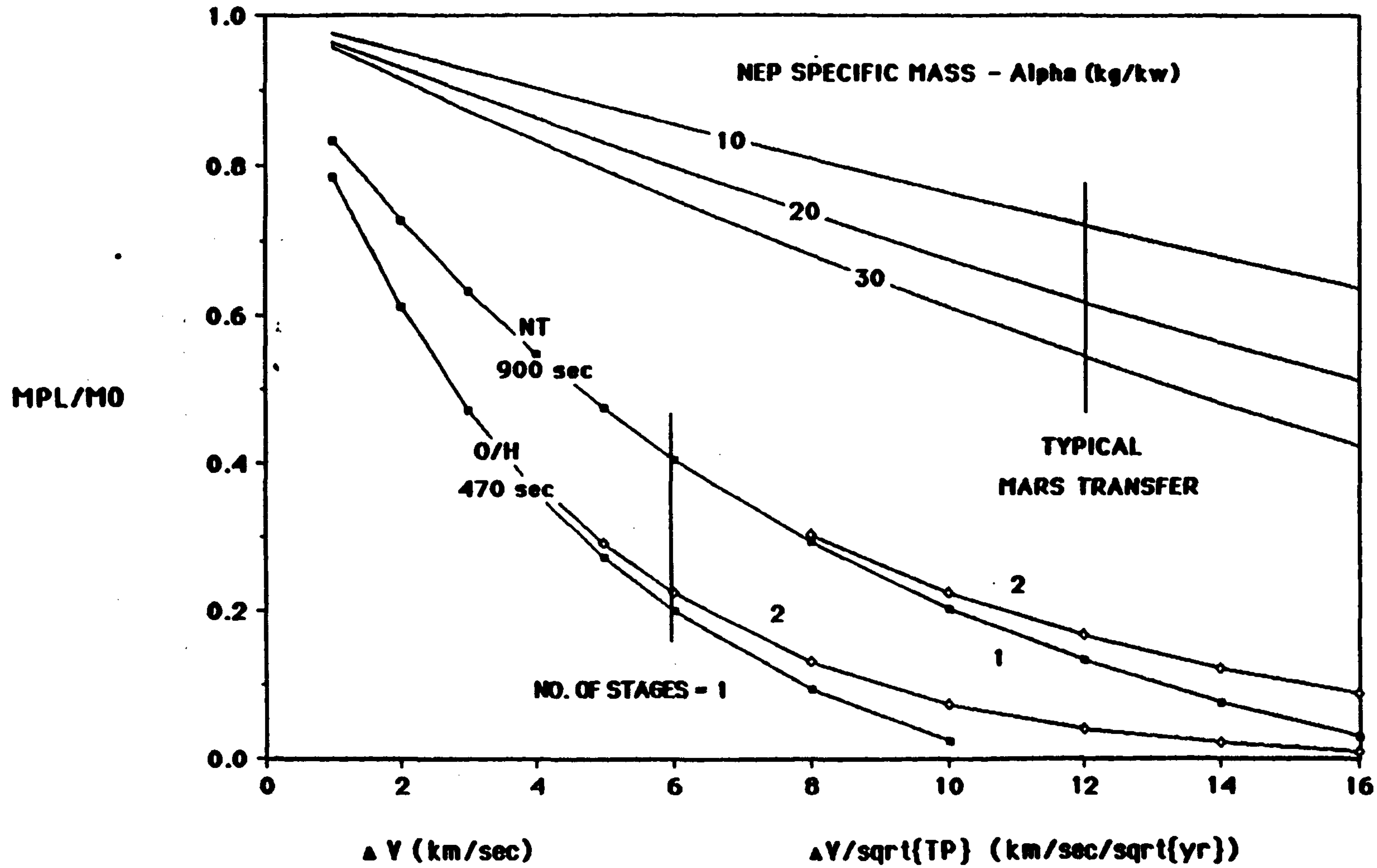

PAYLOAD MASS FRACTION CHARACTERISTICS

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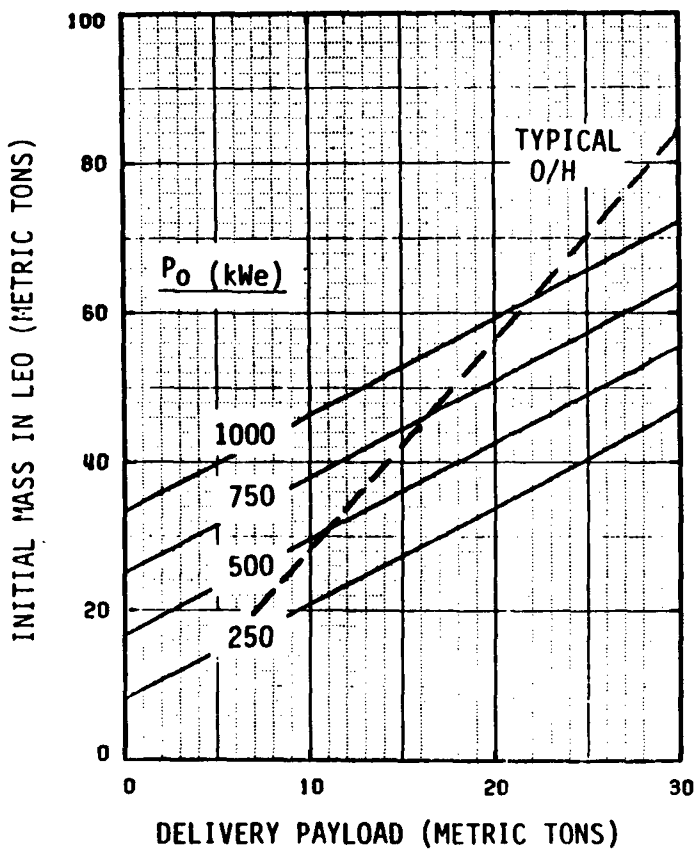

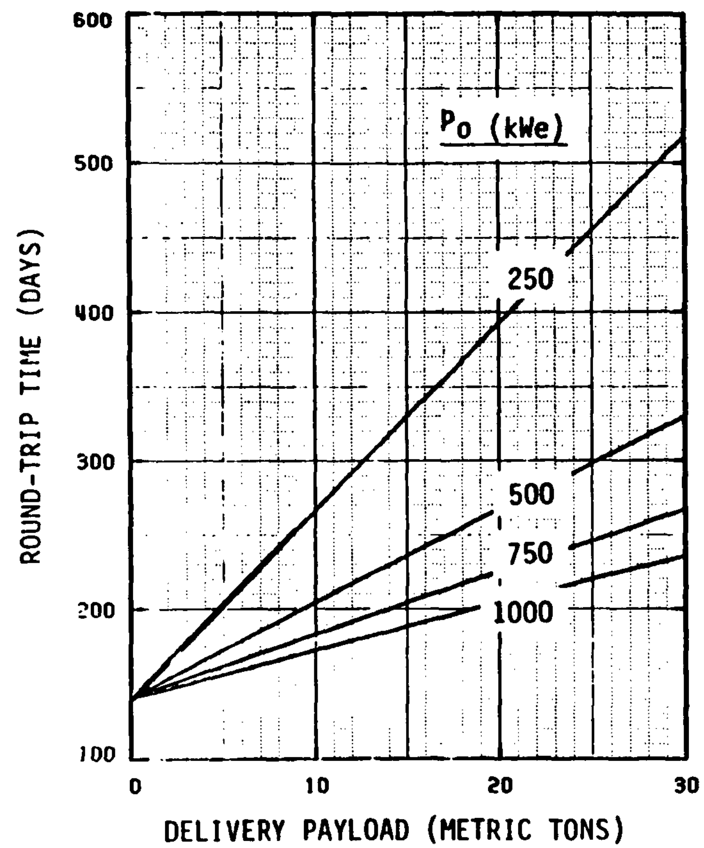

PAYLOAD DELIVERY TO LUNAR ORBIT WITH NUCLEAR ELECTRIC PROPULSION

LEO-TO-LLO (100 km)
RETURN PAYLOAD = 0

I_{sp} = 4000 sec, η = 0.76
α = 20 kg/kWe, k_{T+R} = 0.1

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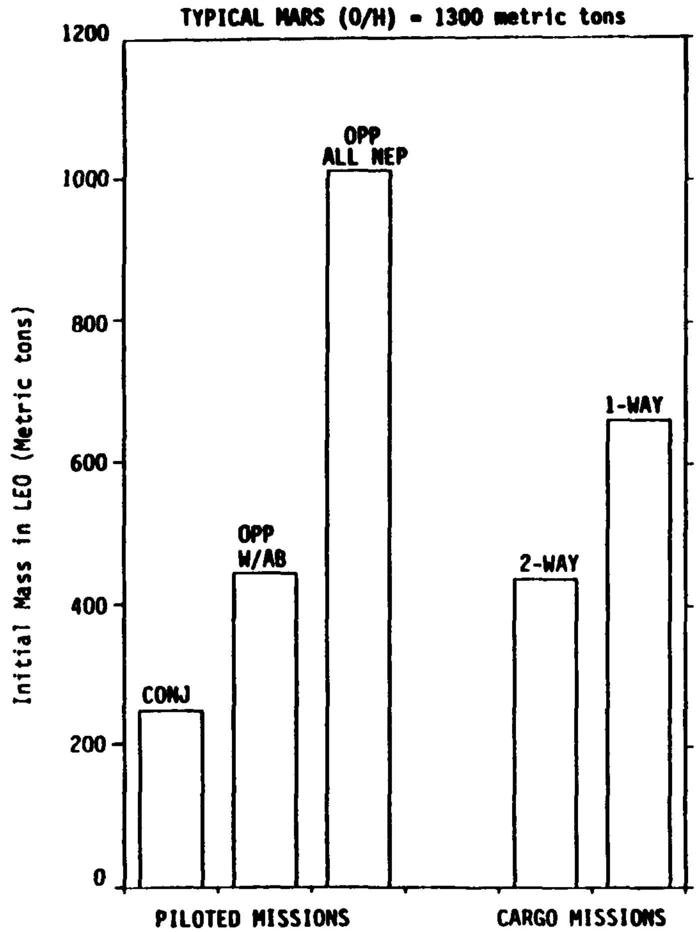
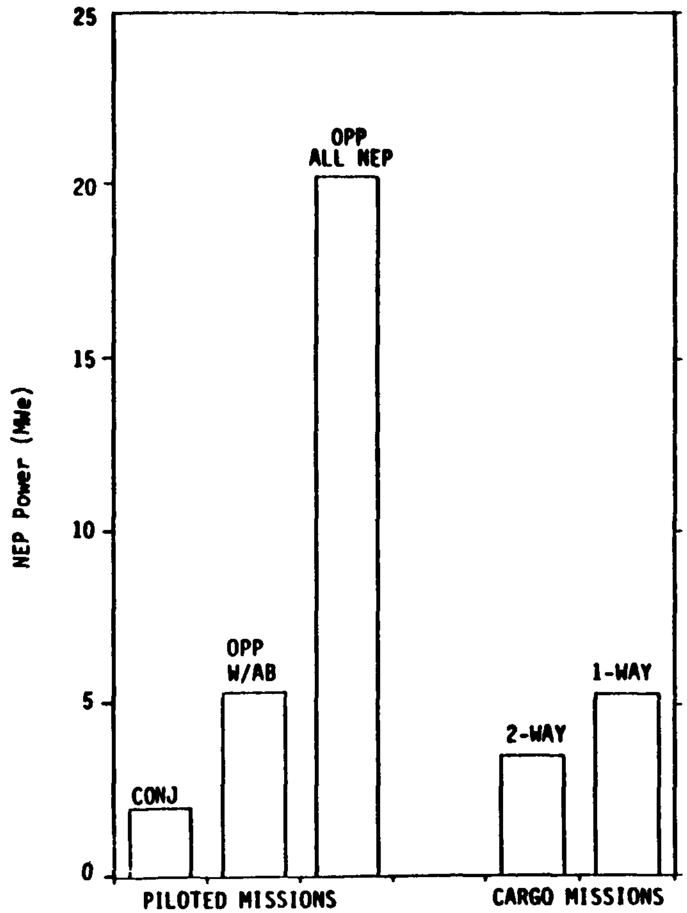
TRIP TIME CHARACTERISTICS FOR MANNED MARS MISSIONS

FLIGHT MODE	ROUND-TRIP TIME (DAYS)	MARS STAY TIME (DAYS)
■ HIGH-THRUST PROPULSION		
SPLIT SPRINT		
CARGO MISSION	205 - 340 (1-WAY)	415 - 580 (TO REND.)
PILOTED MISSION	440 - 470	20 - 40
OPPOSITION-CLASS	500 - 565	50 - 70
VENUS SWINGBY	405 - 735	50 - 70
CONJUNCTION-CLASS	955 - 1015	330 - 560
■ LOW-THRUST PROPULSION[(a)]		
OPPOSITION-CLASS	600 - 700[(b)]	70 - 85
CONJUNCTION-CLASS	1090 - 1150[(c)]	215 - 240

(a) CREW ROUND-TRIP TIME LISTED
(b) ADDITIONAL 125 - 200 DAYS OF EARTH ESCAPE/CAPTURE SPIRALS (UNMANNED)
(c) ADDITIONAL 490 - 660 DAYS OF EARTH ESCAPE/CAPTURE SPIRALS (UNMANNED)

PERFORMANCE SENSITIVITY TO MISSION TYPE/CLASS FOR 2007 LAUNCH
(Ion Thrusters, Isp = 6000 sec, Specific Mass = 15 kg/kW)





MISSION CLASS:

CONJ	CONJUNCTION WITH 7 MONTH STAYTIME AT MARS
OPP W/AB	OPPOSITION WITH 2 MONTH STAYTIME AT MARS, AEROBRAKING AT MARS AND EARTH
ALL NEP	OPPOSITION WITH 2 MONTH STAYTIME AT MARS, NO AEROBRAKING
1-WAY	EARTH TO MARS DELIVERING 400 MT
2-WAY	EARTH TO MARS DELIVERING 227 MT, MARS TO EARTH RETURNING 57 MT

SPACE SCIENCES

PILOTED MISSION MASS ASSUMPTIONS: EARTH DEPARTURE WITH 133 MT PAYLOAD
MARS ARRIV[illegible] ITH 72 MT TO SURFACE
EARTH R[illegible] ITH 61 MT PAYLOAD

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COMPARATIVE CHARACTERISTICS SUMMARY

ITEM	HIGH-THRUST	LOW-THRUST
PLANETOCENTRIC MANEUVERS (ESCAPE/CAPTURE)	FAST HYPERBOLIC PATHS	SLOW SPIRAL PATHS
HELIOCENTRIC TRAJ. SHAPE	GENERALLY SIMILAR, BUT DEPENDS ON MISSION MODE	
THRUST-ON TIME	MINUTES-HOURS	HUNDREDS OF DAYS
TRIP TIME EFFECT ON MASS PERFORMANCE	VARIABLE BUT WITH DISCRETE OPTIMA	CONTINUOUSLY VARIABLE (LONGER IS BETTER)
LAUNCH OPPORTUNITY EFFECT	MORE SENSITIVE	LESS SENSITIVE
KEY PERFORMANCE PARAMETER	ISP & PROP. INERTS	PROP. SPECIFIC MASS
MISSION ΔV	NEAR-IMPULSIVE	1.5 TO 3 TIMES HIGHER
SPECIFIC IMPULSE	450 - 1000 SEC	10 TIMES HIGHER
PROPELLANT MASS FRACTION	40 - 90 %	10 - 40 %
PAYLOAD MASS FRACTION	5 - 50 %	30 - 80 %

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HYBRID PROPULSION AND OTHER TOPICS

PRESENTATION TO THE

ADVANCED SPACE PROPULSION WORKSHOP

BY

ALAN FRIEDLANDER
SCIENCE APPLICATIONS INTERNATIONAL CORPORATION

AT

NASA LEWIS RESEARCH CENTER

APRIL 13, 1988

POTENTIAL BENEFITS OF HYBRID PROPULSION

- **PERFORMANCE**
  - LOWER MASS REQUIREMENT
  - SHORTER TRANSIT TIME
- **CREW SAFETY**
  - MORE FAILURE-RESPONSE OPTIONS VIA REDUNDANCY
  - ENHANCED ABORT MODES
  - USE PERFORMANCE GAINS TO INCREASE SAFETY

HYBRID PROPULSION DEFINITION

- THE USE OF MORE THAN ONE TYPE OF PROPULSION SYSTEM TO ACCOMPLISH AN END-TO-END MISSION OBJECTIVE

- IMPLEMENTATION OPTIONS

  - STAGED (SEPARABLE) SYSTEMS APPROPRIATE FOR DIFFERENT MISSION PHASES

    E.G. *HIGH-THRUST CHEMICAL PROPULSION FOR EARTH ESCAPE*

    LOW-THRUST ELECTRIC PROPULSION FOR INTERPLANETARY TRANSFER

  - INTEGRATED (BIMODAL) SYSTEMS

    E.G. *NUCLEAR THERMAL/NEP USING COMMON REACTOR POWER SOURCE*

CANDIDATE PROPULSION SYSTEMS FOR PILOTED MARS ROUND TRIP

	CHEMICAL CRYOGENIC	CHEMICAL STORABLE	NUCLEAR THERMAL	NUCLEAR ELECTRIC	SUPPLEMENTAL AEROBRAKING
EARTH ESCAPE	X		⊗		
MARS TRANSFER		X[a]		⊗	
MARS ORBIT CAPTURE	X	X[b]	X	⊗	X[b]
MARS ENTRY & LANDING		X[c]			X
MARS ASCENT & RENDEZVOUS	X	X			
MARS ESCAPE	X		X	⊗	
EARTH TRANSFER		X[a]		⊗	
EARTH ORBIT CAPTURE	X	X[b]	⊗		X[b]

a Trajectory correction maneuvers
b Periapsis raise after aerocapture
c Deorbit and terminal soft landing

⊗ Hybrid propulsion example

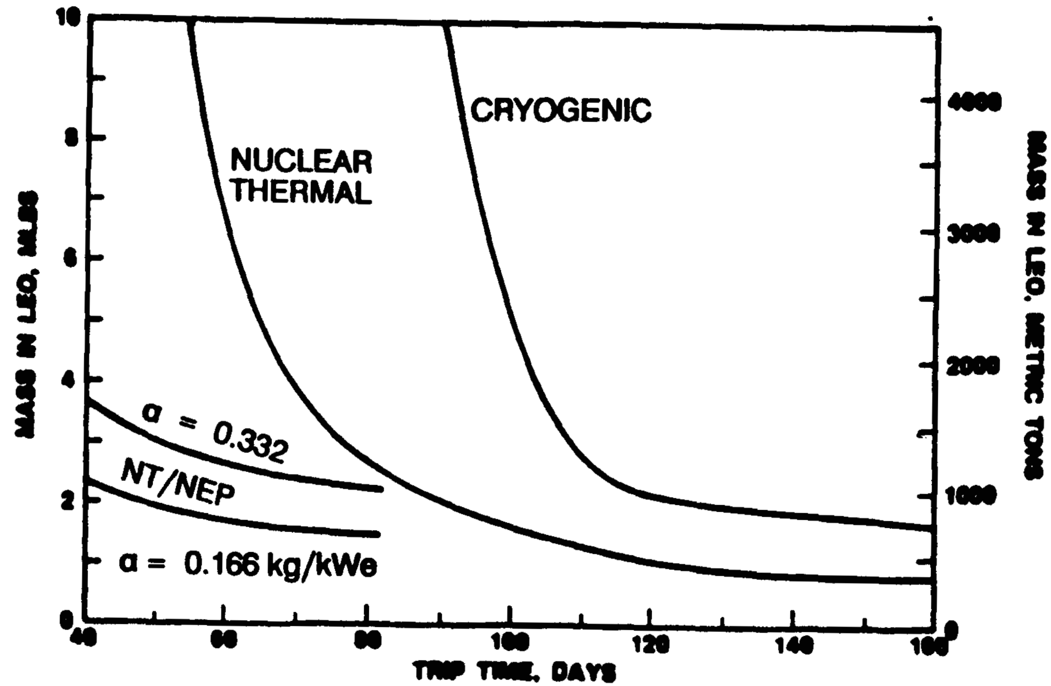
OUTBOUND PAYLOAD = 182.5 METRIC TONS

CRYOGENIC: I_{SP} = 470 SEC

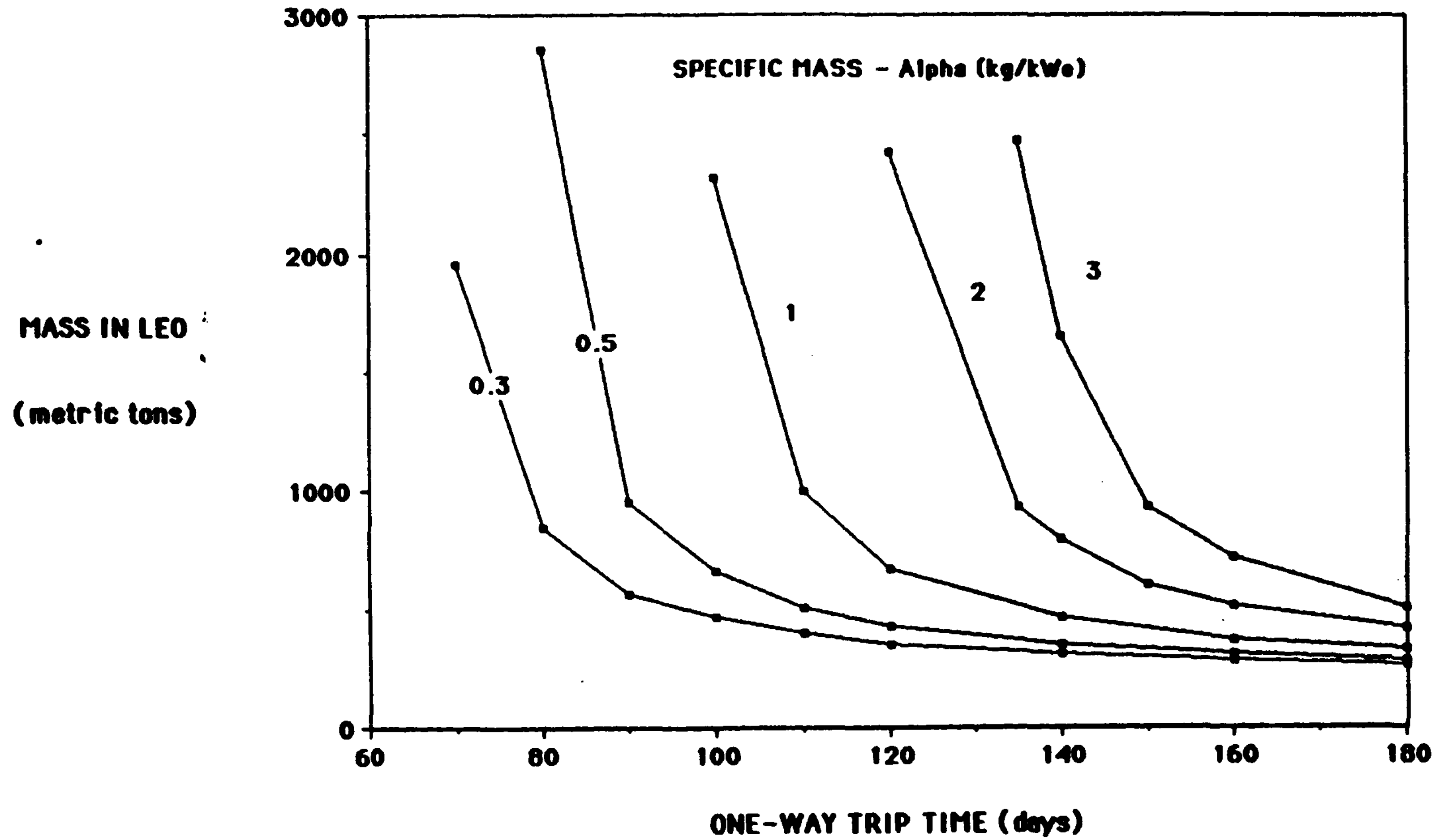
NUCLEAR THERMAL: I_{SP} = 1000 SEC, P_R = 4000 MWt, M_{R+S} = 11 METRIC TONS

HYBRID NT/NEP: NT FOR EARTH ESCAPE
NEP FOR MARS TRANSFER & CAPTURE
I_{SP} = 5000 SEC, P_e = 1320 MWe

COMMON REACTOR



MASS IN LEO VS. ONE-WAY TRIP TIME TO MARS (DATA FROM HARRIS AND PERRY/ROCKETDYNE)



PERFORMANCE OF HYBRID NUCLEAR THERMAL/NEP PROPULSION

MARS ROUND-TRIP WITH PAYLOAD: 133 Mt OUT, 61 Mt RETURN

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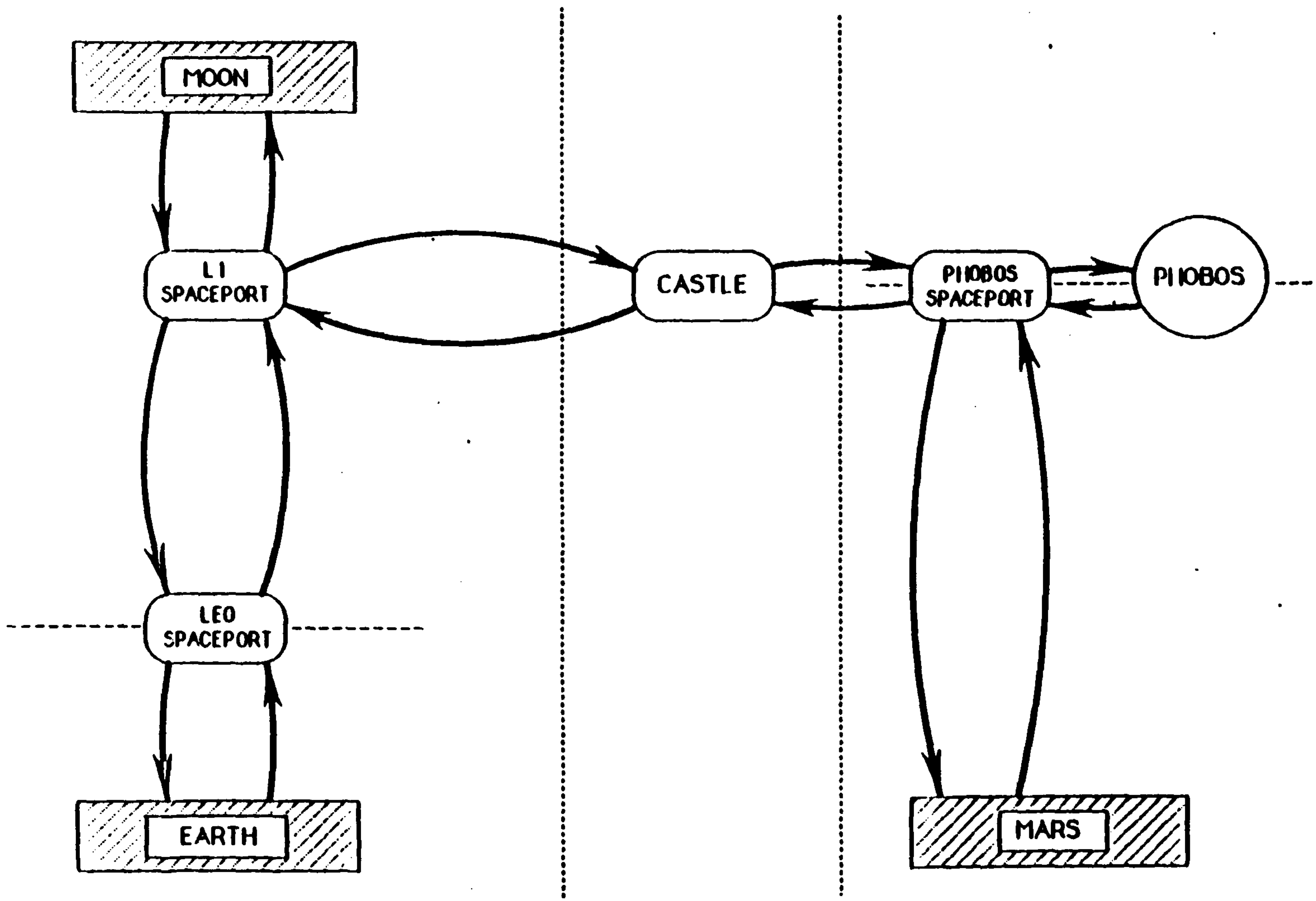
CANDIDATE STAGING SCENARIO FOR SUPPORT OF A SUSTAINED MARS BASE

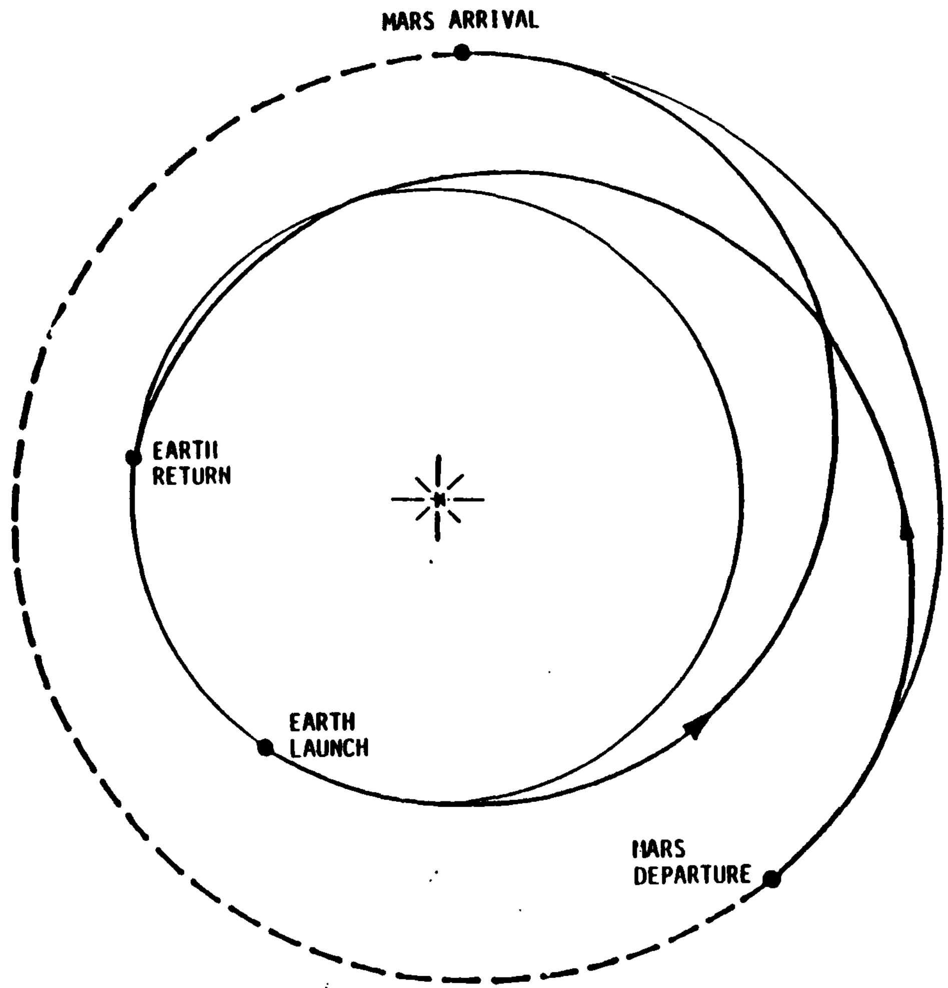


STUDY OBJECTIVES

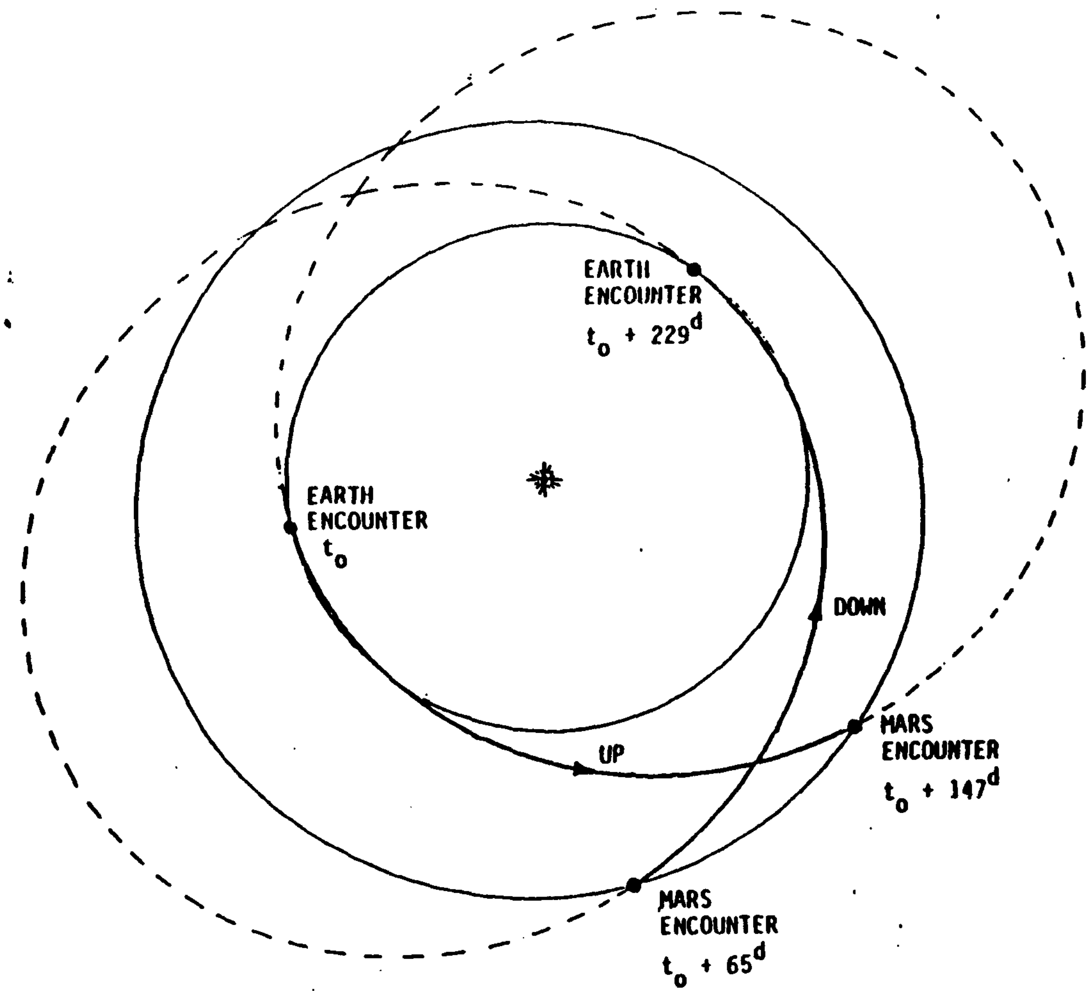
- OBJECTIVES
  - IDENTIFY A MINIMUM PROPELLANT TRANSPORTATION SYSTEM
  - EVALUATE THE PERFORMANCE OF CIRCULATING ORBITS IN AN "OPERATIONAL" SCENARIO

- APPROACH
  - ASSUME THE EXISTANCE OF A MARS BASE AND INFRASTRUCTURE ELEMENTS
  - ASSEMBLE INFRASTRUCTURE ELEMENTS INTO A REASONABLE TRANSPORTATION SYSTEM
  - DETERMINE ACTUAL ΔVs AND FLIGHT TIMES OVER A 15-YEAR CYCLE
  - CALCULATE THE PROPELLANT REQUIREMENTS FOR ALL VEHICLES
  - COMPARE RESULTS FROM CONJUNCTION, UP/DOWN ESCALATOR AND VISIT ORBITS

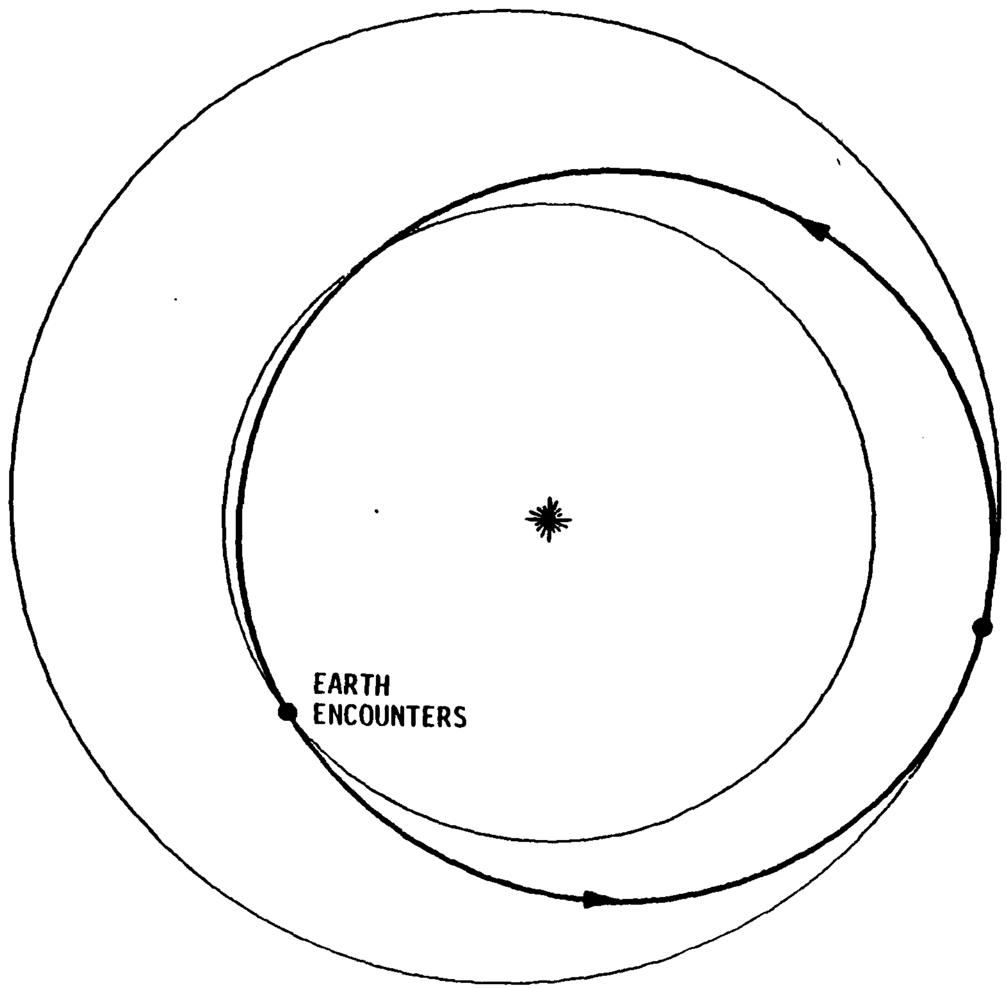
TRAJECTORY OPTION 1: CONJUNCTION CLASS



TRAJECTORY OPTION 2: UP/DOWN ESCALATOR CLASS



TRAJECTORY OPTION 3: VISIT CLASS



GENERAL ASSUMPTIONS

- CREW REQUIREMENTS
  - 20 PEOPLE ON THE SURFACE OF MARS (AVERAGE)
  - 6 PEOPLE OPERATING THE PHOBOS SPACEPORT
  - 6 PEOPLE OPERATING THE CASTLE

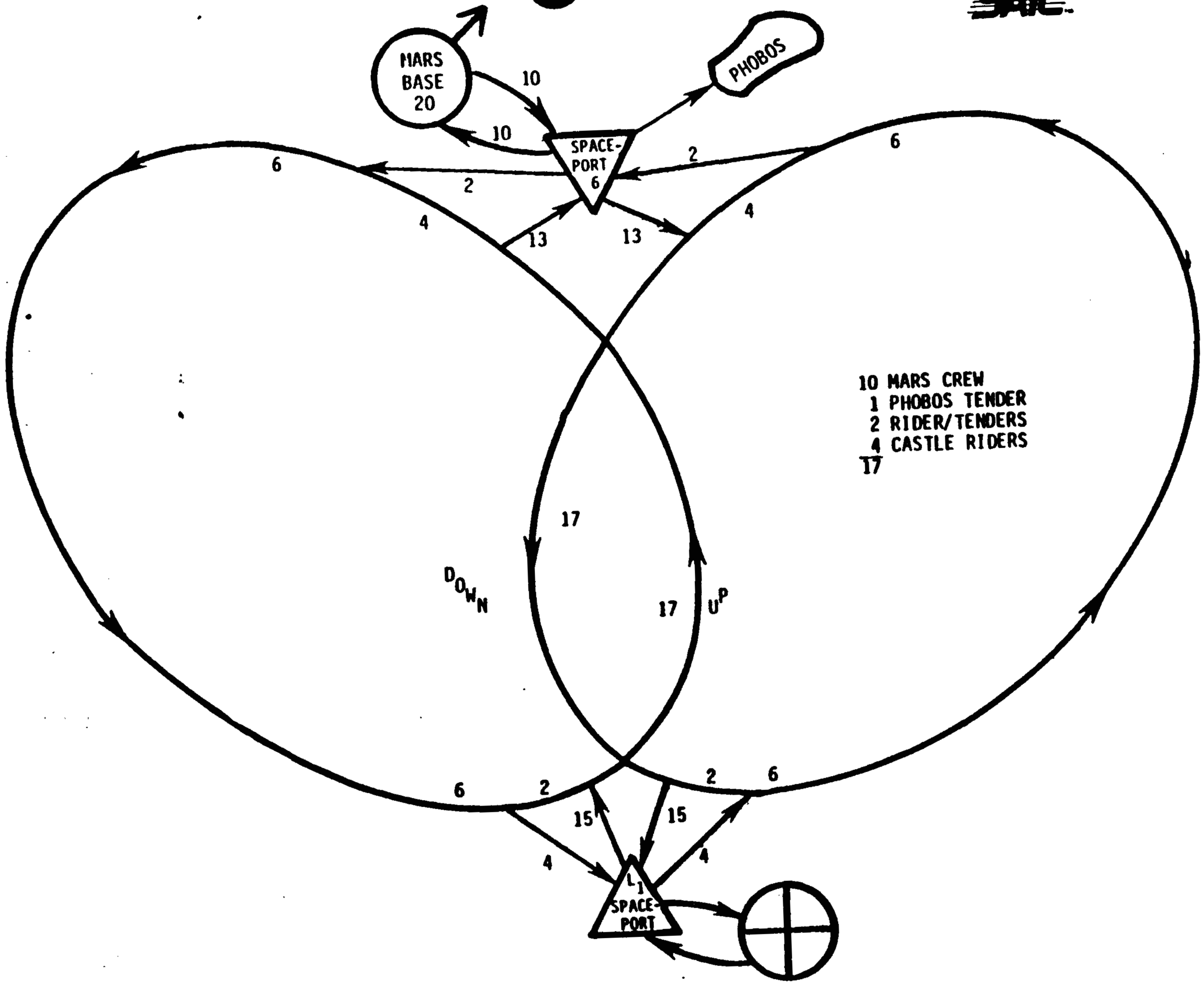
- CASTLE VEHICLE
  - 400 M TONS PLUS PROPULSION INERTS (CONJUNCTION MISSIONS)
  - 460 M TONS (CIRCULATING ORBITS)
  - ARTIFICIAL GRAVITY
  - CLOSED AIR AND WATER CYCLES
  - OTHER CONSUMABLES = 3 kg/PERSON/DAY

SPACECRAFT ASSUMPTIONS

- ALL MANEUVERS WITH LOX/LH_2, I_{SP} = 460 SEC, 7:1 MIXTURE RATIO
- CASTLE-RESIDENT PROPULSION INERTS = 10% OF MAXIMUM PROPELLANT LOADING
-

TRANSPORTATION ELEMENT	MAX. PROPELLANT LOADING (M TONS)	TOTAL INERTS INCLUDING CREW/CARGO MODULES (M TONS)	AUXILIARY TANKS FOR PROPULSION
LUNAR-L_1 TANKER	50	8.5	NO
LEO-L_1 TANKER/TRANSPORTER	42	17	NO
TAXI	42	17	NO
PHOBOS-BASED OTV	42	7	YES
MARS-PHOBOS SHUTTLE	50	17.9	NO

- AUXILIARY TANK INERTS = 10% OF MAXIMUM PROPELLANT LOADING



CREW LOADING SCHEMATIC OF UP/DOWN ESCALATOR TRANSPORTATION MODE

SUMMARY COMPARISON OF PROPELLANT REQUIREMENTS OVER 15 YEARS ASSUMING CRYOGENIC PROPULSION IN SUPPORT OF A SUSTAINED MARS BASE

	CONJUNCTION WITH STOPOVERS	UP/DOWN ESCALATOR	DOWN ESCALATOR	DOWN ESCALATOR PHOBOS-SUPPLIED VIA NEP
LOX FROM MOON	22,519	21,975	16,216	---
LH_2 FROM LEO	3,217	3,138	2,317	---
LOX FROM MARS	919	919	459	459
LH_2 FROM MARS	131	131	66	66
LOX FROM PHOBOS	2,992	5,834	4,027	6,030
LH_2 FROM PHOBOS	427	833	575	861
NEP PROPELLANT FROM EARTH	---	---	---	1,505
TOTAL PROPELLANT (M TONS)	30,205	32,830	23,660	8,921

TRANSPORTATION MODE COMPARISON CHART

	CONJUNCTION	UP/DOWN ESCALATOR	DOWN ESCALATOR
NUMBER OF CASTLES	2	2	1
TOUR OF DUTY, YEAR	4.8	5	6.5
MARS CREW FLIGHTTIME, YEAR	1.6	.9	2.1
MARS STAYTIME, YEAR	3.2	4.1	4.4
MARS STAYTIME, % OF TOUR	67	82	67
PHOBOS TENDER CREW TOUR OF DUTY, YEAR	4.8	4.3-5.0	6.5
CASTLE RIDER CREW TOUR OF DUTY, YEAR	---	4.3	4.3
CASTLE CREW CAPACITY	17	17	17
NUMBER OF SORTIES PER 15 YEARS	7	14	7
NUMBER OF PERSONNEL TO MARS VICINITY AND RETURN IN 15 YEARS	119	133	105
CASTLE UTILIZATION EFF, % TIME USED TO TRANSPORT MARS SURFACE CREW	39	21	100

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ASAO

EVALUATION OF ADVANCED PROPULSION/ POWER CONCEPTS

PRESENTED
TO

ADVANCED SPACE PROPULSION
WORKSHOP

BY
ADVANCED SPACE ANALYSIS OFFICE
SVERDRUP/NASA-LERC

APRIL 12-13, 1988

C-10

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EVALUATION OF ADVANCED PROPULSION CONCEPTS

ASAO

BACKGROUND:

- A REQUEST HAS BEEN MADE BY CODE Z TO ASAO/LeRC TO PROVIDE AN EVALUATION OF ADVANCED PROPULSION CONCEPTS AND THEIR SUITABILITY FOR FUTURE NASA INITIATIVES

- SPECIFICALLY - WHAT PROPULSION CONCEPTS ARE AVAILABLE?
  - WHAT CONCEPTS MAKE SENSE FOR FUTURE MISSIONS?

- REQUESTED INFORMATION TO INCLUDE AN ASSESSMENT OF THE CAPABILITIES OF VARIOUS PROPULSION/POWER COMBINATIONS AND THEIR ASSOCIATED PROS AND CONS

- THIS TASK EXAMINED OVER 700 POWER/PROPULSION COMBINATIONS OBTAINED FROM OVER 50 REFERENCES DURING A FOUR WEEK PERIOD

- THIS WORK IS IN RESPONSE TO THE DECEMBER 1987 REQUEST FOR RESULTS TO BE PRESENTED TO THE SSTAC WORKING GROUP FOR LUNAR AND PLANETARY MISSION PROPULSION ON MARCH 10-11, 1988

PRESENTATION OUTLINE

- OBJECTIVE OF STUDY
- APPROACH (SUCCESSIVE FILTERING/SCREENING PROCESS)
  - DEFINE FILTERS (WHAT/WHY)
  - ILLUSTRATE FILTER IMPLEMENTATION
- IDENTIFICATION OF "CREDIBLE" PROPULSION/POWER CONCEPTS (SUMMARY MATRIX)
- PROS AND CONS OF CREDIBLE SYSTEMS
- SUMMARY
- SUPPORTING INFORMATION (APPENDIX)

STUDY PARTICIPANTS

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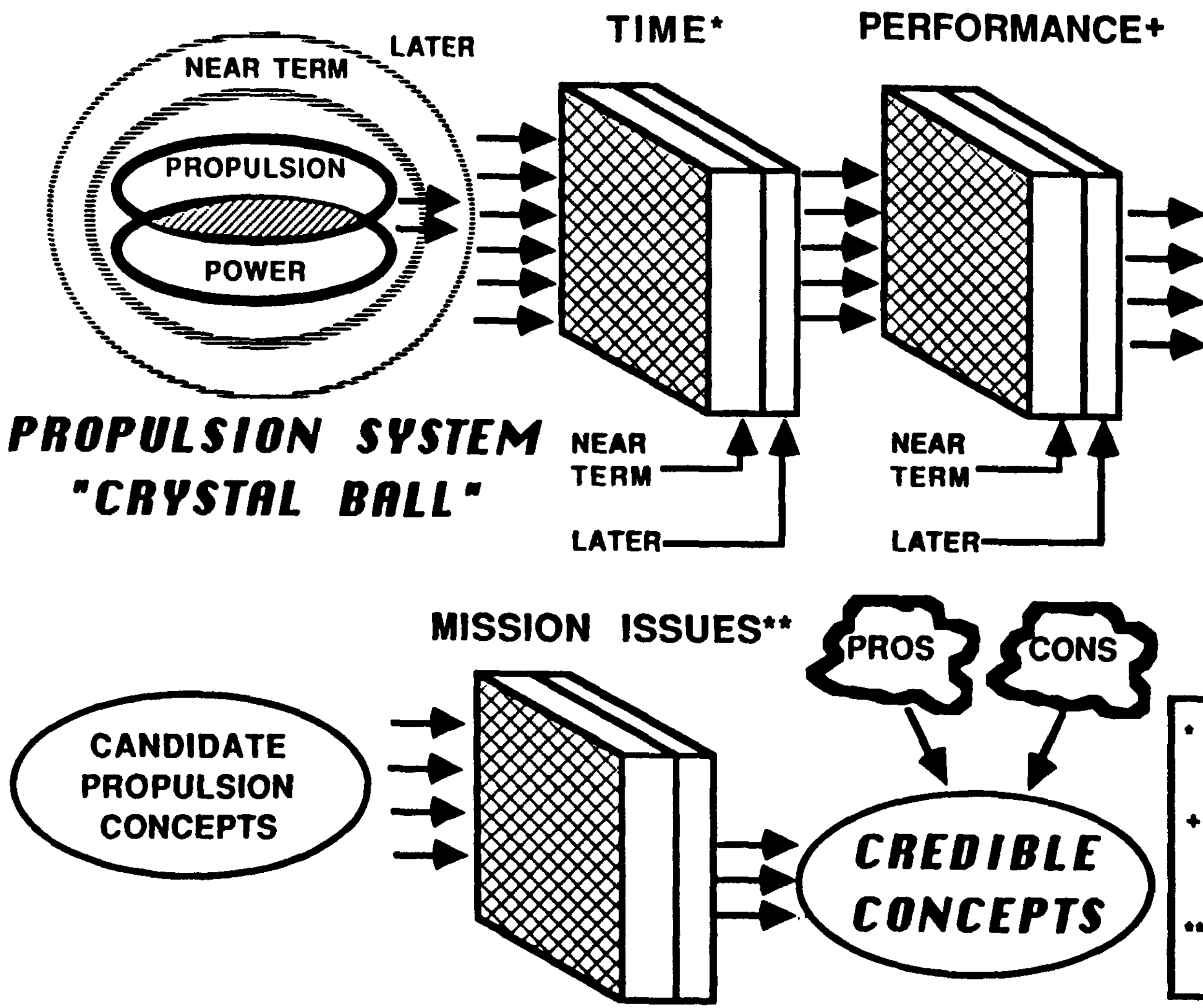
EVALUATION OF ADVANCED PROPULSION CONCEPTS

- OBJECTIVES: (1) ASSEMBLE THE NECESSARY INFORMATION TO ASSESS THE CAPABILITIES AND POTENTIAL OF VARIOUS PROPULSION AND POWER TECHNOLOGIES

  (2) CONSTRUCT A SUMMARY MATRIX OF "CREDIBLE" PROPULSION/POWER CONCEPTS FOR POSSIBLE CONSIDERATION IN FUTURE NASA MISSIONS

FILTERING APPROACH USED IN EVALUATION



* STATE-OF-THE-ART TECHNOLOGY MATURITY

\+ PARAMETRIC COMPARISON

** COMPATIBILITY WITH MISSION NEEDS

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WHY THE NEED FOR A FILTERING/ SCREENING PROCESS

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- A LOT OF PROPULSION CONCEPTS EXIST IN WHICH PRIMARY POWER IS USED EITHER:
  - (1) DIRECTLY (CHEMICAL PROPULSION, SOLAR SAILS)
  - (2) INDIRECTLY (CONVERSION TO ELECTRICITY FOR EP)
- A LOT OF POWER SOURCE/CONVERSION CONCEPTS EXIST

POWER SOURCES	POWER CONVERSION TECHNIQUES
SOLAR: PHOTON	PHOTOVOLTAIC (PV)
THERMAL	THERMOELECTRIC (TE)
NUCLEAR: RADIOISOTOPE	THERMIONIC (TI)
FISSION REACTOR:	BRAYTON
- HEAT-PIPE COOLED	RANKINE
- FLUID COOLED	STIRLING
- LIQUID	MHD:
- GAS:	- SEEDED GAS
- SOLID CORE	- LIQUID METAL
- PARTICLE BED	ELECTROSTATIC (ES)
- GAS CORE	MAGNETIC INDUCTION
FUSION REACTOR:	
- MAGNETIC	
- INERTIAL	

- DISCRIMINATOR FOR "CREDIBLE" VS. "NOT-SO-CREDIBLE" CONCEPTS REQUIRED

FILTER DEFINITION

- FILTERS USED:

  (1) TIME - OPERATIONAL DATE FOR SYSTEM AVAILABILITY, TECHNOLOGY MATURITY/READINESS

  LEVELS: (1) SIGNIFICANT GROUND/FLT. TEST/DEMO
  (2) LABORATORY DEVICE
  (3) DESIGN CONCEPT/IDEA

  NEAR TERM: (1), (2)
  LATER: (2), (3)

- STATE-OF-THE-ART DATA (AVAILABLE/PROJECTED) FOR CANDIDATE PROPULSION SYSTEMS AND KEY TECHNOLOGIES (EP AND POWER) HAS BEEN ASSEMBLED AND USED IN OUR EVALUATION

FILTER DEFINITION
(Continued)

(2) PERFORMANCE COMPARISON - PROPULSION/POWER SYSTEM PERFORMANCE PARAMETERS. THESE ARE CALCULATED FROM ASSEMBLED DATA.

(3) MISSION/PROPULSION SYSTEM COMPATIBILITY

- COMPARE "GENERIC" MISSION REQUIREMENTS AND PROPULSION SYSTEM CHARACTERISTICS
- SHORT TRIP TIMES AND HIGH PAYLOAD FRACTION ARE TWO FAVORABLE REGIONS OF PERFORMANCE
- THESE REGIONS ARE ALSO DEFINED USING DATA FROM PREVIOUS MISSION STUDIES USING HIGH PERFORMANCE SYSTEMS

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POWER/PROPULSION COMPONENTS

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	POWER	PROPULSION
SOLAR		ELECTRIC PROPULSION (EP)
	PHOTOVOLTAIC (PV)	RESISTOJET, ARCJET, MICROWAVE THRUSTER
	DYNAMIC (SD)	ION
		PULSED ELECTROTHERMAL THRUSTER (PET)
NUCLEAR		MAGNETOPLASMADYNAMIC THRUSTER (MPD)
ISOTOPE	THERMOELECTRIC (TE)	PULSED INDUCTIVE THRUSTER (PIT)
	DYNAMIC	RAIL GUN
		MASS DRIVER
FISSION	THERMOELECTRIC (TE)	
	THERMIONIC (TI)	SOLAR SAIL
	DYNAMIC	SOLAR THERMAL ROCKET (STR)
		LASER THERMAL ROCKET (LTR)
FUSION	DYNAMIC	
	ELECTROSTATIC (ES)	NUCLEAR FISSION
	INDUCTION	SOLID CORE ROCKET (SCR)
		GAS CORE ROCKET (GCR)
MASS	DYNAMIC	
ANNIHILATION		NUCLEAR FUSION
		MAGNETIC CONFINEMENT FUSION (MCF)
		-TOKAMAK FUSION ROCKET (TFR)
		INERTIAL CONFINEMENT FUSION (ICF)
		-INERTIAL FUSION ROCKET (IFR)
		-LIVERMORE IFR CONCEPT (VISTA)
		MASS ANNIHILATION ROCKET (MAR)

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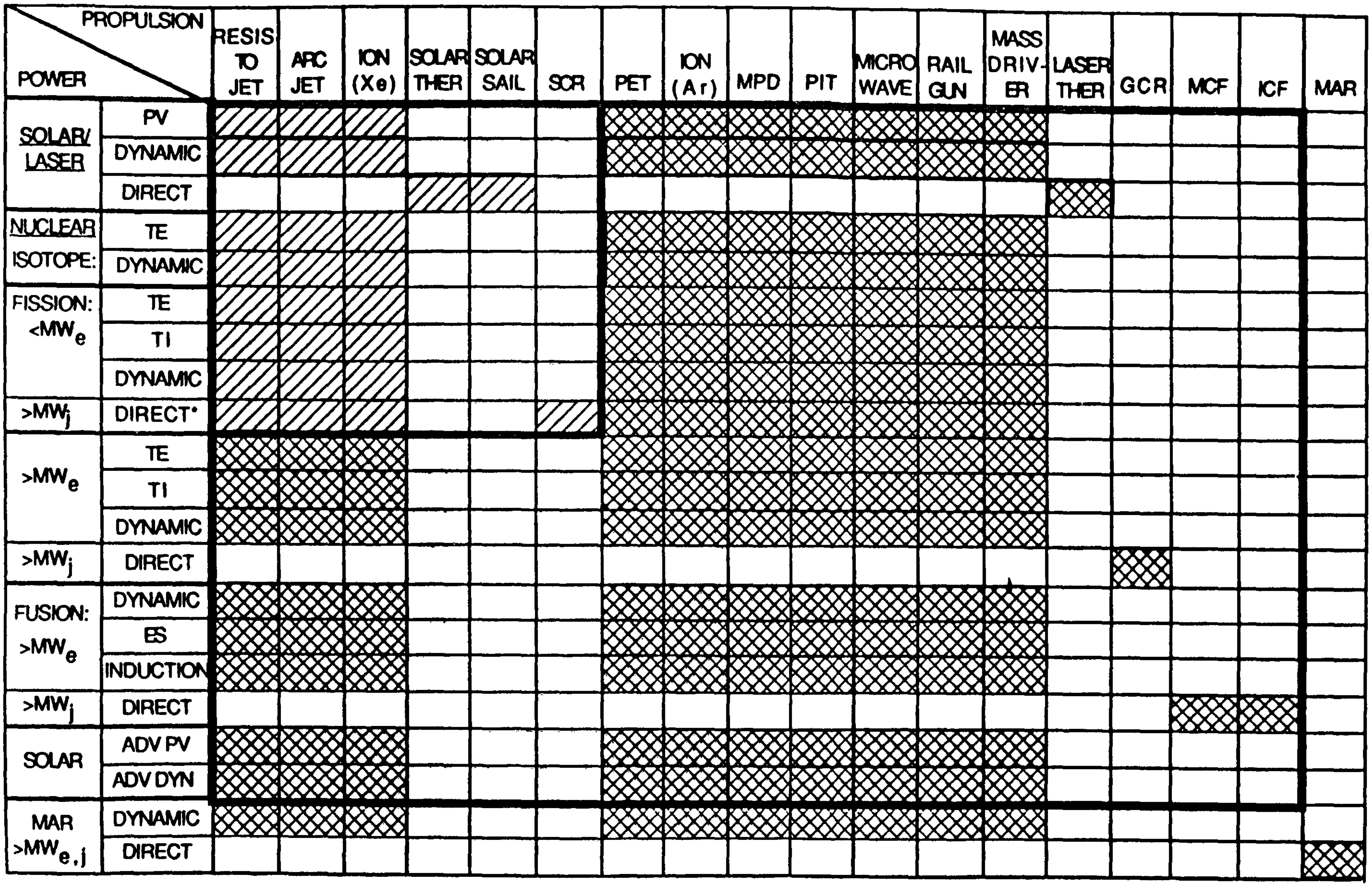
POSSIBLE PROPULSION/POWER CONCEPTS FILTERED ACCORDING TO TIME



*NOTE: DIRECT NUCLEAR ROCKETS CAN BE INTEGRATED WITH POWER CONVERSION AND EP TO OBTAIN A "HYBRID" SYSTEM WITH THE ADVANTAGES OF BOTH SYSTEMS.

NEAR TERM (10 - 15 YEARS)   LATER TERM (> 15 YEARS)

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KEY PERFORMANCE PARAMETERS

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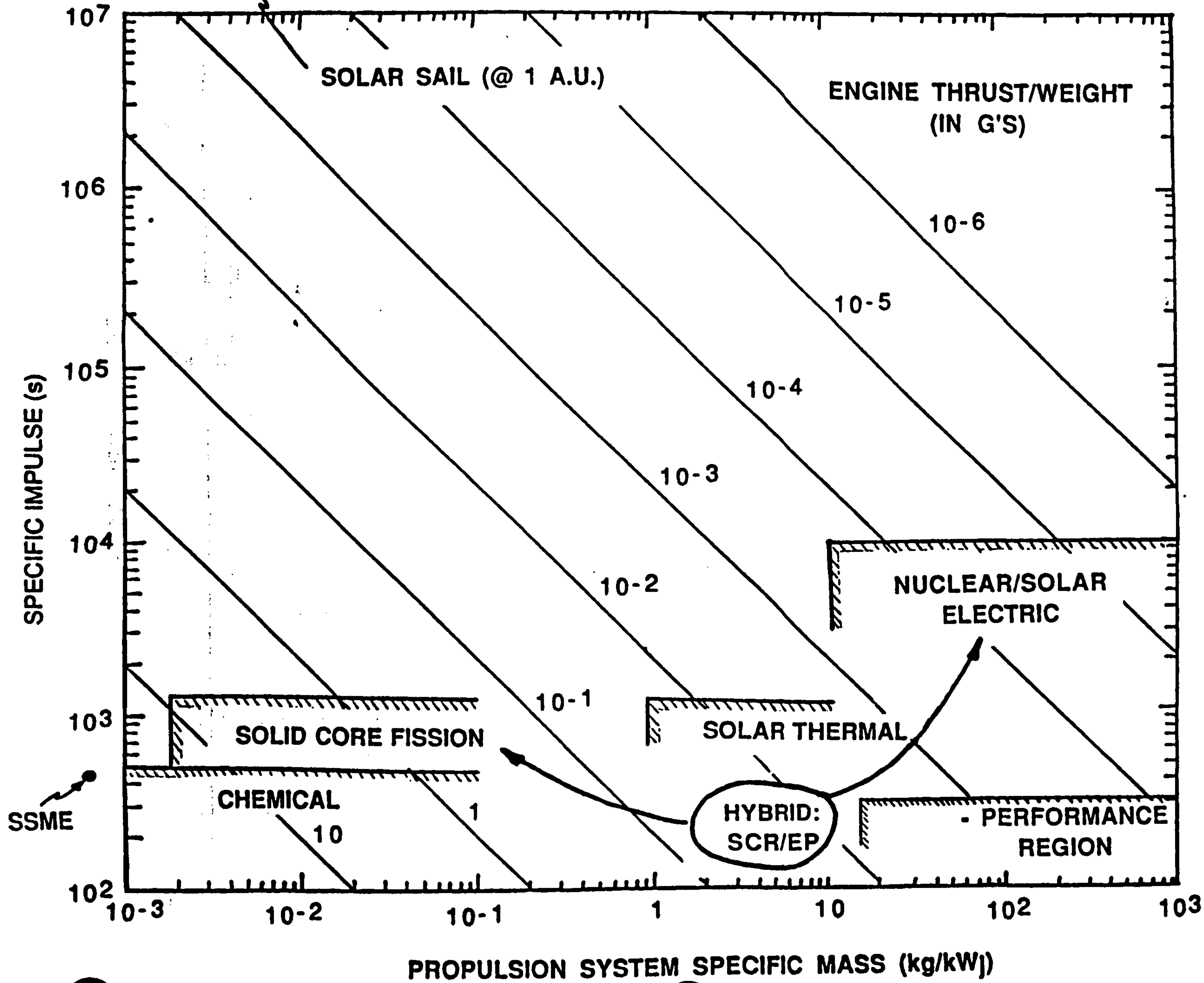
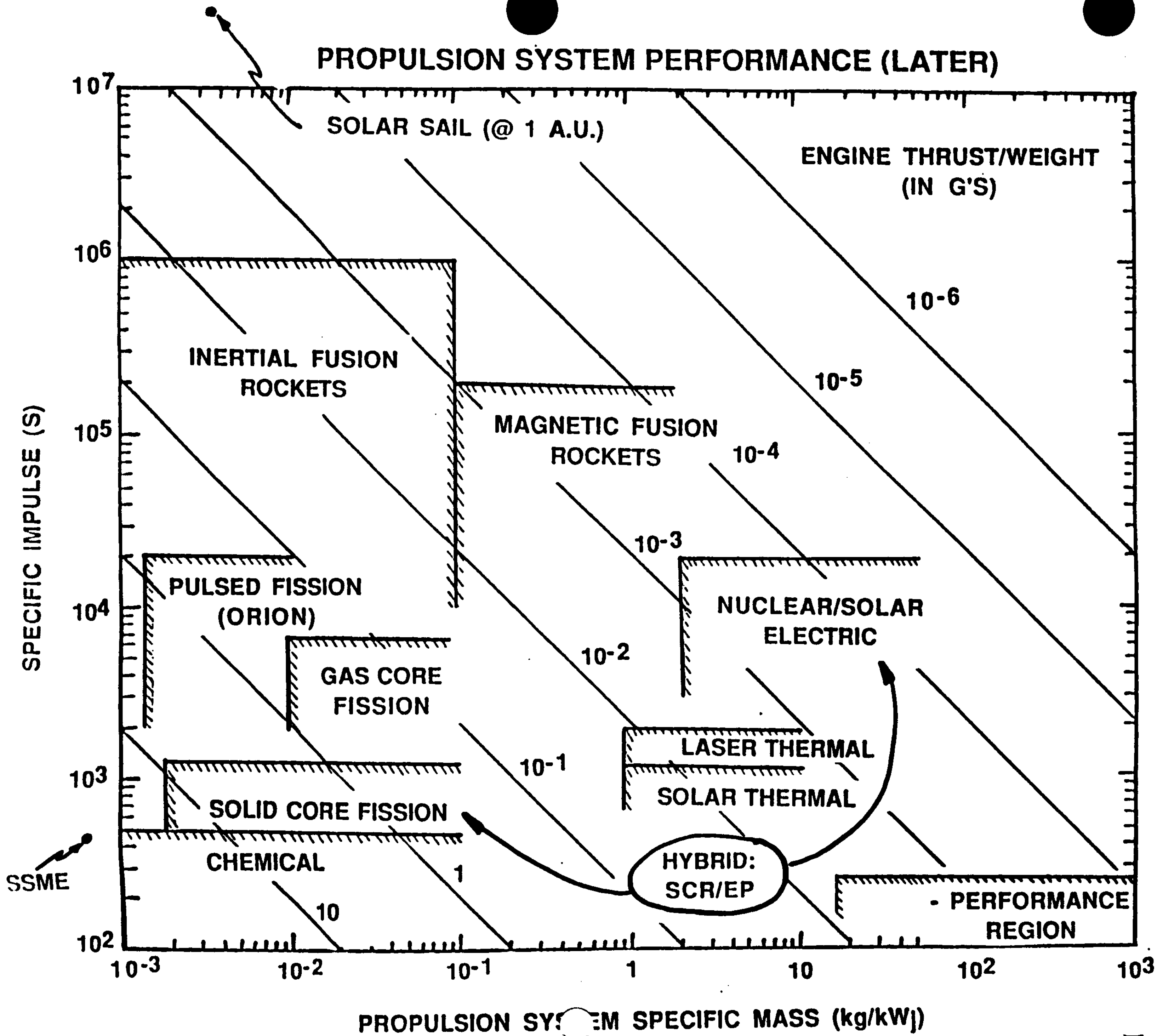
PARAMETER	GOODNESS	IMPORTANCE
SPECIFIC IMPULSE (SECONDS)	HIGH	FUEL EFFICIENCY
SPECIFIC MASS (RECIPROCAL OF SPECIFIC POWER, kg/kW_j)	HIGH	ENGINE POWER PRODUCING CAPABILITY
ENGINE THRUST/WEIGHT	HIGH	PROPULSION SYSTEM ACCELERATION CAPABILITY

PROPULSION SYSTEM PERFORMANCE (NEAR TERM)



PROPULSION SYSTEM PERFORMANCE (LATER)



BASIS FOR PERFORMANCE COMPARISON

COMPARISON WITH:

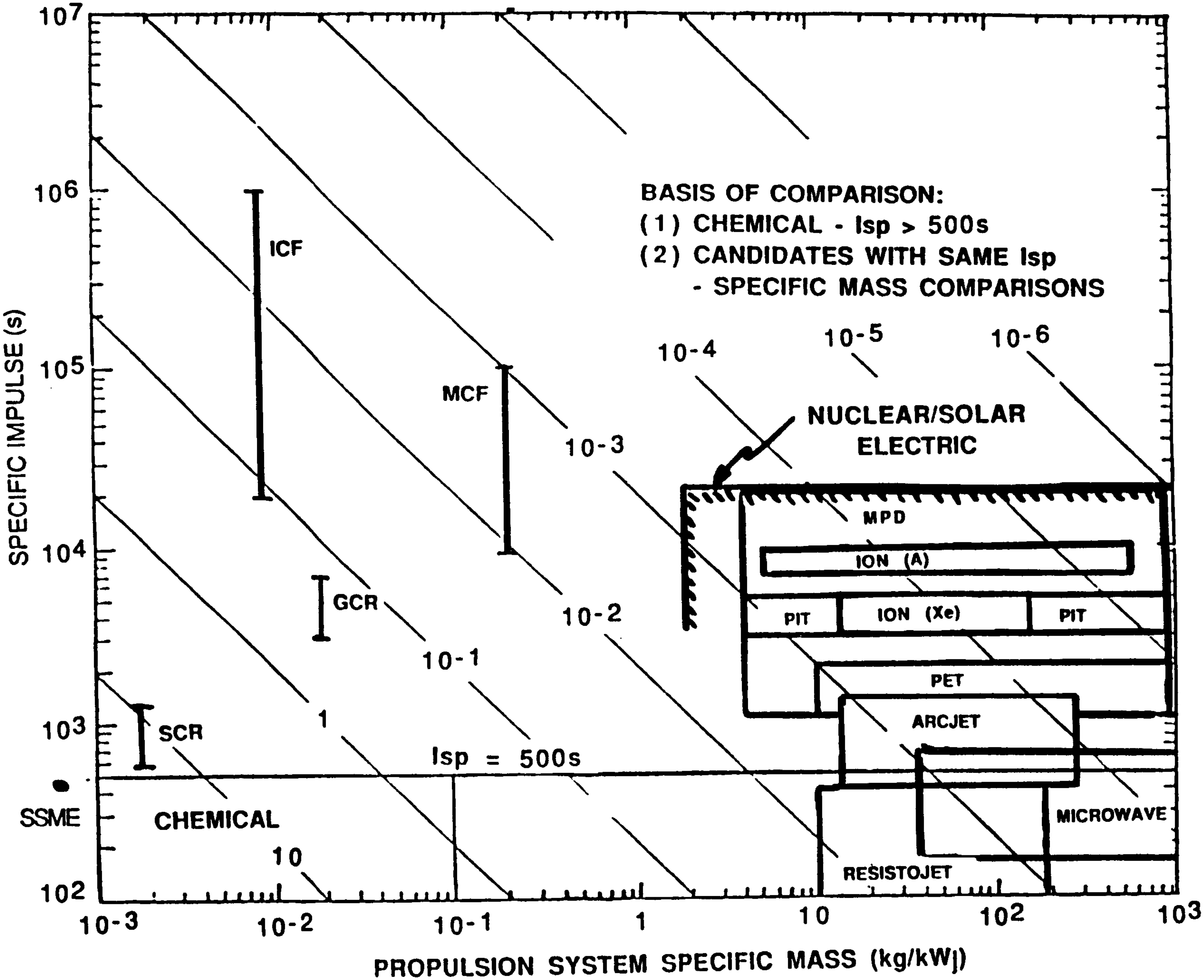
- CHEMICAL PROPULSION

  SYSTEMS WITH SPECIFIC IMPULSE LESS THAN OR EQUAL TO CHEMICAL STATE OF THE ART CAN BE ELIMINATED FOR PRIMARY PROPULSION APPLICATIONS.

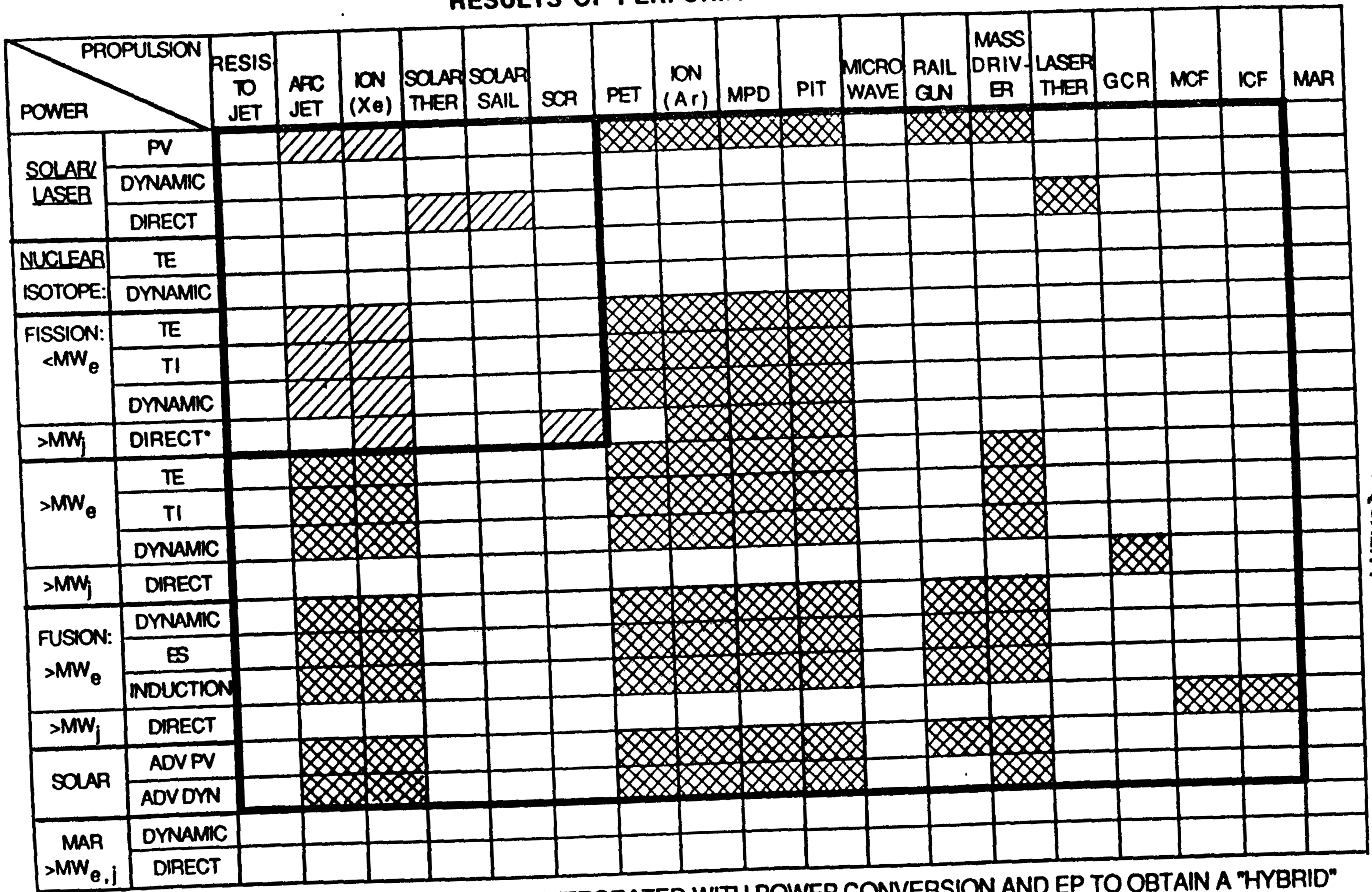
- SIMILAR PROPULSION SYSTEMS

  FOR PROPULSION SYSTEMS WITH COMPARABLE SPECIFIC IMPULSE CAPABILITY, THOSE CONCEPTS WITH SIGNIFICANTLY HIGHER SPECIFIC MASS CAN ALSO BE ELIMINATED.

PROPULSION SYSTEM PERFORMANCE (LATER)



RESULTS OF PERFORMANCE FILTER



*NOTE: DIRECT NUCLEAR ROCKETS CAN BE INTEGRATED WITH POWER CONVERSION AND EP TO OBTAIN A "HYBRID" SYSTEM WITH THE ADVANTAGES OF BOTH SYSTEMS.

NEAR TERM (10 - 15 YEARS)

LATER TERM (> 15 YEARS)

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MISSION/PROPULSION SYSTEM COMPATIBILITY CONSIDERATIONS

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	MISSION CATEGORY	PROPULSION SYSTEM FEATURES	EXAMPLE SYSTEMS	MISSION IMPACT
NEAR	SPRINT	LOW I_{SP}/HIGH ENGINE THRUST/WEIGHT	CHEMICAL WITH AEROBRAKE	HIGH PROPELLANT FRACTION /MODERATE-SHORT TRIPTIME
	CARGO	HIGH I_{SP}/LOW ENGINE THRUST/WEIGHT	NUCLEAR ELECTRIC PROPULSION	HIGH PAYLOAD FRACTION /LONG TRIPTIME
LATER	SPRINT + CARGO	HIGH I_{SP}/HIGH ENGINE THRUST/WEIGHT	GAS CORE ROCKET /FUSION ROCKET	HIGH PAYLOAD FRACTION /SHORT TRIPTIME

MISSION ANALYSIS CONSIDERATIONS

- DETAILED MISSION ANALYSIS REQUIRES ADDITIONAL DATA (SUCH AS DESTINATION AND PAYLOAD) TO PROVIDE AN ACCURATE ASSESSMENT OF TRIP TIME AND PAYLOAD FRACTION.

- A COMPARISON OF SYSTEM MISSION PERFORMANCE REQUIRES CONSISTENT GROUND RULES AND CONDITIONS.

- SYSTEM MISSION PERFORMANCE IS LIMITED BY SPECIFIC IMPULSE AND SPECIFIC POWER CAPABILITIES. WITHIN THESE LIMITS, A RANGE OF PERFORMANCE (TRIP TIME, PAYLOAD FRACTION) CAN BE ATTAINED.

- PRIMARY PROPULSION CAN BE APPLIED TO A RANGE OF MISSIONS, FROM ORBITAL TRANSFER TO INTERPLANETARY EXPLORATION. A CHANGE IN MISSION DEMANDS MAY ALTER THE BOUNDARIES BETWEEN CARGO AND COURIER REGIONS.

- PROPULSION SYSTEM POWER LEVELS ARE NOT FIXED BY SPECIFIC IMPULSE AND SPECIFIC POWER. SYSTEM POWER REQUIREMENTS HAVE NOT BEEN FILTERED INTO THIS STUDY.

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REFERENCE ENGINE DESIGNS

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	SYSTEM	DESTINATION	TRIP TIME+	PAYLOAD MASS* / INITIAL MASS	APPENDIX REFERENCES
1)	NEP/ION (SP-100)	NEPTUNE	12 YEARS (PROBE-1 WAY)	0.10	8
2)	LTR (1 MW)	LEO-GEO	28 DAYS	0.44	4
3)	STR (2 MW)	LEO-GEO	30 DAYS	0.54	6
4)	SEP/ION (300 KW_e)	MOON	370 DAYS	0.57	7
5)	PEGASUS/MPD (8.5 MW_e)	MARS	1000 DAYS	0.45	13
6)	NEP/ION (300 kW_e)	NSO-MARS	770 DAYS (CARGO-1 WAY)	0.44	25
7)	NEP/ION (3 MW_e)	MOON-MARS	413 DAYS (CARGO- 1 WAY)	0.54	25

\+ ROUND TRIP UNLESS OTHERWISE INDICATED
* TO DESTINATION ONLY

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REFERENCE ENGINE DESIGNS (CONTINUED)

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	SYSTEM	DESTINATION	TRIP TIME+	PAYLOAD MASS* / INITIAL MASS	APPENDIX REFERENCES
8)	SCR/NERVA (5000 MW)	MARS	720 DAYS	0.23	40
9)	NEP/ION (200 MW_e)	MARS	~ 180 DAYS	0.0	39
10)	GCR (8500 MW)	MARS	80 DAYS / 280 DAYS	0.075 / 0.25	15
11)	ORION (43,000 MW)	MARS	250 DAYS	0.23	16
12)	TFR (7500 MW)	MARS	77 DAYS	0.06	17
13)	VISTA (225,000 MW)	MARS	100 DAYS	0.017	20
14)	IFR (200,000 MW)	MARS / PLUTO	55 DAYS / 20 MONTHS	0.26 / 0.14	17,18

\+ ROUND TRIP UNLESS OTHERWISE INDICATED

* TO DESTINATION ONLY

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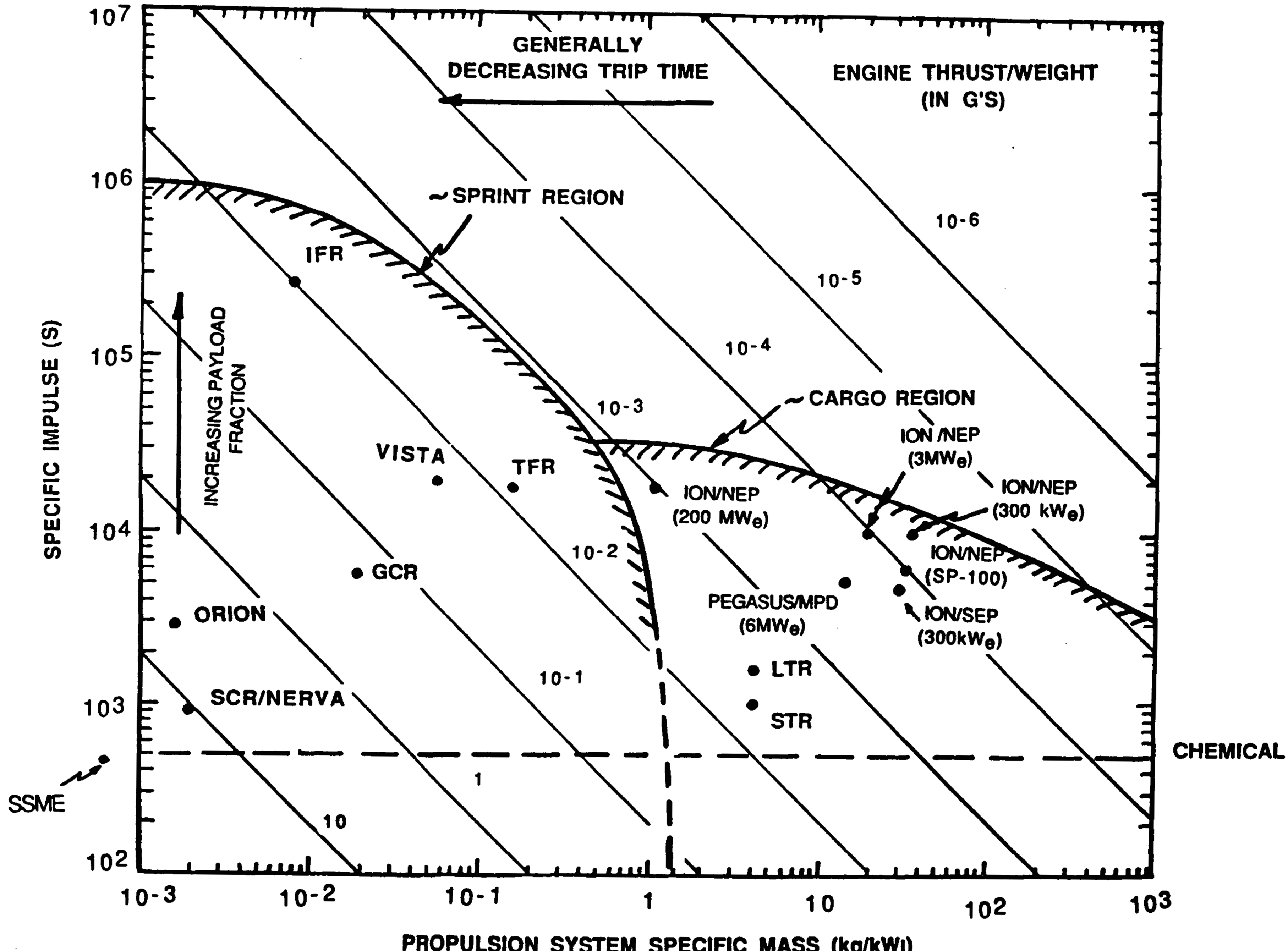
MISSION REQUIREMENTS REGIONS



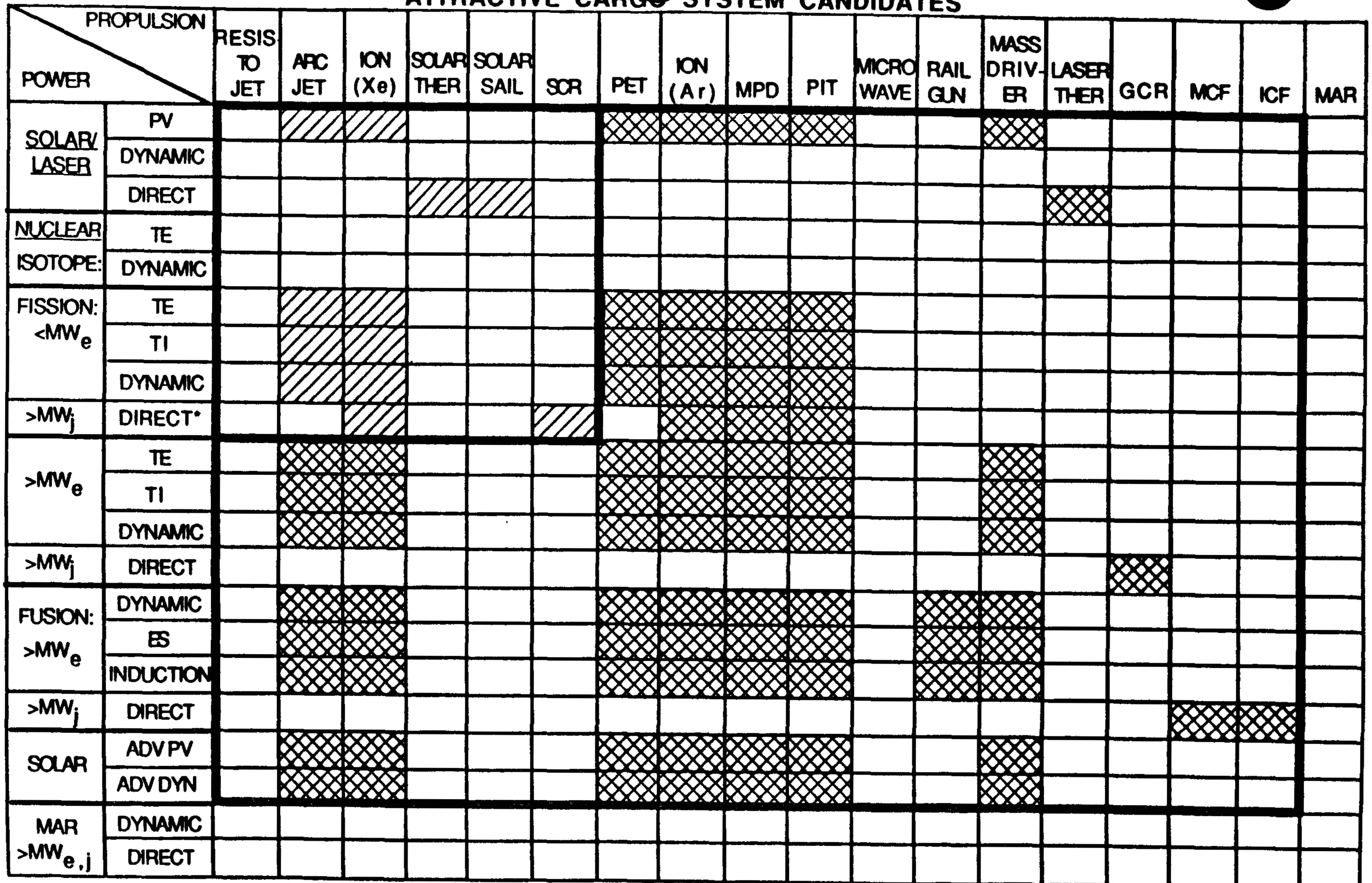
ATTRACTIVE SPRINT SYSTEM CANDIDATES

POWER	PROPULSION	RESISTO JET	ARC JET	ION (Xe)	SOLAR THER	SOLAR SAIL	SCR	PET	ION (Ar)	MPD	PIT	MICRO WAVE	RAIL GUN	MASS DRIVER	LASER THER	GCR	MCF	ICF	MAR
SOLAR/ LASER	PV																		
	DYNAMIC																		
	DIRECT																		
NUCLEAR ISOTOPE:	TE																		
	DYNAMIC																		
FISSION: $<MW_e$	TE																		
	TI																		
	DYNAMIC																		
$>MW_j$	DIRECT*																		
$>MW_e$	TE																		
	TI																		
	DYNAMIC																		
$>MW_j$	DIRECT																		
FUSION: $>MW_e$	DYNAMIC																		
	ES																		
	INDUCTION																		
$>MW_j$	DIRECT																		
SOLAR	ADV PV																		
	ADV DYN																		
MAR $>MW_{e,j}$	DYNAMIC																		
	DIRECT																		

*NOTE: DIRECT NUCLEAR ROCKETS CAN BE INTEGRATED WITH POWER CONVERSION AND EP TO OBTAIN A "HYBRID" SYSTEM WITH THE ADVANTAGES OF BOTH SYSTEMS.

NEAR TERM (10 - 15 YEARS)   LATER TERM (> 15 YEARS)

ATTRACTIVE CARGO SYSTEM CANDIDATES



*NOTE: DIRECT NUCLEAR ROCKETS CAN BE INTEGRATED WITH POWER CONVERSION AND EP TO OBTAIN A "HYBRID" SYSTEM WITH THE ADVANTAGES OF BOTH SYSTEMS.

NEAR TERM (10 - 15 YEARS)   LATER TERM (> 15 YEARS)

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ADDITIONAL CONSIDERATIONS FOR SURVIVING SYSTEMS

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SOLAR ELECTRIC PROPULSION (SEP)

ADVANTAGES

- SPACE SOLAR POWER IS A WELL-DEVELOPED TECHNOLOGY
- NO RADIOACTIVITY

DISADVANTAGES

- POWER LEVEL DECREASES WITH INCREASING DISTANCE FROM SUN
- DEGRADATION IN VAN ALLEN RADIATION BELT
- SOLAR ENERGY COLLECTION REQUIRES LARGE SURFACE AREA, PARTICULARLY AT HIGH POWER
- OPERATION IN PLANETARY SHADOW REQUIRES ENERGY STORAGE (A MASS PENALTY) OR REDUCES AVERAGE POWER
- SUN TRACKING AND ARRAY POINTING ADD SYSTEM AND TRAJECTORY COMPLEXITY
- LOW THRUST/WEIGHT REQUIRES EXTENDED PERIODS OF OPERATION
- RELIABILITY DEPENDS ON COLLECTOR, POWER CONVERSION, AND THRUSTERS, IMPOSING STRINGENT DESIGN REQUIREMENTS

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ADDITIONAL CONSIDERATIONS FOR SURVIVING SYSTEMS

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NUCLEAR ELECTRIC PROPULSION (NEP)

ADVANTAGES

- COMPACT POWER SOURCE
- POWER LEVEL INDEPENDENT OF ENVIRONMENT (EXAMPLE: SUN, RADIATION BELT)
- FAVORABLE MASS SCALING TO HIGH POWERS
- ADVANTAGEOUS FOR MISSIONS WHERE REACTOR IS PART OF THE PAYLOAD
- HYBRID SYSTEM SHOWS ADVANTAGES OVER DIRECT NUCLEAR OR NEP

DISADVANTAGES

- RADIATION REQUIRES SHIELDING FOR MEN AND INSTRUMENTS
- RADIATORS REQUIRE LARGE SURFACE AREA, PARTICULARLY AT HIGH POWER
- LOW THRUST/WEIGHT REQUIRES EXTENDED PERIODS OF OPERATION
- RELIABILITY DEPENDS ON REACTOR, POWER CONVERSION, AND THRUSTERS, IMPOSING STRINGENT DESIGN REQUIREMENTS

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ADDITIONAL CONSIDERATIONS FOR SURVIVING SYSTEMS

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SOLID CORE THERMAL ROCKET

ADVANTAGES

- TECHNOLOGY DEVELOPED (NERVA PROGRAM: 10 YEARS/$1.5 BILLION INVESTED)
- OPERATION OF 19 REACTORS DEMONSTRATED:
  - CORE DESIGN
  - NON-NUCLEAR COMPONENTS
  - A VARIETY OF CONTROL/SAFETY SYSTEMS
- FLIGHT ENGINE DESIGNED BUT NOT TESTED
- FACTOR OF 2 ADVANTAGE IN I_{sp} OVER CHEMICAL PROVIDES ROBUSTNESS (LOWER INITIAL MASS, SHORTENED TRIP TIME, PROPULSIVE BRAKING OPTION)

DISADVANTAGES

- LARGER ENGINE MASS THAN CHEMICAL
- RADIATION POSES PROBLEMS FOR LOCAL MANNED OPERATION IMMEDIATELY AFTER SHUTDOWN
- "SPECIAL" DOCKING FACILITIES REQUIRED
- RADIATION RAISES ENVIRONMENTAL CONCERNS ABOUT ENGINE SAFETY DURING STARTUP IN LEO AND THE USE OF AEROBRAKING

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ADDITIONAL CONSIDERATIONS FOR SURVIVING SYSTEMS

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HYBRID NUCLEAR THERMAL ROCKET

ADVANTAGES

- HYBRID POSSESSES ADVANTAGES OF BOTH HIGH THRUST/WEIGHT AND HIGH PAYLOAD CAPABILITY SYSTEMS
- TECHNOLOGIES DEMONSTRATED IN NERVA/EP PROGRAMS
- DUAL-MODE OPERATION (FOR THRUST/POWER GENERATION) MORE EFFECTIVELY UTILIZES INHERENT ENERGY POTENTIAL OF ENGINE
- AT HIGH POWER LEVELS, CHARACTERISTICS OF ADVANCED FISSION ROCKETS (GCR) MAY BE POSSIBLE (SHORT TRIP TIME)

DISADVANTAGES

- POWER CONVERSION SYSTEM ADDS WEIGHT TO SOLID CORE ROCKET
- HIGH POWER (MWe) OPERATION CAN SIGNIFICANTLY ALTER REACTOR DESIGN
- TECHNOLOGICAL FEASIBILITY REMAINS TO BE DEMONSTRATED

ADDITIONAL CONSIDERATIONS FOR SURVIVING SYSTEMS

SOLAR/LASER THERMAL PROPULSION

ADVANTAGES

- NO POWER SOURCE ON BOARD
- NO RADIOACTIVITY
- SOLAR THERMAL CONCEPT EXPERIMENTALLY DEMONSTRATED

DISADVANTAGES

- POINTING AND TRACKING REQUIREMENTS ADD SYSTEM AND TRAJECTORY COMPLEXITY
- LASER SOURCE REQUIRES LARGE INPUT POWER LEVELS
- MULTIPLE LASERS REQUIRED FOR CONTINUOUS ACCELERATION NEAR EARTH
- LASER CONCEPT STILL EXPERIMENTAL
- SOLAR THERMAL I_{SP} LIMITED BY MATERIALS

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ADDITIONAL CONSIDERATIONS FOR SURVIVING SYSTEMS

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GAS CORE THERMAL ROCKET

ADVANTAGES

- OPERATION OF URANIUM FUEL IN HIGH TEMPERATURE GASEOUS STATE ALLOWS HIGH I_{sp} OPERATION WITH SIMULTANEOUS HIGH THRUST
- ONLY FISSION ROCKET CONCEPT IDENTIFIED CAPABLE OF 60 DAY ROUND TRIP COURIER MISSION TO MARS
- THE GCR'S ABILITY TO DISPOSE OF RADIOLOGICALLY DANGEROUS FISSION PRODUCTS IN SPACE (WITH WIDE DISPERSAL) SIGNIFICANTLY REDUCES POST-SHUTDOWN HAZARDS
- BOTH CLOSED CYCLE (NO FUEL LOSS) AND OPEN CYCLE CONCEPTS HAVE BEEN STUDIED ANALYTICALLY AND EXPERIMENTALLY (NERVA PROGRAM)
- KEY OPERATIONAL FEATURES OF BOTH CONCEPTS DEMONSTRATED IN EXPERIMENTS (CRITICALITY OF GASEOUS UF_6 DEMONSTRATED)

DISADVANTAGES

- LOSS OF URANIUM (~ 1/4 - 1% OF PROPELLANT FLOW RATE) IN OPEN CYCLE CONCEPT INCREASES FUEL COSTS AND LEADS TO ENVIRONMENTAL CONCERNS FOR SYSTEMS USED IN NEAR EARTH ORBIT
- SIGNIFICANT CONCEPT TESTING UNDER NUCLEAR CONDITIONS REMAINS TO BE DONE

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ADDITIONAL CONSIDERATION FOR SURVIVING SYSTEMS

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FUSION PROPULSION (MAGNETIC/INERTIAL)

ADVANTAGES

- ABUNDANT FUSION FUEL (10 TRILLION TONS OF DEUTERIUM IN EARTH'S OCEANS)
- HIGH I_{sp} INHERENT IN FUSION-GRADE PLASMAS
- THE HIGH SPECIFIC POWER/IMPULSE OF IFR'S CAN RESULT IN ROUND TRIP TRAVEL TIMES TO PLUTO OF < 20 MONTHS
- ADVANCED-FUEL FUSION SYSTEMS (USING DEUTERIUM AND HELIUM (He-3)) PRODUCE MINIMAL NEUTRON RADIATION (LEADS TO LOWER INITIAL WEIGHTS FOR MFR'S AND HIGHER SPECIFIC POWERS)
- THE USE OF "SPIN POLARIZED" DHe^3 FUEL MAY RESULT IN "CLEAN" FUSION ROCKETS (NO NEUTRON RADIATION)
- NASA CAN CAPITALIZE ON LARGE RESEACH EFFORT IN FUSION (~ $1 BILLION WORLDWIDE/YEAR)

DISADVANTAGES

- MAGNETIC FUSION SYSTEMS REQUIRE HEAVY SUPERCONDUCTING MAGNETS AND COMPLICATED PLASMA EXTRACTION TECHNIQUES
- INERTIAL FUSION SYSTEMS REQUIRE DRIVERS WITH HIGH EFFICIENCY, ENERGY, OPERATING TEMPERATURE AND REP RATE (DRIVERS POSSESSING ALL OF THESE FEATURES DO NOT CURRENTLY EXIST)
- PROPULSION-SPECIFIC (LIGHTWEIGHT) TECHNOLOGIES ARE AT LEAST TWO DECADES AWAY

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ADDITIONAL CONSIDERATIONS FOR SURVIVING SYSTEMS

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PULSED FISSION ROCKET (ORION)

ADVANTAGES

- HIGH I_{sp} OPERATION POSSIBLE WITHOUT MAGNETIC NOZZLE DUE TO SHORT INTERACTION TIME OF PULSE WITH VEHICLE
- CAN UTILIZE ESTABLISHED TECHNOLOGIES
- CONCEPT FEASIBILITY DEMONSTRATED USING CONSECUTIVE CHEMICAL EXPLOSIONS

DISADVANTAGES

- CRITICALITY REQUIREMENTS LIMIT MINIMUM BOMBLET YIELD (~0.01 kTONS TNT)
- LARGE VEHICLES REQUIRED TO EFFICIENTLY HANDLE ENERGY
- POLITICAL/ENVIRONMENTAL PROBLEMS ASSOCIATED WITH FISSION BOMB USAGE IN SPACE
- ENGINE TESTING VIRTUALLY RULED OUT DUE TO NUCLEAR TEST BAN TREATY

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SUMMARY

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- THIS TASK HAS BEEN RESPONSIVE TO THE REQUEST TO EVALUATE ADVANCED PROPULSION/POWER CONCEPTS. AN EXTENSIVE DATA BASE HAS BEEN ASSEMBLED AND USED TO DERIVE TWO SUMMARY MATRICES (ONE SPRINT, ONE CARGO) OF CREDIBLE PROPULSION SYSTEMS

- CONCEPTS WERE IDENTIFIED USING THREE FILTERS

  (1) TIME: NEAR TERM
  LATER TERM

  (2) SYSTEM PERFORMANCE: COMPARISONS MADE TO CHEMICAL PROPULSION AND TO OTHER ADVANCED PROPULSION SYSTEMS

  (3) MISSION COMPATIBILITY: SYSTEM CAPABILITIES FOR MISSION GOALS (HIGH PAYLOAD FRACTION, SHORT TRIP TIME)

- FURTHER SCREENING IS MISSION SPECIFIC; HOWEVER, SOME POTENTIAL ADVANTAGES/DISADVANTAGES IN THIS REGARD HAVE BEEN PRESENTED

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SUPPORTING INFORMATION
(APPENDIX)

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- REFERENCES
- DEFINITIONS OF KEY PROPULSION SYSTEM PERFORMANCE PARAMETERS
- STATE-OF-THE-ART DATA SUMMARY (FOR VARIOUS PROPULSION SYSTEMS AND TECHNOLOGIES)
- PROS AND CONS CHARTS FOR EP AND POWER SYSTEM TECHNOLOGIES
- ORGANIZATIONS (GOVERNMENT AGENCIES/INDUSTRY/UNIVERSITIES) ADVOCATING/STUDYING DIFFERENT CONCEPTS
- DATA USED IN SYSTEMS ASSESSMENT

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KEY PROPULSION SYSTEM PERFORMANCE PARAMETERS

ASAO

- SPECIFIC IMPULSE: $I_{sp}(s)$; $V_{ex} \equiv g_o I_{sp}$ = EXHAUST VELOCITY
- THRUST: $F(N) = \dot{m}_p(kg/s) g_o I_{sp}$; $\dot{m}_p$ = PROPELLANT FLOW RATE
- JET POWER: $P_{jet}(kW_j) = 4.9 \times 10^{-3}\ F(N)\ I_{sp}(s)$
- ENGINE SPECIFIC POWER
  - DIRECT THRUST: $\alpha_p(kW_j/kg) \equiv P_{jet}/M_e$; M_e = TOTAL ENGINE SYSTEM MASS
  - EP SYSTEM: $\alpha_p(kW_j/kg) \equiv \eta_{ej} P_e / M_e \equiv \eta_{ej}/\alpha_m (kg/kW_e)$
  - BEAMED ENERGY SYSTEM: $\alpha_p(kW_j/kg) \equiv (\eta_{elj} P_l$ OR $\eta_{sj} P_s)/M_e$

WHERE -

$P_{e,l,s}$ = ELECTRICAL, LASER, OR SOLAR POWER
$\eta_{e,l,sj}$ = CONVERSION EFFICIENCY OF ELECTRICAL, LASER, OR SOLAR POWER TO JET POWER

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KEY PROPULSION SYSTEM PERFORMANCE PARAMETERS (Continued)

ASAO

- THRUST-TO-ENGINE WEIGHT RATIO

  $$F/g_o M_e \equiv \frac{2000}{g_o^2} \quad \frac{\alpha_p (kW_j/kg)}{I_{sp}(s)}$$

- GROSS PROPULSION SYSTEM POWER

  $P_g = P_{jet}/\eta_j$ FOR DIRECT THRUST SYSTEMS

  $= P_e/\eta_e$ FOR ELECTRIC PROPULSION SYSTEMS

  $= P_l$ OR P_s FOR LASER OR SOLAR BEAMED ENERGY SYSTEMS

  $\eta_{e,j}$ ARE THE ELECTRICAL AND JET POWER CONVERSION EFFICIENCIES

- PROPULSION SYSTEM REQUIRED OPERATIONAL LIFETIME: τ_{op}

NOTE: THE FUNCTIONAL LIFETIME OF AN ENGINE CAN BE LESS THAN τ_{op} IMPLYING A NEED FOR A MULTIPLE ENGINE PROPULSION SYSTEM

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NEAR TERM PROPULSION SYSTEM STATE-OF-THE-ART DATA SUMMARY

ASAO

PROPULSION CONCEPT / PERFORMANCE *(DEMONSTRATED)*	I_{sp} (ks)	α_m (kg/kW_e)	$\eta_{e,i,sj}$ (%)	α_p (kWj/kg)	F/g_oM_e	F/N[a] (N)	P_{jet}/N[a] (kWj)	P_g (MW)	τ_{op} (hrs.)
CHEMICAL: SHUTTLE MAIN ENGINES	0.455	NA[b]	NA	1635	75	2.08×10^6	4.64×10^6	≥5000	7.5
DIRECT FISSION: SCR/NERVA PROGRAM- Ref.2	0.85 -1.2	NA	NA	~120 -435	3.3 -10.7	2.45×10^5 -1.12×10^6	9.12×10^5 -3.6×10^6	1100 -5000	1.0 -10.0
LASER THERMAL (OTV-Ref.4)	0.9-1.5 (1.5)	NA	10-80 (46)	(0.26)	(3.7×10^{-3})	(63)	(466)	(1.0)	(684)
SOLAR THERMAL (OTV-Ref.6)	0.8-1.2 (0.9)	NA	(50)	(0.28)	(6.4×10^{-3})	(222)	(980)	(2.0)	(720)
SOLAR ELECTRIC:ION (LUNAR FERRY-Ref.7)	3.5-10.0 (4.5)	(26)	60-80 (75)	(3×10^{-2})	(1.4×10^{-4})	.2-10 (10/10)	3.5-500 (225/10)	(0.3)	$\leq 10^4$ ($\sim 9 \times 10^3$)
NUCLEAR ELECTRIC:ION (NEPTUNE PROBE-Ref.8)	3.5-10.0 (5.3)	(44)	60-80 (77)	(1.7×10^{-2})	(6.6×10^{-5})	.2-10 (2.84/5)	3.5-500 (74/5)	(1.4)	$\leq 10^4$ (5×10^4)
NUCLEAR ELECTRIC:ARCJET (OTV-Ref.9)	0.8-1.2 (1.0)	(30)	30-50 (45)	(7.5×10^{-3})	(1.6×10^{-4})	10 (8.4/4)	10-50 (41/4)	(1.4)	500-2000 ($\sim 3 \times 10^3$)
DUAL MODE NUCLEAR (SCR/ION-Ref.11)	(0.895) / 5.3	(34.5)	77	(64) / 2.1×10^{-2}	(1.5) / 8.4×10^{-5}	(5×10^4) / 2.84	(2.2×10^5) / 74	(263) / 0.4	NA

[a] F,P_{jet}/N - THRUST, JET POWER PER UNIT ENGINE

[b] NOT APPLICABLE OR DATA NOT AVAILABLE

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PROJECTED PERFORMANCE FOR LATER TERM PROPULSION SYSTEMS

ASAO

PROPULSION CONCEPT / PERFORMANCE [DEMONSTRATED]	I_{sp} (ks)	α_m (kg/kW_e)	$\eta_{e,I,sj}$ (%)	α_p (kWj/kg)	F/g_oM_e	F/N^a (kN)	P_{jet}/N^a (MWj)	P_g (MW)	τ_{op} (hrs.)
SOLAR SAIL (MARS CARGO MISSION-Ref.21[c])	30,600	NA[b]	NA	(292)	($2x10^{-4}$)	($3.7x10^{-2}$)	(5600)	NA	($3.7x10^4$)
NUCLEAR ELECTRIC:MPD (PEGASUS-Ref.13)	*2-5* (5)	5-10 (8.5)	*30*-50 (50)	($6x10^{-2}$)	($2.5x10^{-4}$)	0.01-0.2 (0.122)	0.25-.5 (3)	2-40 (27)	10^4-$2x10^4$ ($1.24x10^4$)
DIRECT FISSION: GCR/MMM (Ref.14)	3-7 (5.7)	NA	NA	10-50 (47)	.04-0.2 (0.17)	22-440 (220)	540-11000 (6150)	750-15000 (8500)	(109)
DUAL MODE NUCLEAR (SCR/MPD-Ref.11)	(0.98) / *5*	(11)	*50*	*(20)* / $4.6x10^{-2}$	*(0.42)* / $1.9x10^{-4}$	*(45.4)* / $2x10^{-2}$	*(218)* / 0.5	*(260)* / 4	NA
PULSED FISSION (ORION-Ref.16)	2-1.0 (2.5)	NA	NA	(470)	(3.9)	(3500)	(43,000)	(NA)	(~1)
FUSION PROPULSION:MCF (MCF/MMM-Ref.17)	10-200 (20)	NA	NA	~2-10 (5.75)	($6.3x10^{-3}$)	(60)	(6000)	(7500)	($2x10^3$)
ICF (ICF/RT PLUTO MISSION-Ref.18)	20-1000 (270)	NA	NA	(110)	($8.4x10^{-3}$)	(40)	(53,000)	(200,000)	($1.4x10^4$)

a $F, P_{jet}/N$ - THRUST, JET POWER PER UNIT ENGINE
b NOT APPLICABLE OR DATA NOT AVAILABLE
c PERFORMANCE PERAMETERS GIVEN AT 1 A.U.

NOTATION: SCR,GCR-SOLID,GAS CORE ROCKET
MCF, ICF-MAGNETIC, INERTIAL FUSION ROCKET
MMM-MANNED MARS MISSION

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DATA SUMMARY FOR NEAR/LATER TERM POWER TECHNOLOGIES

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SOURCE	SYSTEM	AVAIL	PWR RANGE (kWe)	EFF (%)	SP MASS (kg/kWe)	COMMENTS
ISOTOPE	RTG	NEAR	.01 - 1	7	120	MODULAR RTG WITH GPHS
	DIPS	NEAR	1 - 10	28	125	CBC, DUEL LOOP, GPHS
SOLAR	PV-Si	NEAR	1 - 100	6	90	SS MISSION, NO STORE, TOT STRUCT
	PV-AdvPln	NEAR	1 - 100	25	3	LeRC, NO STORAGE, MIN STRUCTURE
	PV-CONC	LATE	1 - 100	25	5	LeRC, NO STORAGE, MIN STRUCTURE
	SD-SOTA	NEAR	10 - 50	33	200	SS MISSION, NO STORE, TOT STRUCT
	SD-Adv	LATE	10 - 50	34	16	LeRC, NO STORAGE, FP STIRLING
NUCLEAR	T/E	NEAR	100 - 1000	7	40	SP-100, SiGe (GaP) CELLS
	DYNAMIC	NEAR	100 - 300	26	40	SP-100, FP STIRLING
	AMTEC	LATE	1000 - 40,000	22	4	MMWe, LIQ METAL REACTOR
	T/I	LATE	1000 - 10,000	13	5	MMWe, IN-CORE
	DYNAMIC	LATE	1000 - 50,000	21	4	MMWe, LMR, K-RANKINE
	MHD	LATE	10,000-100,000	40	5	NERVA DERIVATIVE REACTOR LeRC

NEAR - NEAR TERM TECHNOLOGY READINESS
LATE - FAR TERM TECHNOLOGY READINESS

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DATA SUMMARY FOR NEAR/LATER TERM ELECTRIC PROPULSION TECHNOLOGIES

ASAO

THRUSTER CONCEPT / PERFORMANCE *[DEMONSTRATED]*	I_{sp} (ks)	α_{ep}[a] (kg/kW$_e$)	η_{ej} (%)	F (N)	P_e (kW$_e$)	τ[b] (hrs.)
ELECTROTHERMAL: RESISTOJET-Ref.22,23	*0.1-0.4*	5	*50-80*	*0.05-1*	*0.01-30*	*4000*
ARCJET-Ref.9,10	*0.4-1.0*-1.2	5	*25-55* 50-60	*1-10*	*1-100*	*750* 1500
MICROWAVE-Ref.27	*0.1-0.6*	NA	*5-45*	*0.02-0.1*	*0.2-2*	NA
ELECTROSTATIC: ION(X_e)-Ref.29,30	*3.0-5.0*	*10*	*60-70*	*0.2* 2	*3-7*	*>1000* 10,000
ION(A)-Ref.31	7.0-10.0	5	78-80	1-10	50-500	10,000
ELECTROMAGNETIC: MPD-Ref.32,33,34	*1.0-10.0* 1.0-20.0	1	*30* 50	*10-200*	*100-1000*	*500* 2000
PIT-Ref.35	*3.0-5.0*	1	50	10-100	1000-5000	NA
PET-Ref.28	*1.0-2.0*	5	50	*0.05-1.0*	0.5-50	NA

[a] SPECIFIC MASS OF THRUSTER
[b] OPERATIONAL LIFETIME OF THRUSTER

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POWER SYSTEM TECHNOLOGIES PROS AND CONS

PHOTOVOLTAIC: ADVANTAGES/DISADVANTAGES

- ADVANTAGES
  - PROVEN, LOW RISK TECHNOLOGY
  - LOW ARRAY SPECIFIC MASS
  - SINGLE CELL FAILURE WILL NOT LEAD TO SYSTEM FAILURE
  - CAPABILITY FOR DEPLOYMENT & RETRACTION
  - CONCENTRATED PV CAN DECREASE SOLAR COLLECTION AREA

- DISADVANTAGES
  - LARGE SOLAR COLLECTION AREA
  - POWER LEVELS RESTRAINED BY AREA
  - LOW CONVERSION EFFICIENCY
  - SOLAR RADIATION DEGRADATION
  - HEAVY BATTERIES REQUIRED FOR SHADE PERIODS

POWER SYSTEM TECHNOLOGIES PROS AND CONS

SOLAR DYNAMIC: ADVANTAGES/DISADVANTAGES

- ADVANTAGES
  - HIGH SYSTEM EFFICIENCY
  - HIGH GROWTH POTENTIAL
  - EVOLUTIONARY STEP TOWARDS NUCLEAR DYNAMIC
  - TERRESTRIAL EXPERIENCE

- DISADVANTAGES
  - UNPROVEN IN SPACE
  - HIGH SYSTEM COMPLEXITY
  - ROTATING MACHINERY
  - COLLECTOR POINTING ACCURACY
  - SYSTEM PERFORMANCE DICTATED BY COMPONENTS

POWER SYSTEM TECHNOLOGIES PROS AND CONS

RADIOISOTOPE THERMOELECTRIC GENERATORS: ADVANTAGES/ DISADVANTAGES

- ADVANTAGES
  - STATIC CONVERSION (NO MOVING PARTS)
  - LONG LIFE CAPABILITY
  - DEMONSTRATED, RELIABLE TECHNOLOGY (APOLLO & VOYAGER PROGRAMS)
  - NO CRITICAL MASS, NO ACTIVE CONTROL REQUIREMENTS

- DISADVANTAGES
  - HEAT SOURCE FUEL IS EXPENSIVE AND SCARCE
  - POWER LEVELS RESTRAINED BY FUEL AVAILIBILITY
  - ENVIRONMENTALLY HAZARDOUS FUEL (ISSUE OF SAFETY AT LAUNCH)
  - LOW CONVERSION EFFICIENCY
  - SHIELDING REQUIRED FOR MANNED APPLICATIONS AND RADIATION SENSITIVE EQUIPMENT

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POWER SYSTEM TECHNOLOGIES PROS AND CONS

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DYNAMIC ISOTOPE POWER SYSTEMS: ADVANTAGES/DISADVANTAGES

- ADVANTAGES
  - HIGH CONVERSION EFFICIENCY
  - HIGHER POWER LEVELS DUE TO INCREASED EFFICIENCY
  - NO CRITICAL MASS, NO ACTIVE CONTROL REQUIREMENTS
- DISADVANTAGES
  - FUEL REQUIRED IS COSTLY, SCARCE, AND VERY HAZARDOUS
  - DYNAMIC CONVERSION (MOVING PARTS, COMPLEX DESIGN)
  - SYSTEM PERFORMANCE DICTATED BY COMPONENTS
  - SHIELDING REQUIRED FOR MANNED APPLICATIONS AND RADIATION SENSITIVE EQUIPMENT

POWER SYSTEM TECHNOLOGIES PROS AND CONS

NUCLEAR STATIC: ADVANTAGES/DISADVANTAGES

- ADVANTAGES
  - NO MOVING PARTS
  - CONVERTER CONSISTS OF MANY CELLS ARRANGED IN PARALLEL AND SERIES TO INSURE REDUNDANCY AND RELIABILITY
  - SEMICONDUCTING THERMOELECTRIC MATERIALS ARE TECHNOLOGICALLY READY

- DISADVANTAGES
  - LOW CONVERSION EFFICIENCY
  - REQUIRES START-UP POWER
  - REACTOR SWELLING LEADS TO DIFFICULTY IN MAINTAINING CELL GAP CLEARANCE
  - SHIELDING REQUIRED FOR MANNED APPLICATIONS AND RADIATION SENSITIVE EQUIPMENT

POWER SYSTEM TECHNOLOGIES PROS AND CONS

NUCLEAR DYNAMIC: ADVANTAGES/DISADVANTAGES

- ADVANTAGES
  - HIGH CONVERSION EFFICIENCY
  - HIGH POWER DENSITY (kW_e/m^3)
  - ABILITY TO PROVIDE FOR PEAK POWER REQUIREMENTS
- DISADVANTAGES
  - DYNAMIC CONVERSION (MOVING PARTS, COMPLEX DESIGN)
  - LARGE RADIATOR MASS
  - SYSTEM PERFORMANCE DICTATED BY COMPONENTS
  - REQUIRES START-UP POWER
  - SHIELDING REQUIRED FOR MANNED APPLICATIONS AND RADIATION SENSITIVE EQUIPMENT

ELECTRIC PROPULSION TECHNOLOGY PROS AND CONS

ELECTROTHERMAL THRUSTER
RESISTOJETS: ADVANTAGES/DISADVANTAGES

- ADVANTAGES
  - ESTABLISHED TECHNOLOGY
  - FLIGHT READY
  - PRECISE THRUST LEVELS
  - "OMNIVOROUS" - CAN USE NEARLY ANY PROPELLANT
  - COMPACT/SIMPLE
  - DC POWER
- DISADVANTAGES
  - ISP LIMITED BY MATERIAL TEMPERATURES
  - LONGER LIFETIMES REQUIRED

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ELECTRIC PROPULSION TECHNOLOGY PROS AND CONS

ASAO

ELECTROTHERMAL THRUSTERS (CONT'D)
ARCJETS: ADVANTAGES/DISADVANTAGES

- ADVANTAGES
  - COMPACT/SIMPLE
  - DC POWER
  - OMNIVOROUS
  - NEUTRAL PLASMA

- DISADVANTAGES
  - PRESENTLY UNDER DEVELOPMENT
  - LIFETIME IN QUESTION
  - ISP LIMITED BY MATERIAL TEMPERATURES

ELECTRIC PROPULSION TECHNOLOGY PROS AND CONS

ELECTROTHERMAL THRUSTERS (CONT'D)
MICROWAVE: ADVANTAGES/DISADVANTAGES

- ADVANTAGES
  - ELECTRODELESS - LONGER LIFE
  - EFFICIENT MICROWAVE GENERATION
  - POTENTIALLY SCALABLE TO HIGH POWER LEVELS
  - NEUTRAL PLASMA

- DISADVANTAGES
  - LABORATORY DEVICE
  - SYSTEM COMPLEXITY
  - THRUSTER WEIGHT AN ISSUE

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ELECTRIC PROPULSION TECHNOLOGY PROS AND CONS

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ELECTROSTATIC THRUSTERS
ION: ADVANTAGES/DISADVANTAGES

- ADVANTAGES
  - HIGH EFFICIENCY
  - HIGH ISP
  - ESTABLISHED TECHNOLOGY
  - FLIGHT READY
  - "OMNIVOROUS": Hg, Xe, Kr, A
  - POSSIBLE LONG LIFE
- DISADVANTAGES
  - COMPLEX POWER CONDITIONING
  - MUST INCREASE AREA TO INCREASE THRUST
  - LONG LIFE DURING HIGH POWER OPERATION REMAINS UNDEMONSTRATED

ELECTRIC PROPULSION TECHNOLOGY PROS AND CONS

ELECTROMAGNETIC THRUSTER
MPD (STEADY STATE): ADVANTAGES/DISADVANTAGES

- ADVANTAGES
  - COMPACT/SIMPLE
  - NEUTRAL PLASMA
  - POTENTIAL FOR HIGH ISP AND HIGH EFFICIENCY

- DISADVANTAGES
  - LABORATORY DEVICE
  - QUESTIONABLE LIFETIME OF ELECTRODES

WHO'S DOING WHAT?

A VARIETY OF ORGANIZATIONS (GOVERNMENT AGENCIES/INDUSTRY/UNIVERSITIES) ARE ADVOCATING/STUDYING DIFFERENT PROPULSION CONCEPTS.

EXAMPLE: A CONSORTIUM HAS BEEN FORMED TO PROMOTE AND DEVELOP A NERVA-DERIVATIVE ADVANCED NUCLEAR THERMAL ROCKET ENGINE. (WORK SUPPORTED BY AIR FORCE ASTRONAUTICS LABORATORY.)

Idaho National Engineering Laboratory
Program Integration/Test Facility

MARTIN MARIETTA
Stage Integration/Mission Analysis

Rockwell International
Rocketdyne Division
Engine System

SAIC - Science Applications International Corporation
Flight Safety/Mission Analysis

Westinghouse Electric Corporation
Advanced Power Systems Divisions
Nuclear Subsystem

WHO'S DOING WHAT? (CONTINUED)

NUCLEAR FISSION THERMAL ROCKET (OTHERS INCLUDE)

- LOS ALAMOS NATIONAL LABORATORY (LANL)
- ARGONNE NATIONAL LABORATORY (ANL)
- BROOKHAVEN NATIONAL LABORATORY (BNL)
- GRUMMAN AEROSPACE
- GENERAL ELECTRIC COMPANY
- BABCOCK AND WILCOX COMPANY
- AEROJET TECH SYSTEMS COMPANY
- MCDONNEL DOUGLAS

HYBRID NUCLEAR THERMAL ROCKET (THRUST PLUS ELECTRICITY)

- ROCKWELL INTERNATIONAL (ROCKETDYNE DIVISION)
- WESTINGHOUSE (ADVANCED POWER SYSTEMS DIVISION)
- MARTIN MARIETTA
- INEL
- AEROJET TECH SYSTEMS COMPANY (DURING NERVA PROGRAM)

GAS CORE FISSION THERMAL ROCKET

- INNOVATIVE NUCLEAR SPACE POWER INSTITUTE (INSPI) AT CALIFORNIA STATE UNIVERSITY - LONG BEACH
- SDIO
- INEL
- UNIVERSITY OF FLORIDA
- UNITED TECHNOLOGY RESEARCH LABORATORIES

WHO'S DOING WHAT? (CONTINUED)

<u>PULSED FISSION ROCKET</u> (ORION)

- GENERAL ATOMIC (DID SIGNIFICANT WORK DURING ORION PROGRAM FUNDED BY BOTH NASA & ASAF).

<u>FUSION PROPULSION</u> (BOTH MAGNETIC AND INERTIAL)

- AIR FORCE ASTRONAUTICS LAB (AFAL)
- JET PROPULSION LABORATORY (JPL)
- LAWRENCE LIVERMORE NATIONAL LAB (LLNL)
- NASA/LeRC
- ROCKETDYNE
- UNIVERSITY OF WISCONSIN
- GENERAL ATOMIC
- MCDONNELL DOUGLAS

WHO'S DOING WHAT? (CONTINUED)

ARCJETS
- NASA LeRC
- AEROJET TECH SYSTEMS/TECHNION
- ROCKET RESEARCH CENTER (RRC)
- AIR FORCE ASTRONAUTICS LAB (AFAL)
- JPL
- PRINCETON
- JAPAN

ION
- NASA LeRC
- JPL
- HUGHES
- BOEING
- ELECTRIC PROPULSION LABORATORY
- JAPAN
- GERMANY - GIESSEN INSTITUTE OF PHYSICS
- ENGLAND
- COLORADO STATE

WHO'S DOING WHAT? (CONTINUED)

MPD

- NASA LeRC
- JPL
- ESA
- GERMANY (UNIVERSITY OF STUTTGART)
- JAPAN
- PRINCETON
- MIT

SOLAR THERMAL

- AFAL
- ROCKETDYNE

SOLAR SAIL

- WORLD SPACE FOUNDATION
- JPL

WHO'S DOING WHAT? (CONTINUED)

PIT
- TRW

PET
- NASA LeRC
- GT DEVICES, INCORPORATED

MICROWAVE
- NASA LeRC
- MICHIGAN STATE
- PENN STATE

LASER THERMAL
- LAWRENCE LIVERMORE LAB (LLL)
- PHYSICAL APPLICATION, INCORPORATED
- KMS FUSION
- LOCKHEED MISSILES & SPACE
- UNIVERSITY OF TENNESSEE SPACE INSTITUTE
- UNIVERSITY OF ILLINOIS

WHO'S DOING WHAT? (CONTINUED)

RTG

- US DOD
- NASA - LeRC
- GENERAL ELECTRIC COMPANY
- TELEDYNE ENERGY SYSTEMS
- JPL

DIPS

- US DOD
- NASA - LeRC
- ROCKWELL, ROCKETDYNE DIVISION
- GARRETT FLUID SYSTEMS
- SUNDSTRAND CORPORATION
- GENERAL ELECTRIC COMPANY
- GRUMMAN SPACE SYSTEMS

WHO'S DOING WHAT? (CONTINUED)

SOLAR DYNAMICS

- NASA - LeRC
- ROCKWELL, ROCKETDYNE DIVISION
- GARRET FLUID SYSTEMS
- SUNDSTRAND CORPORATION
- GRUMMAN SPACE SYSTEMS
- MECHANICAL TECHNOLOGY, INCORPORATED

PV

- NASA - LeRC
- ROCKWELL, ROCKETDYNE DIVISION
- FORD AEROSPACE
- LOCKHEED MISSILES AND SPACE
- NASA - MSFC

WHO'S DOING WHAT? (CONTINUED)

SP-100

- NASA - LeRC
- US DOD
- GENERAL ELECTRIC COMPANY
- US DOD
- THERMO ELECTRON CORPORATION
- GA TECHNOLOGIES
- MECHANICAL TECHNOLOGY, INCORPORATED

SNAP-DYN

- ROCKWELL, ROCKETDYNE DIVISION
- GARRET FLUID SYSTEMS COMPANY
- SUNDSTRAND CORPORATION

WHO'S DOING WHAT? (CONTINUED)

MMWe

- GENERAL ATOMIC TECHNOLOGIES
- WESTINGHOUSE CORPORATION
- GARRET
- ORNL
- ROCKWELL
- FORD AEROSPACE
- THERMACORE
- JPL
- NASA - LeRC
- THERMO-ELECTRON CORPORATION

DATA USED IN SYSTEMS ASSESSMENT

ORIGINAL PAGE IS
OF POOR QUALITY

INPUT STATUS
0

SPECIFIC MASS OF PROPULSION SYSTEMS

SOURCE	CONV	CONFIG	Resistojet	Arcjet	Ion(Xe)	Sol Therm.	Solar Sail	SCR
[illegible]	T/E	[illegible]	733.00	1000.90	839.43			
		[illegible]	700.33	317.30	103.71			
		[illegible]						
	[illegible]	CPS	790.03	909.94	103.74			
	[illegible]	[illegible]						
	[illegible]	[illegible]	910.07	373.90	107.00			
SOLAR	[illegible]	DIRECT						
	PV	BI-PLN	443.07	807.01	300.04	4.00	3.4E-03	
		BI-PLN	103.31	247.87	140.79			
		BI-PLN	13.00	77.90	70.00			
		GaAs PLN	13.03	20.79	10.00			
		GaAs CONC						
		BI-ALPHA						
		HP						
		[illegible]						
		[illegible]						
	[illegible]	CPS	410.40	813.00	390.01			
	[illegible]	CPS	430.33	800.00	400.00			
	[illegible]	[illegible]	300.13	940.38	319.01			
	[illegible]	[illegible]	431.17	831.79	300.14			
	[illegible]	FP	439.00	897.00	300.00			
	[illegible]	FP						
[illegible]	[illegible]	CPS	190.17	281.20	173.87			
	[illegible]	[illegible]	147.20	213.90	129.03			
	[illegible]	FP						
SP-100	T/E	[illegible]	80.11	84.10	90.70			
FISSION INDIRECT		[illegible]	73.00	117.90	71.43			
	T/I	BI CORE	87.00	80.33	90.30			
	[illegible]	FP	70.24	110.00	73.00			
	[illegible]	[illegible]	77.00	110.00	73.00			
[illegible]	[illegible]	[illegible]						
	[illegible]	[illegible]						
	T/I	BI CORE						
	T/E	AM						
	MHD	TURBOAMPD						
SCR	MHD	[illegible]						
FUSION INDIRECT	[illegible]							
FISSION DIRECT	SCR							8.3E-03
	GCR							
FUSION DIRECT	MCF							
	ICF							
Isp (s)			100.00 400.00	400.0 1700.0	3000.00 5000.00	800.0 1000.0	1000.0 30000.0	830.0 1700.0

	Ion(Ar)	MPD	PIT	PET	Microwave	Laser Ther	GCR	MCF	ICF	RAILGUN	MASS DRIVER
Isp (s)	7000.00 10000.00	1000.00 70000.00	3000.00 5000.00	1000.0 7000.0	100.0 800.0	1000.0 2000.0	3000.0 7000.0	10000.0 100000.0	20000.0 1000000.0	1000.0 2000.0	1000.0 2000.0

SPECIFIC MASS OF PROPULSION SYSTEMS

INPUT STATUS
P

	CONV	CONFIG	Resistojet	Arcjet	Ion(Xe)	Sol Therm.	Solar Sail	SCR	Ion(Ar)	MPD	PIT	PET	Microwave	Laser Ther*	GCR	MCF	ICF	NUCLEAR	MASS DRIVER
ISOTOPE	T/E	PbTe							848.90	871.00	871.00	870.00	1402.07					1018.20	800.00
		SiGe(GaP)							135.00	242.00	242.00	250.00	433.33					800.00	200.00
		AMTEC	108.33	182.90	108.00				80.00	122.00	122.00	130.00	233.33					320.00	100.00
	RANKINE	ORC							153.27	238.73	238.73	247.23	420.72					384.47	277.23
	STIRLING	ORC/TE	102.34	273.91	103.43				155.50	218.81	218.81	218.81	301.35					407.81	240.81
	BRAYTON	CBC							101.75	232.00	232.00	200.00	430.00					800.00	200.00
SOLAR	DIRECT	DIRECT												10.00					
	PV	Si PLN							332.90	820.00	820.00	834.00	800.02					1120.10	804.00
		Si-PLN							122.00	180.70	180.70	180.70	347.00					480.81	220.20
		Si-PLN							10.00	10.00	10.00	10.00	40.67					80.00	40.00
		GaAs PLN							0.13	0.00	0.00	10.00	44.33					83.20	40.00
		GaAs-CONC	10.07	25.00	21.43				11.25	12.00	12.00	20.00	30.00					100.00	30.00
		Si-ALPHA	13.83	20.73	10.00				0.13	0.00	0.00	10.00	44.33					83.20	40.00
		InP	13.83	20.73	10.00				0.13	0.00	0.00	10.00	44.33					83.20	40.00
		CASCADE	13.00	22.90	20.00				10.00	10.00	10.00	10.00	40.67					90.00	40.00
		THERMAL	81.07	137.90	83.71				87.50	102.00	102.00	110.00	200.00					200.00	140.00
	RANKINE	ORC							300.55	484.40	484.40	407.40	837.40					1044.93	822.40
	RANKINE	ORC							343.73	543.20	543.20	531.20	835.33					1182.40	801.20
	BRAYTON	CBC							200.93	424.70	424.70	432.20	737.13					824.30	402.20
	BRAYTON	CBC							314.03	407.40	407.40	308.40	830.00					1070.00	830.40
	STIRLING	FP							325.00	514.00	514.00	522.00	800.67					1104.00	502.00
	STIRLING	FP	20.00	62.90	37.14				25.00	34.00	34.00	42.00	80.67					144.00	72.00
SNAP DYN	RANKINE	ORB							144.30	225.00	275.00	233.00	403.00					520.00	203.00
	BRAYTON	CBD							105.40	187.04	187.04	170.04	301.07					401.70	200.04
	STIRLING	FP	134.37	201.00	122.31				80.53	153.24	153.24	101.24	203.40					302.40	181.24
SP-100	T/E	SiGe(GaP)							40.83	50.33	50.33	87.33	170.00					104.00	87.33
FISSION INDIRECT		SiGe(GaP)							85.00	87.00	82.00	90.00	180.67					240.00	120.00
	T/I	IN CORE							42.01	81.22	81.22	80.22	132.03					150.44	80.22
	STIRLING	FP							50.10	87.00	87.00	85.00	175.14					290.17	125.00
	BRAYTON	CBD							87.24	83.50	83.50	83.90	172.09					247.10	122.00
MMW SCR	RANKINE	RRC	10.83	23.70	30.71				10.03	11.00	11.00	10.00	40.33					80.00	40.00
	BRAYTON	CBD	10.33	27.00	22.00				12.50	14.00	14.00	22.00	53.33					104.00	52.00
	T/I	IN CORE	10.33	27.00	22.00				12.50	14.00	14.00	22.00	53.33					104.00	52.00
	T/E	AM	14.83	22.23	10.00				9.00	0.00	0.00	17.00	60.33					95.00	47.00
	MHD	TURBOALT	10.07	25.00	21.43				11.25	12.00	12.00	20.00	80.00					100.00	50.00
GCR	MHD	BRAYTON	8.03	13.75	14.71				5.30	2.00	2.00	10.00	34.33					81.20	40.00
FUSION INDIRECT	ELECTROSTATIC		10.00	15.00	15.71				0.75	4.00	4.00	17.00	30.07					84.00	42.00
FISSION DIRECT	ECR														2.0E-02				
	CCR															2.0E-01			
FUSION DIRECT	MCF																8.0E-03		
	ICF																		
Isp (s)			100.00	400.0	2000.00	800.0	1000.0	850.0	7000.00	1000.00	2000.00	1000.0	150.0	1000.0	2000.0	10000.0	20000.0	1000.0	1000.0
			400.00	1200.0	3000.00	1000.0	20000.0	1200.0	10000.00	20000.00	5000.00	2000.0	800.0	2000.0	7000.0	100000.0	1000000.0	2000.0	2000.0

ORIGINAL PAGE IS
OF POOR QUALITY

ADVANCED PROPULSION DATA MATRIX

	Resistojet	Arcjet	Ion(Xe)	Sol Therm'	Solar Sail	SCR	Ion(Ar)	MPD	PIT
Isp (s)	100.0	400.0	3000.00	800.0	1000.0	850.0	7000.00	1000.00	3000.00
Isp (s)	400.0	1200.0	6000.00	1000.0	30000.0	1200.0	10000.00	20000.00	5000.00
a(kg/kWe)	5.00	8.00	10.00	2.0	3.42E-03	8.30E-03	4.00	1.00	1.00
Thrust eff (%)	60.00	40.00	70.00	60.0	100.0	72.0	80.00	50.00	50.00
POWER (MWe)	1.00	1.00	1.00	1000.0	8.00E+00	1.10E+08	10.00	10.00	1.0
POWER (MWe)	10.00	100.00	10.00	1000.0	8.00E+00	5.00E+08	500.00	10000.0	5000.0
Thrust (N)	0.5	0.1	0.10	100.0	37.3	2.45E+05	1.00	1.0	10.000
Thrust (N)	1.0	10.0	0.20	200.0	37.3	1.12E+06	10.00	200.0	200.0
Life (hrs)	10000.0	1500.0	10000.00		3.68E+04	10.0	10000.00	2000.0	
Status	D	D	D	D	D	D	P	P	P

	PET	Microwave	Laser Ther'	GCR	MCF	ICF	RAILGUN	MASS DRIVER
Isp (s)	1000.0	150.0	1000.0	3000.0	10000.0	20000.0	1000.0	1000.0
Isp (s)	2000.0	800.0	2000.0	7000.0	100000.0	1000000.0	2000.0	2000.0
a(kg/kWe)	5.0	10.0	5.0	0.020	0.2	0.009	20.0	20.0
Thrust eff (%)	50.0	30.0	50.0	72.0	80.0	20.5	25.0	50.0
POWER (MWe)	0.5	0.1	1000.0	7.50E+05	7.50E+08	5.00E+07	100000.0	1000000.0
POWER (MWe)	100.0	2.0	10000.0	1.50E+06	7.50E+08	5.00E+07	1000000.0	10000000.0
Thrust (N)	0.1	0.010	100.0	22000.0	80000.0	40000.0	10.0	10.0
Thrust (N)	1.0	1.0	500.0	440000.0			100.0	1000.0
Life (hrs)								
Status	P	P	P	P	P	P	P	P

ORIGINAL PAGE IS
OF POOR QUALITY

POWER DATA MATRIX

SOURCE	CONV.	POWER CONFIG	DATE	SP MASS kG/kWe	THERMAL EFF	RANGE kWe	RANGE kWe	POWER kWe	TEMP K	RAD AR m2/kWe	COLL AR m2/kWe	LIFE YEARS	REMARKS
ISOTOPE	T/E	PbTe	D	434.8	5.0	0.01	1.0	0.07	900.0	28.57		7.0	SNAP 27, APOLLO 12-17
		SiGe(GaP)	D	120.0	7.6	0.01	1.0	0.34	1273.0	4.67		5.0	GE, MOD-RTG (GPHS)
		AMTEC	P	60.0	22.0	1.0	10.0					7.0	LeRC PTD, MARS SURFACE
	RANKINE ORC	ORC	D	118.6	23.3	1.0	10.0	6.0	673.0	8.88		12.0	DIPS, SUNDSTRAND
	HYBRID	ORC/TE	P	104.4	26.8	1.0	10.0	6.0	1100/650	7.40		12.0	DIPS, SUNDSTRAND
	BRAYTON	CBC	D	125.0	28.0	1.0	10.0	6.0	1034.0	5.57		12.0	DIPS, GARRETT
SOLAR	DIRECT		D										
	PV	Si-PLN	D	262.0	3.0	1.0	100.0	20.2	325.0	2.57	24.76	10.0	DR-02, 2 ARRAYS, 78 PANELS/ARRAY, NIH BATTERIES
		Si-PLN	D	94.1	5.8	1.0	100.0	39.4	325.0	1.32	12.67	10.0	DR-02, 2 ARRAYS, 78 PANELS/ARRAY, NO STORAGE
		Si-PLN	D	4.00	20.0	1.0	100.0	10.0	325.0			10.0	LeRC PTD, NO STORAGE, MINIMAL STRUCTURE
		GaAs-PLN	D	3.30	25.0	1.0	100.0	10.0	325.0				LeRC PTD, NO STORAGE, MINIMAL STRUCTURE
		GaAs-CONC	P	5.00	25.0	1.0	100.0	10.0	350.0				LeRC PTD, NO STORAGE, MINIMAL STRUCTURE
		Si-ALPHA	P	3.30	15.0	1.0	100.0	10.0	325.0				LeRC PTD, NO STORAGE, MINIMAL STRUCTURE
		InP	P	3.30	23.0	1.0	100.0	10.0	325.0				LeRC PTD, NO STORAGE, MINIMAL STRUCTURE
		CASCADE	P	4.00	30.0	1.0	100.0	10.0	325.0				LeRC PTD, NO STORAGE, MINIMAL STRUCTURE
		THERMAL	P	50.00	30.0	1.0	100.0	25.0	2000.0	0.8		10.0	LeRC PTD, NO STORAGE, MINIMAL STRUCTURE
	RANKINE	ORC	D	241.24	27.5	10.0	50.0	28.2	675.0	6.77	7.06	10.0	ROCKETDYNE, DR-02, NO THERMAL STORAGE
	RANKINE	ORC	D	270.60	27.5	10.0	50.0	28.2	675.0	6.8	7.7	10.0	ROCKETDYNE, DR-02, WITH THERMAL ENERGY STORAGE
	BRAYTON	CBC	D	211.14	33.3	10.0	50.0	28.2	1012.0	5.27	6.32	10.0	ROCKETDYNE, DR-02
	BRAYTON	CBC	D	247.70	33.3	10.0	50.0	28.2	1012.0	5.3	6.3	10.0	ROCKETDYNE, DR-02, WITH THERMAL ENERGY STORAGE
	STIRLING	FP	D	256.00	34.0	10.0	50.0	25.0	1080.0	4.6	6.8	10.0	LeRC PTD WITH THERMAL ENERGY STORAGE
	STIRLING	FP	P	16.0	34.0	10.0	50.0		1080.0	4.6		10.0	LeRC PTD, LUNAR SURFACE, NO THERMAL ENERGY STORAGE
SNAP-DYN	RANKINE	ORC	D	111.50	17.7	10.0	50.0	30.0	728.0	6.07		10.0	SNAP10A, SUNDSTRAND/ROCKETDYNE
	BRAYTON	CBC	D	80.32	19.2	10.0	50.0	50.0	922.0	4.68		10.0	SNAP10A, GARRET/ROCKETDYNE
	STIRLING	FP	P	75.62	19.3	10.0	50.0	50.0	922.0	4.04		10.0	SNAP10A, MTI/ROCKETDYNE
SP-100	T/E	SiGe(GaP)	D	28.66	5.5	100.0	300.0	105.6	1350.0			7.0	GE, ASTRO SPACE DIV
		SiGe(GaP)	D	40.00	7.0	100.0	300.0	105.6	1350.0			7.0	LeRC PTD
	T/I	IN-CORE	D	29.61	8.1	100.0	300.0	100.0	2200.0			7.0	MARTIN MARIETTA
	STIRLING	FP	D	42.54	26.0	100.0	300.0	100.3	1350.0	1.86		7.0	GE, ASTRO SPACE DIV
	BRAYTON	CBC	D	41.79	18.5	100.0	300.0	100.3	1150.0	2.77		7.0	GE, ASTRO SPACE DIV
MMW SCR	RANKINE	NRC	P	4.50	20.5	1000.0	5.0E+04	1.0E+04	1450.0	0.14		10.0	LMR, ORNL/ROCKWELL
	BRAYTON	CBC	P	6.00	22.0	1000.0	5.0E+04	1.0E+04	1500.0	0.43		10.0	LMR-NERVA DERIVATIVE, WESTINGHOUSE/GARRETT
	T/I	IN-CORE	P	6.00	12.0	1000.0	5.0E+04	1.0E+04	1900.0	0.30		10.0	GENERAL ATOMICS
	T/E	AM	P	3.90	22.0	1000.0	5.0E+04	1.0E+04	1350.0	0.24		10.0	LMR, WESTINGHOUSE/FORD
	MHD	TURBOAMHD	P	5.00	40.00	1.0E+04	1.0E+05	1.0E+04	2500.0	0.2			NDR(NERVA DERIVATIVE REACTOR), LeRC 1972
GCR	MHD	BRAYTON	P	0.30	40.0	1.0E+05	1.0E+06						ROSA AND MYRABO, 1984
FUSION INDIRECT	ELECTROSTATIC		P	1.00	80.0	1.0E+05	1.0E+06						SOAR, FUS.TECH. INST., U. WIS.
FISSION DIRECT	SCR		D	8.30E-03	70.0	1.00E+06	5.00E+06	1.00E+06	2500.0			10.0	NERVA PROGRAM
	GCR		P	2.00E-02	70.0	7.50E+05	1.50E+06						
FUSION DIRECT	MCF		P	0.2		7.50E+06	7.50E+06	7.50E+06					
	ICF		P	8.00E-03		5.00E+07	5.00E+07	5.00E+07					

STATUS (S) D - DEMONSTRATED
P - PROJECTED

ORIGINAL PAGE IS
OF POOR QUALITY

Extra-Terrestrial Propellant Production

NASA JSC Advanced Projects Office

Eagle Engineering/Lockheed Houston

Bill Stump 713-338-2682

Results

- O Lunar Oxygen in Lander - saves 30% of LEO mass
- O Mars Surface Propellant
  - O_2 only - saves 5%
  - O_2 and fuel - saves 7%
- O Phobos Prop. Production
  - O_2 only - saves 15-20%
  - O_2 and fuel - saves 25%
- O Phobos and Surface
  - O_2 only - saves 30%
  - O_2 and fuel - saves 44%
- O Lunar Oxygen for Mars Missions - Need hydrogen from moon or other fuel

Different Levels of Calculation

1. Mass in LEO savings for a single vehicle/mission - best case.

2. Average mass savings for a scenario (many missions) accounting for plant delivery and ops.

3. Cost comparison - are there really cost savings?

Benefits Other Than LEO Mass or Near-Term Program Cost

- O Self-sufficiency, flexibility, safety
- O Products needed to start a colony

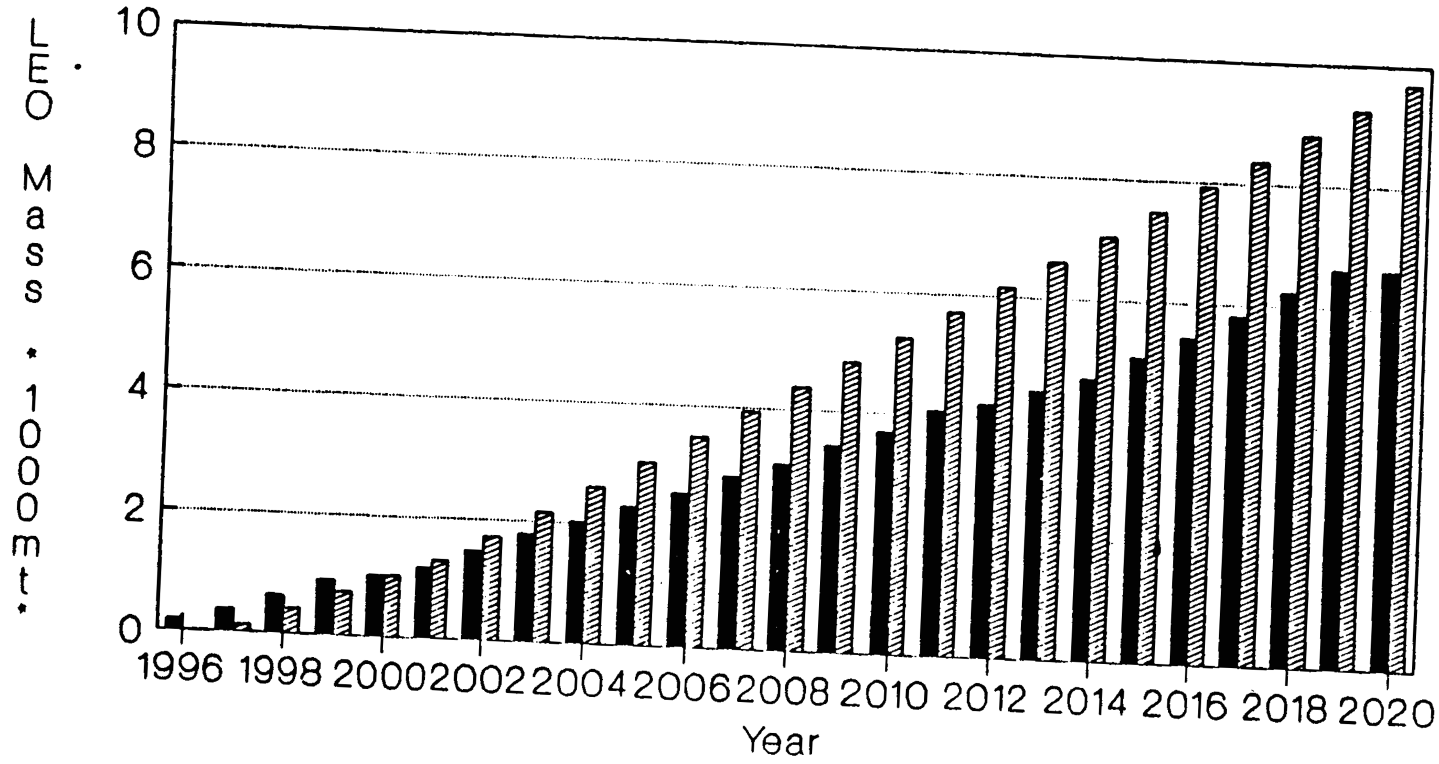
ET Propellant Production is a Long Term Project

- O Sprint scenarios do not benefit
- O Payload is lost in first years
- O Payback is over 10-20 year time scale

Lunar Oxygen for Lunar Landers

- ~ 40 % reduction in LEO propellant - best case
  - Assumes reusable OTVs and single stage lander based on surface or in LLO
  - No plant mass included, two mission calc.
- LSPI model - 30% savings over 20 years
  - Includes plant mass
  - Plot courtesy Kyle Fairchild (NASA JSC)

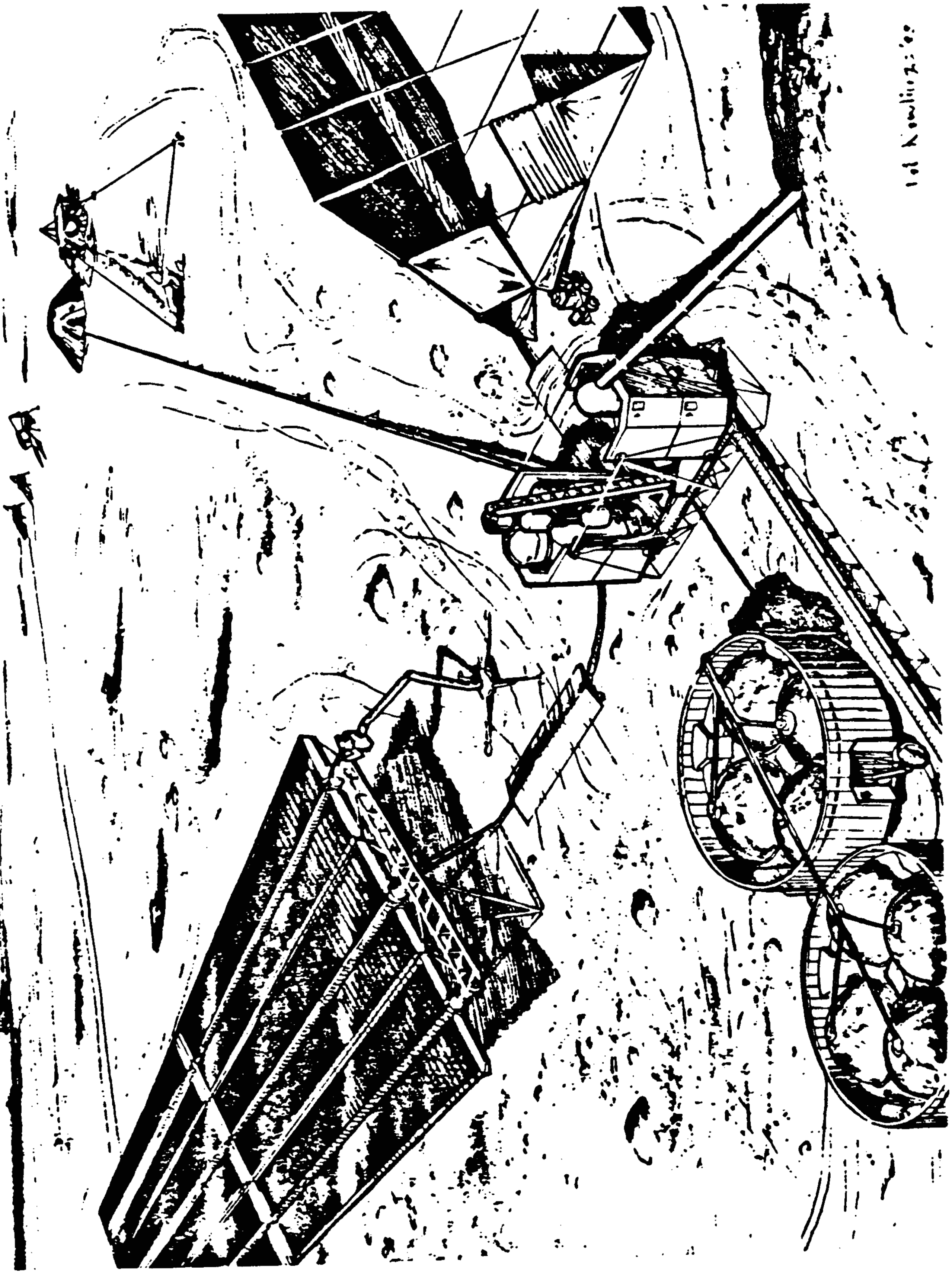
TRADE: L-SRH-1.4E
Accumulated LEO MASS
LLOX Production -vs- No LLOX

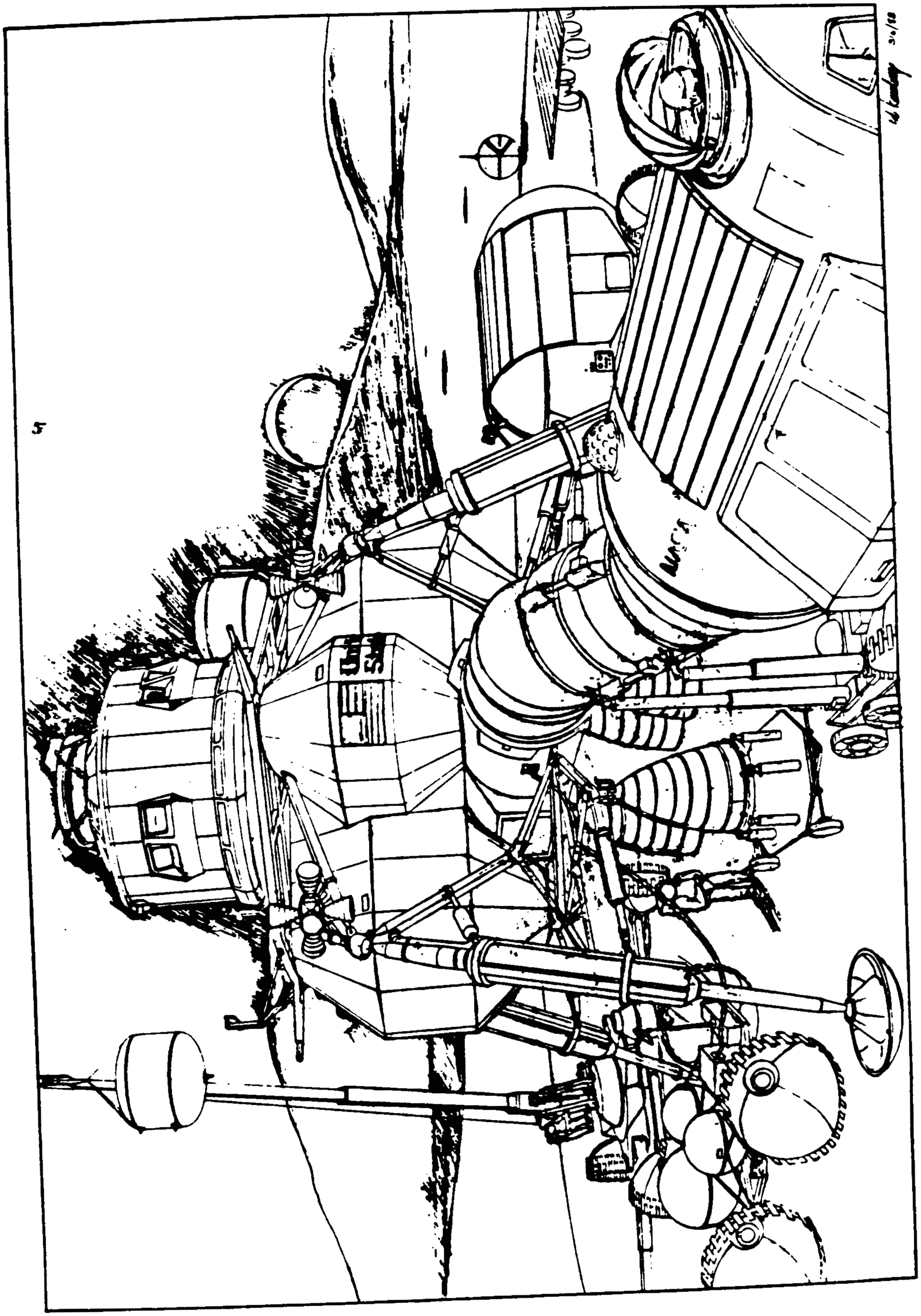


LLOX used roundtrip LLO to LS

Chief Unknowns

- Lander/OTV
  - Can we truly reuse and maintain in space?
  - Can we build the engine?
  - High mixture ratio helps
  - Cost per flight?
- Propellant Plant
  - Practicality of building and operating
  - Process, mass, power, crew req.





Mars Surface and Phobos Prop. Production

- O 10 mission, 20 year sequence
- O Conjunction class missions
- O O_2/H_2 TMI, MOI
- O O_2/Methane TEI, EOI
- O Baseline Mission - 1,300 m tons in LEO
- O Many other assumptions and guesses

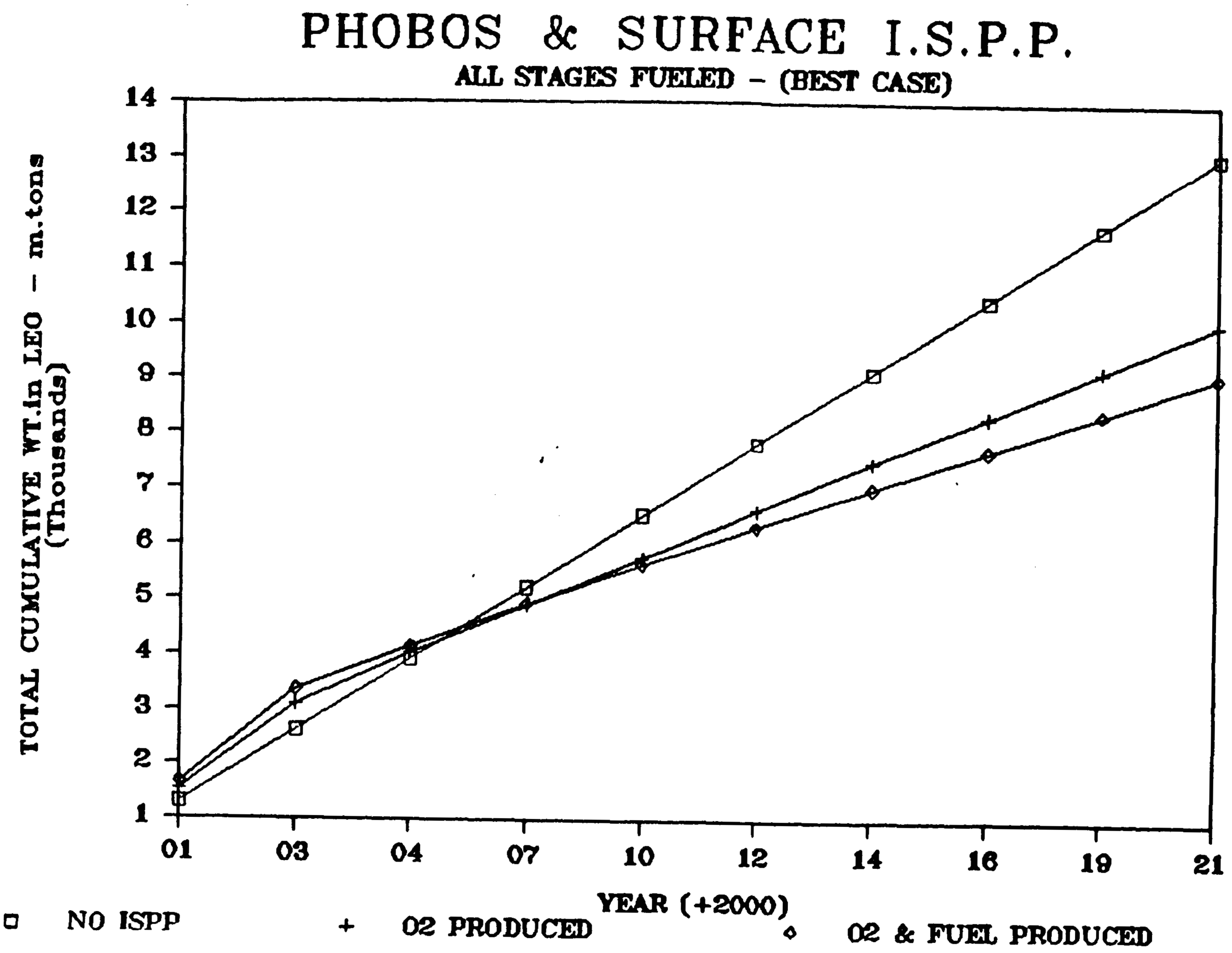

PHOBOS & SURFACE I.S.P.P.
ALL STAGES FUELED - (BEST CASE)
TOTAL CUMULATIVE WT.in LEO - m.tons
(Thousands)
14
13
12
11
10
9
8
7
6
5
4
3
2
1
01
03
04
07
10
12
14
16
19
21
YEAR (+2000)
NO ISPP
02 PRODUCED
02 & FUEL PRODUCED

Fig.2

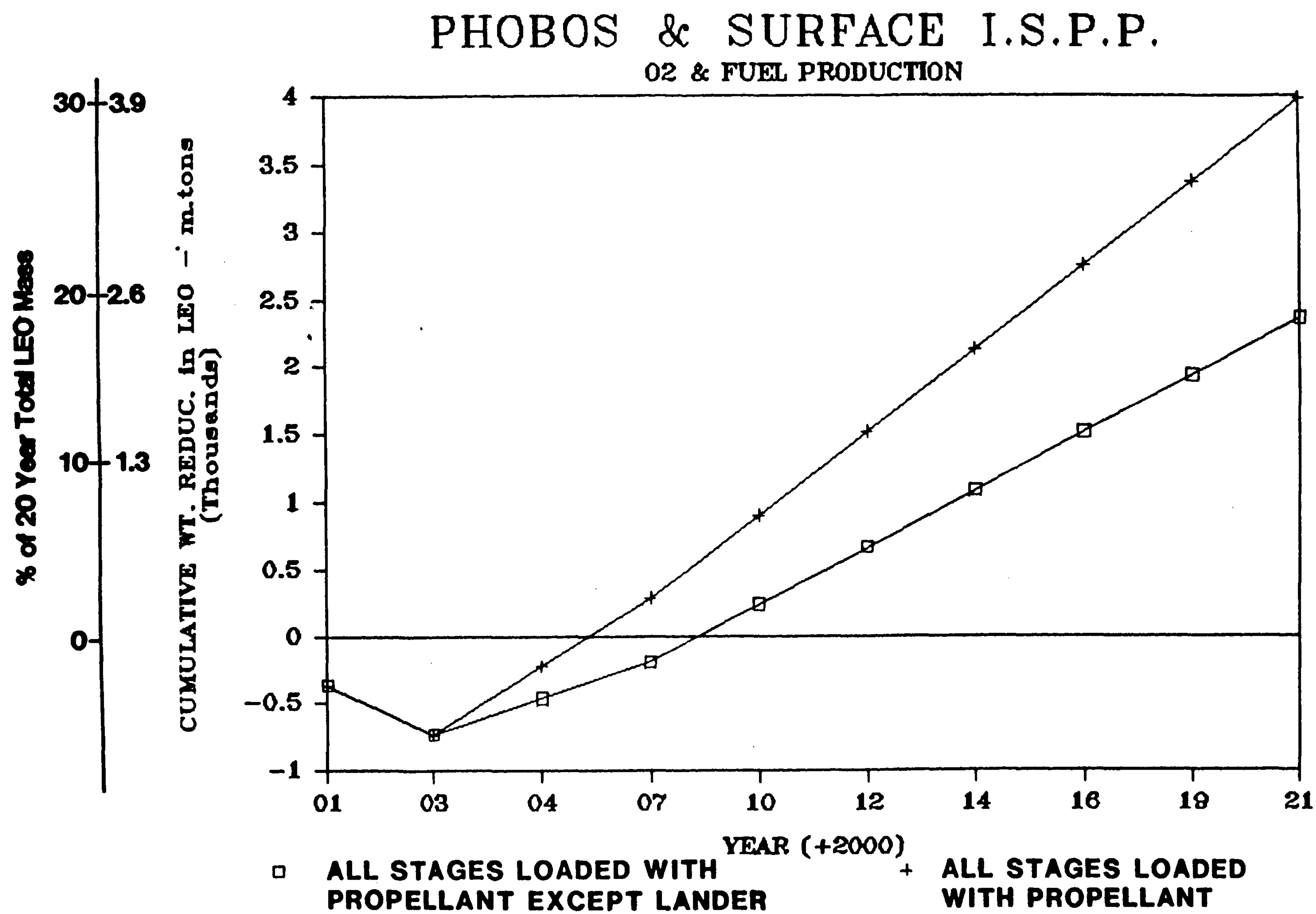

PHOBOS & SURFACE I.S.P.P.
02 & FUEL PRODUCTION
% of 20 Year Total LEO Mass
30
20
10
0
3.9
2.6
1.3
CUMULATIVE WT. REDUC. in LEO – m.tons
(Thousands)
4
3.5
3
2.5
2
1.5
1
0.5
0
−0.5
−1
01
03
04
07
10
12
14
16
19
21
YEAR (+2000)
ALL STAGES LOADED WITH PROPELLANT EXCEPT LANDER
ALL STAGES LOADED WITH PROPELLANT

Fig.3

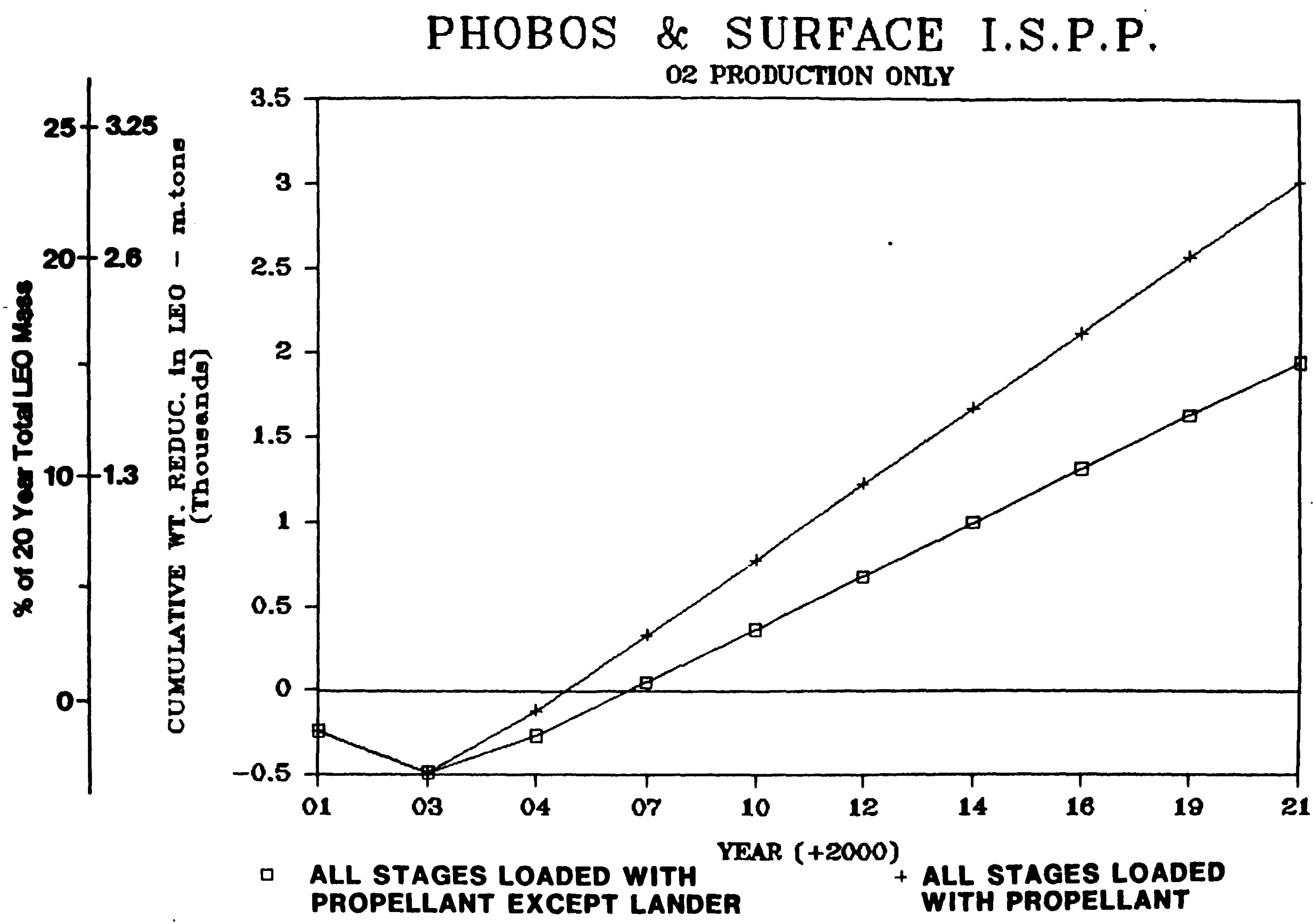

PHOBOS & SURFACE I.S.P.P.
02 PRODUCTION ONLY
% of 20 Year Total LEO Mass
25
20
10
0
3.25
2.6
1.3
CUMULATIVE WT. REDUC. in LEO - m.tons
(Thousands)
3.5
3
2.5
2
1.5
1
0.5
0
-0.5
01
03
04
07
10
12
14
16
19
21
YEAR (+2000)
□ ALL STAGES LOADED WITH PROPELLANT EXCEPT LANDER
+ ALL STAGES LOADED WITH PROPELLANT

Fig.4

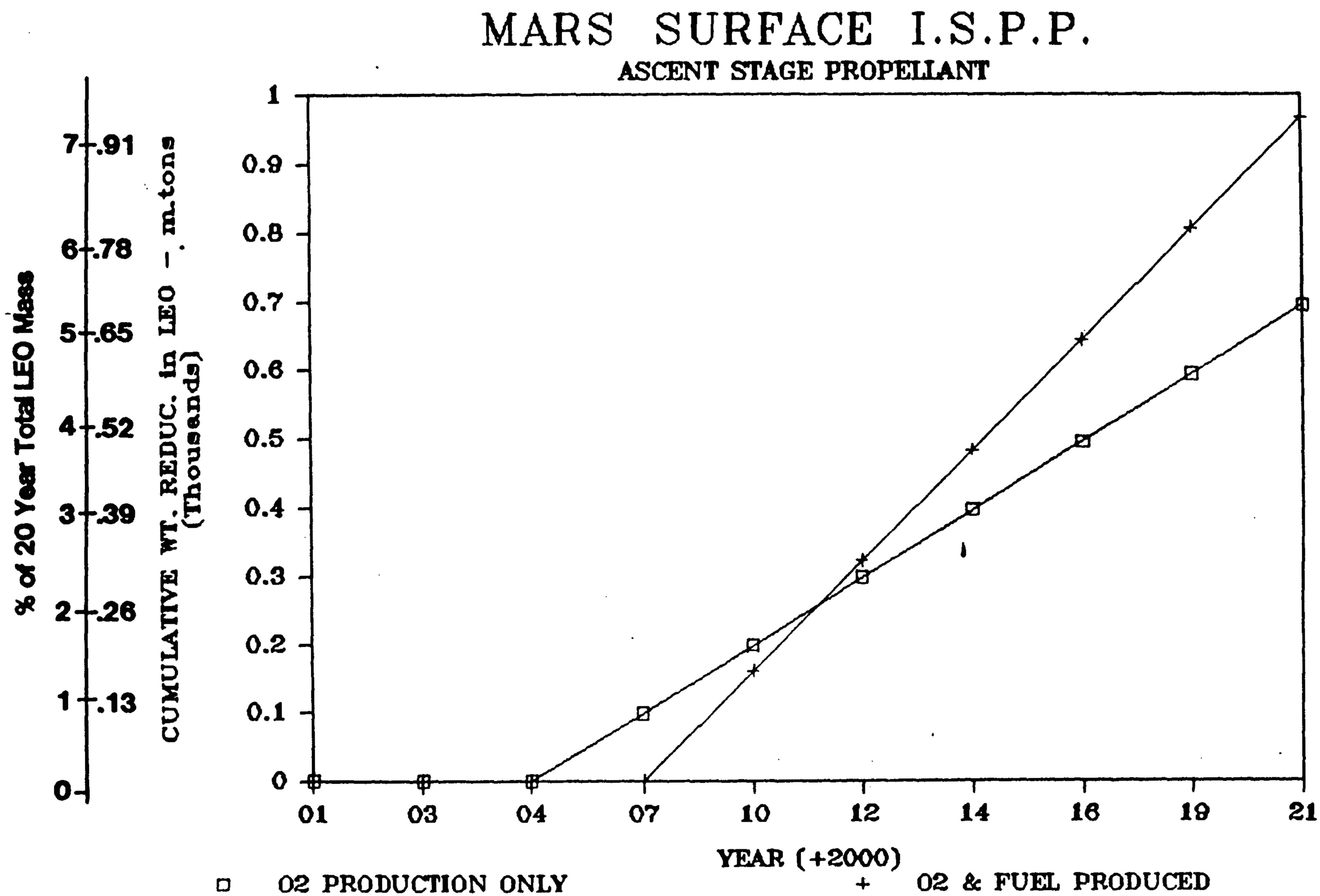

MARS SURFACE I.S.P.P.
ASCENT STAGE PROPELLANT
% of 20 Year Total LEO Mass
CUMULATIVE WT. REDUC. in LEO - m.tons
(Thousands)
0
1
2
3
4
5
6
7
.13
.26
.39
.52
.65
.78
.91
0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0.9
1
01
03
04
07
10
12
14
16
19
21
YEAR (+2000)
□ 02 PRODUCTION ONLY
+ 02 & FUEL PRODUCED

Fig.5

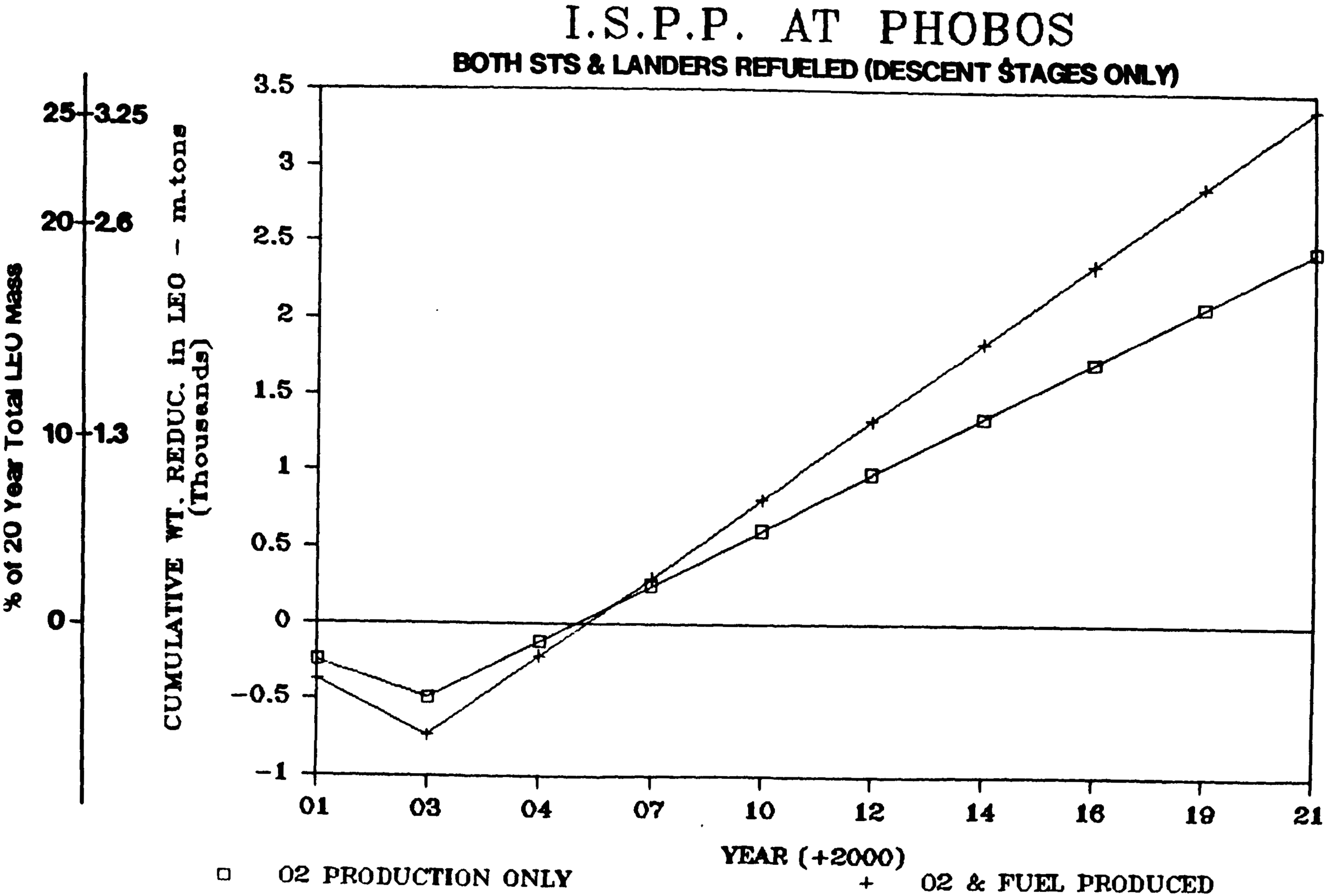
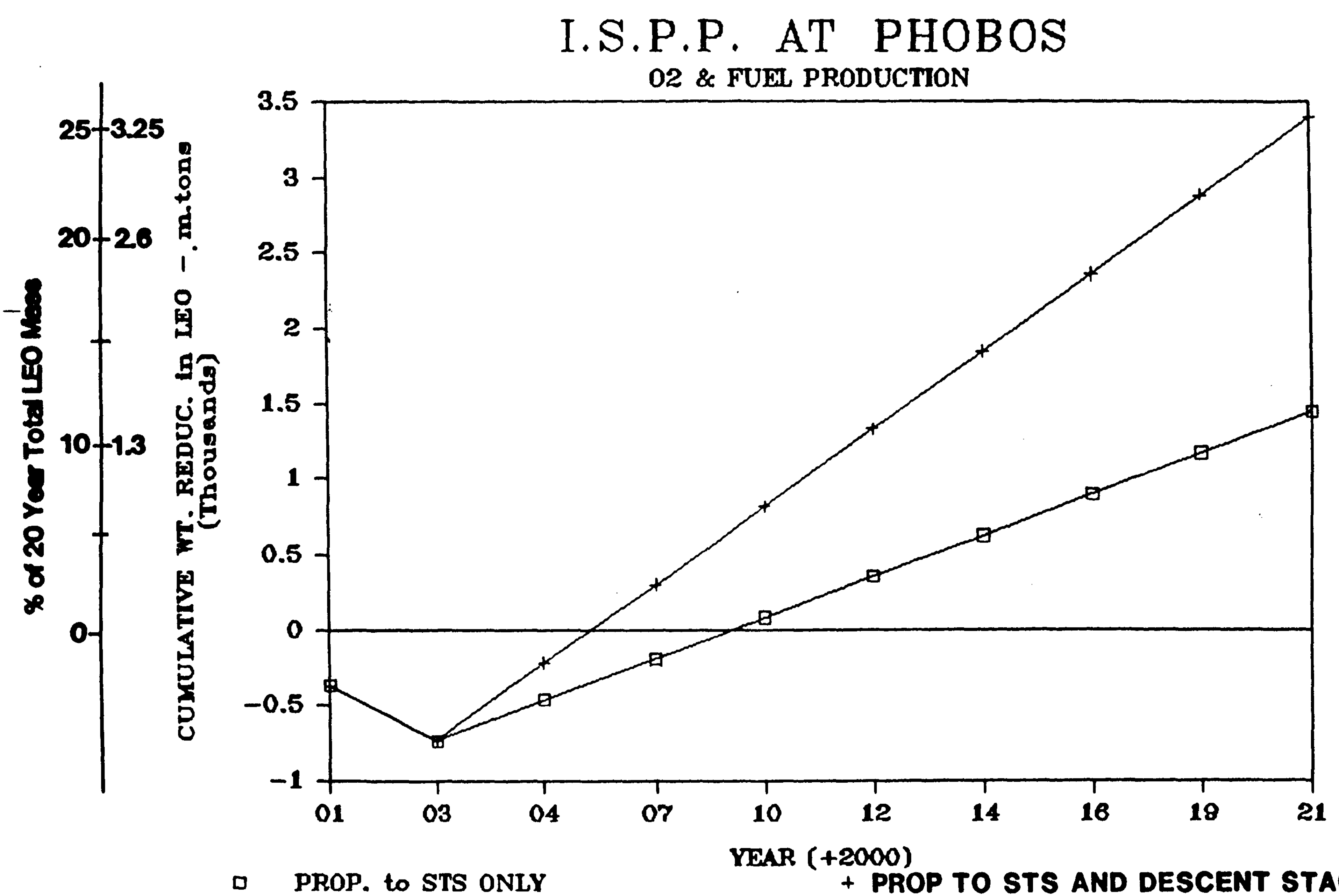
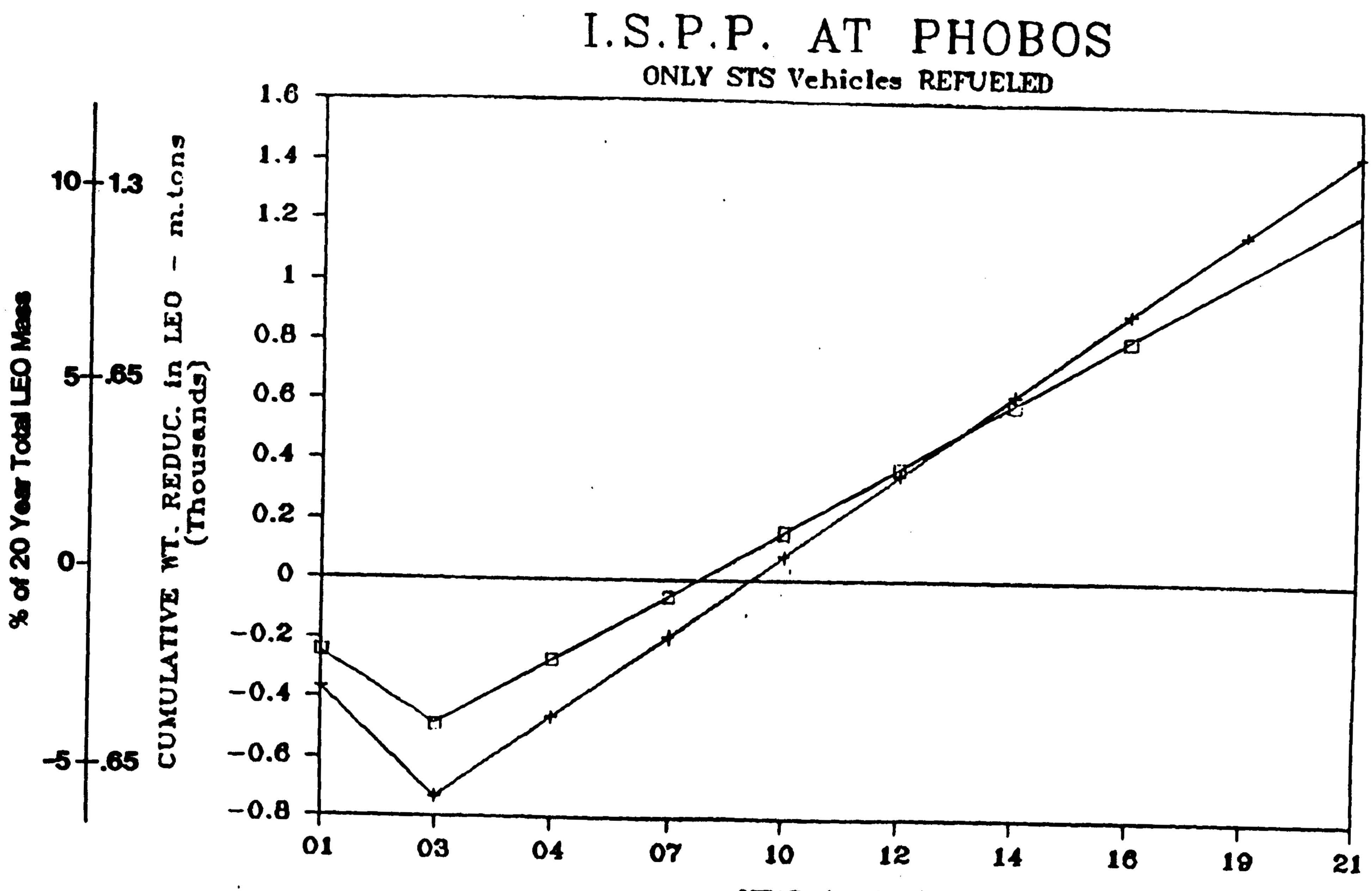

I.S.P.P. AT PHOBOS
BOTH STS & LANDERS REFUELED (DESCENT STAGES ONLY)
% of 20 Year Total LEO Mass
25
20
10
0
3.25
2.6
1.3
CUMULATIVE WT. REDUC. in LEO - m.tons
(Thousands)
3.5
3
2.5
2
1.5
1
0.5
0
-0.5
-1
01
03
04
07
10
12
14
16
19
21
YEAR (+2000)
02 PRODUCTION ONLY
02 & FUEL PRODUCED

Fig.6



I.S.P.P. AT PHOBOS
02 & FUEL PRODUCTION
% of 20 Year Total LEO Mass
25
20
10
0
3.25
2.6
1.3
CUMULATIVE WT. REDUC. in LEO - m.tons
(Thousands)
3.5
3
2.5
2
1.5
1
0.5
0
-0.5
-1
01
03
04
07
10
12
14
16
19
21
YEAR (+2000)
PROP. to STS ONLY
+ PROP TO STS AND DESCENT STA

Fig.7

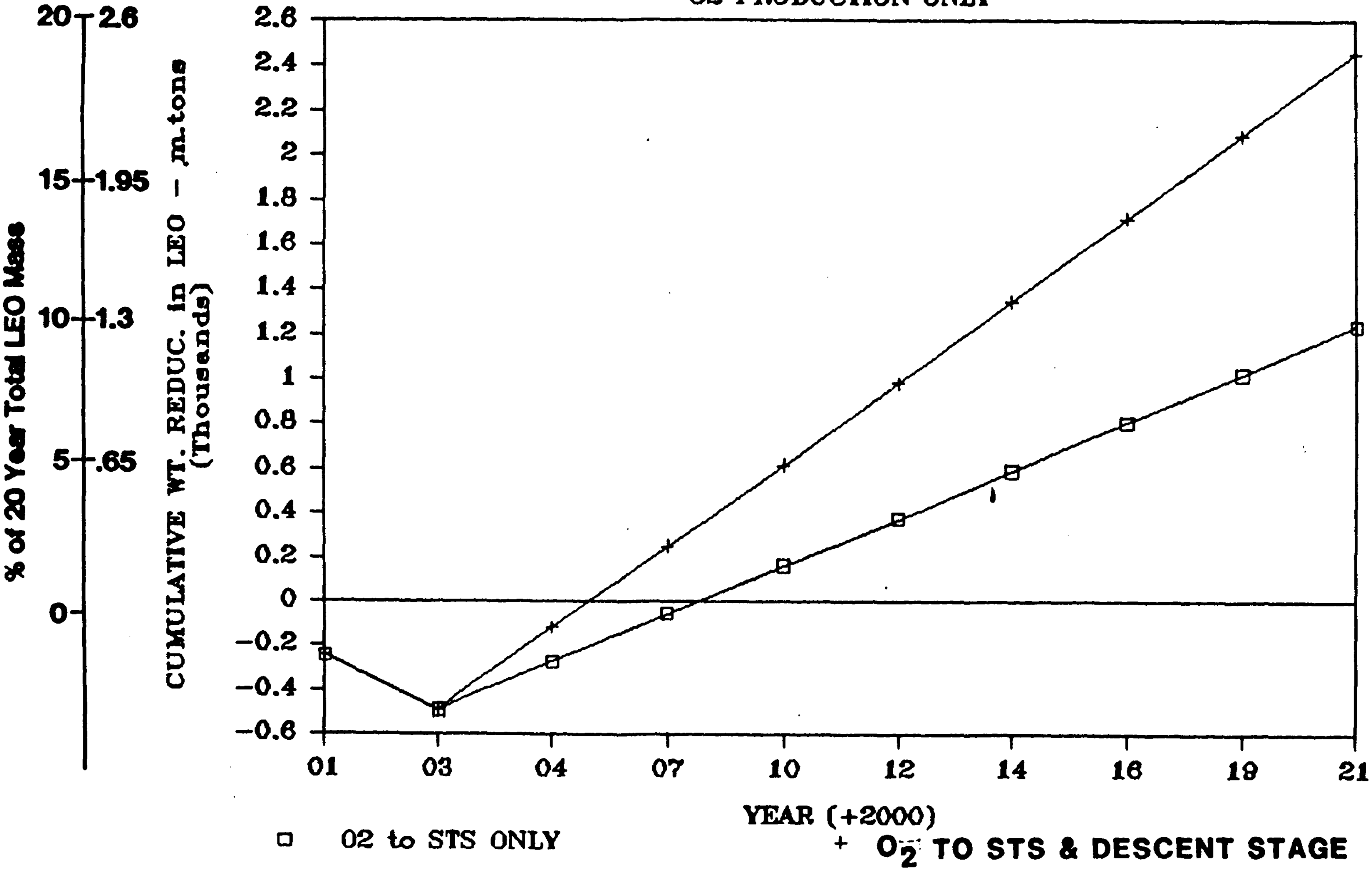


□ O2 PRODUCTION ONLY + .O2 & FUEL PRODUCED

Fig.8

I.S.P.P. AT PHOBOS

02 PRODUCTION ONLY



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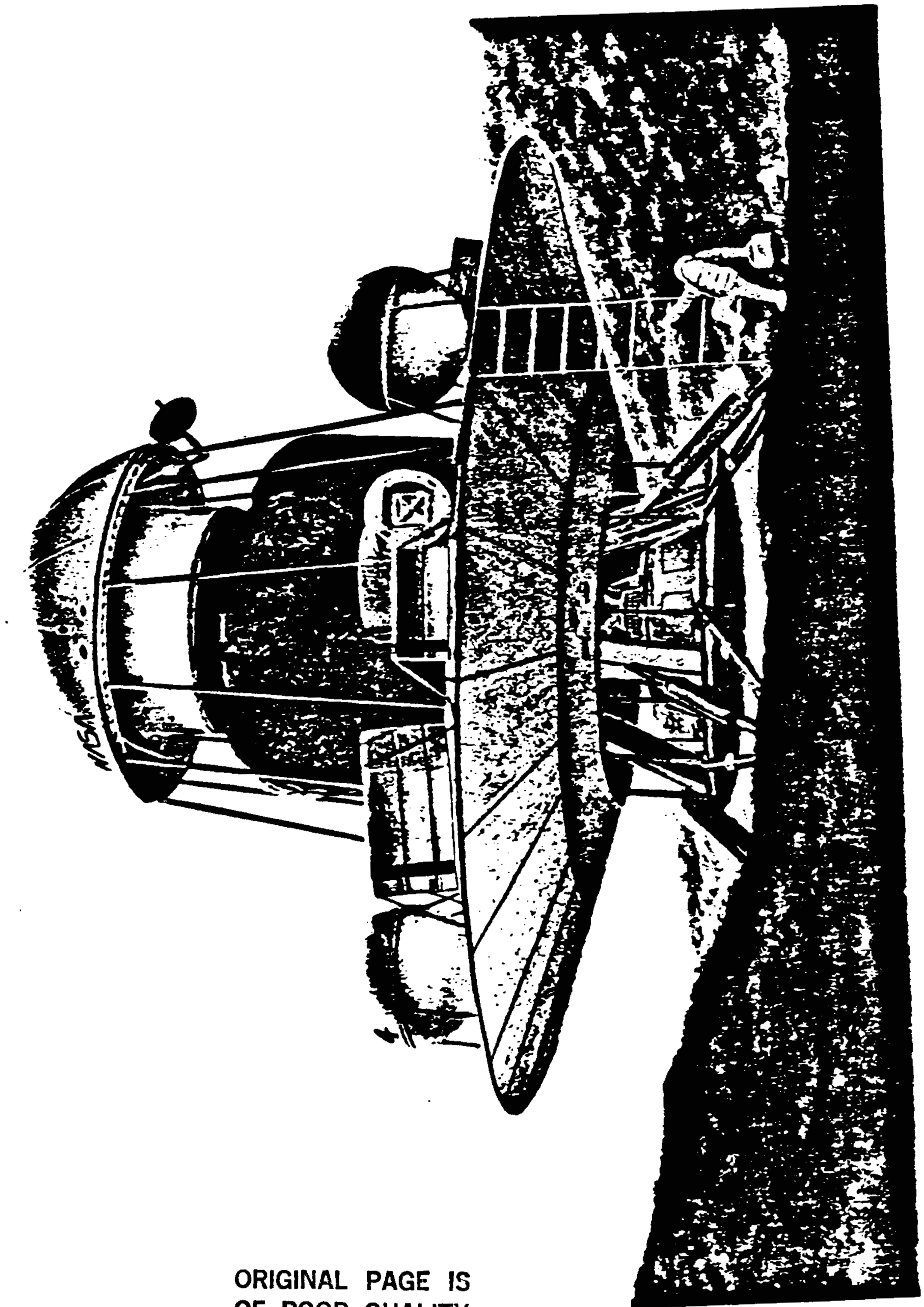

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Lunar Propellants for Mars Missions

- O Need hydrogen production on lunar surface
- O Propellant launch location switches to Moon
- O Cost of lunar launches is key number

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Fig. 1 Launch Requirements Versus Scenario

Oxygen Production Only

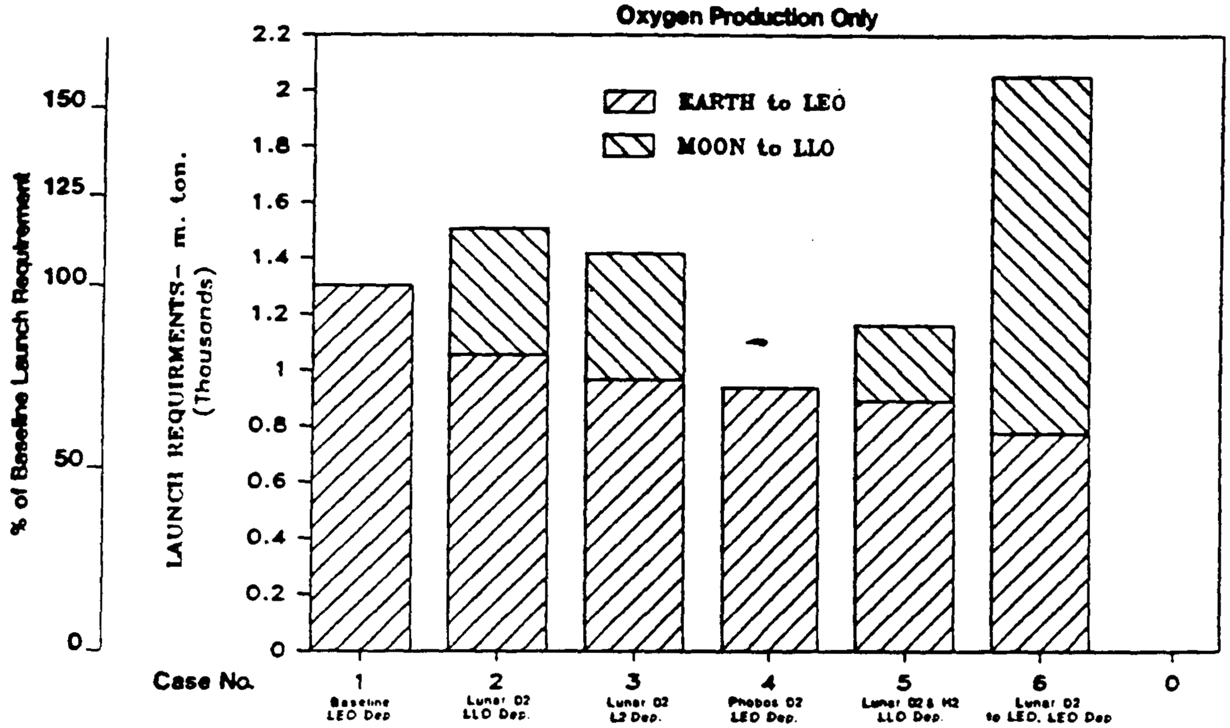


Fig. 2 Launch Requirements Versus Scenario

Oxygen and Hydrogen Production

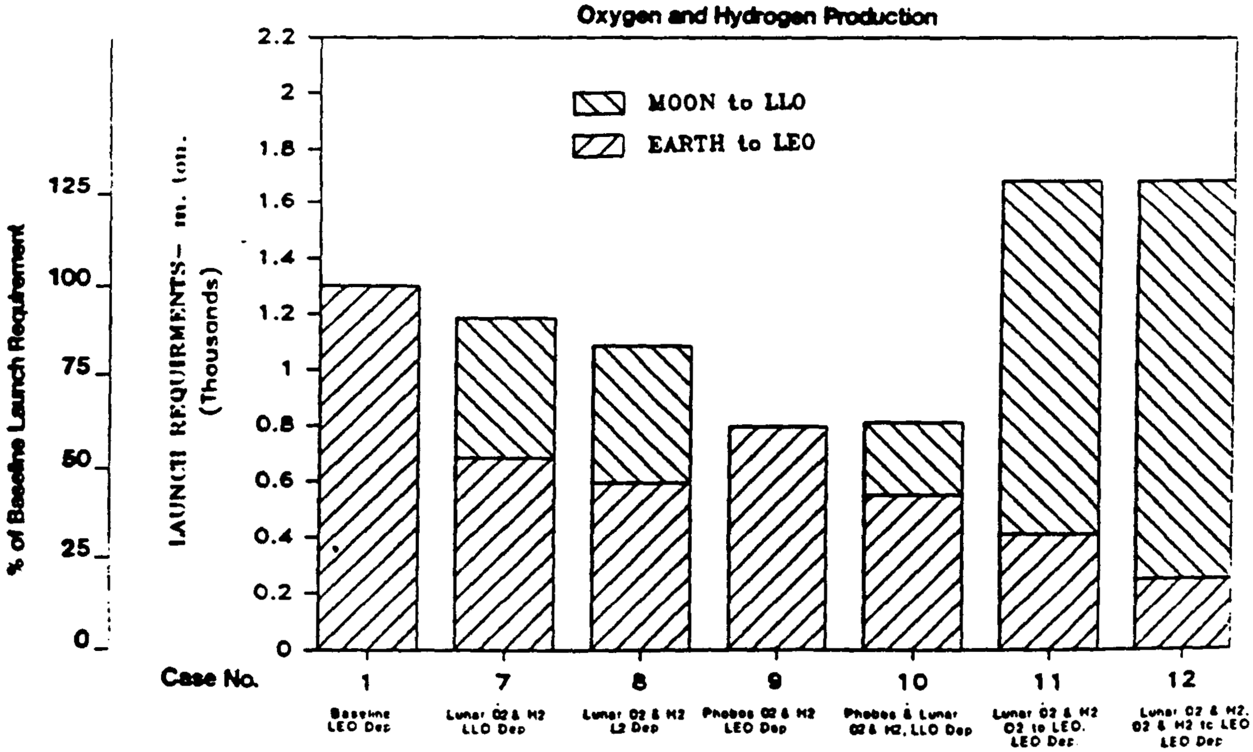


Fig. 3 Launch Cost Ratios

Oxygen Production Only

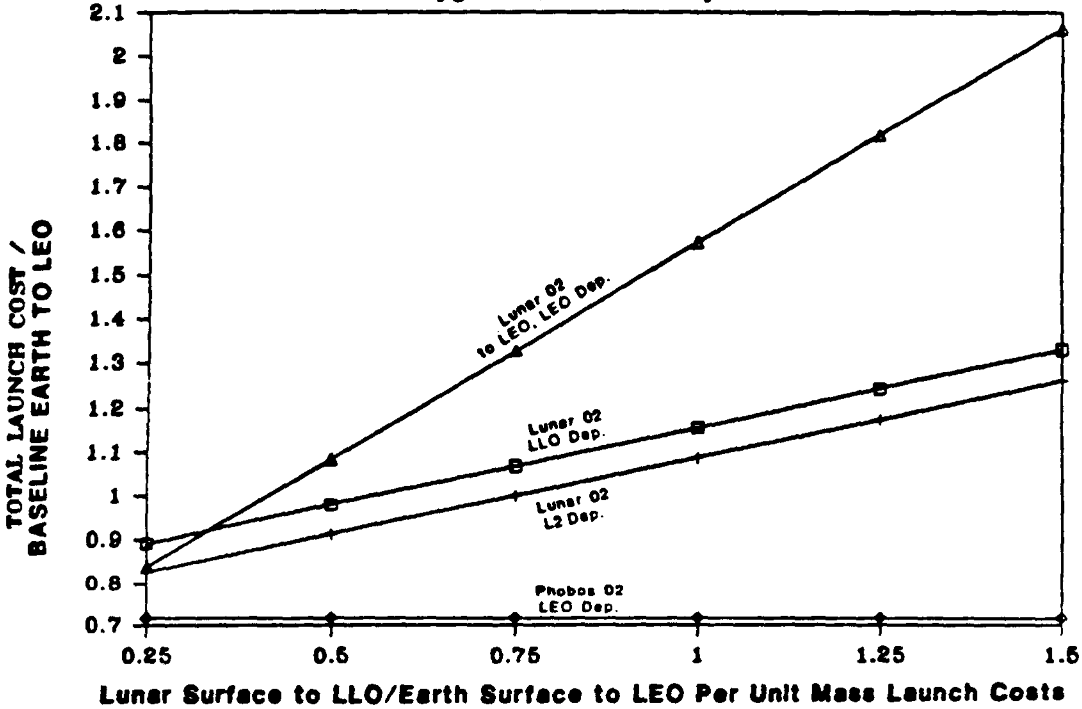
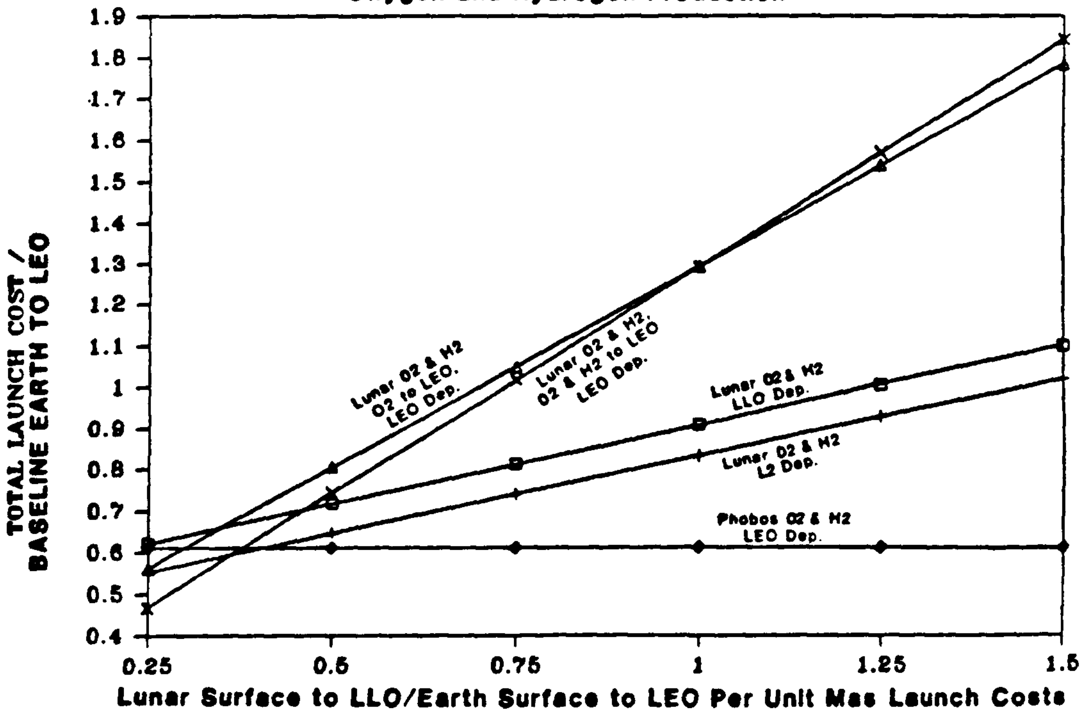
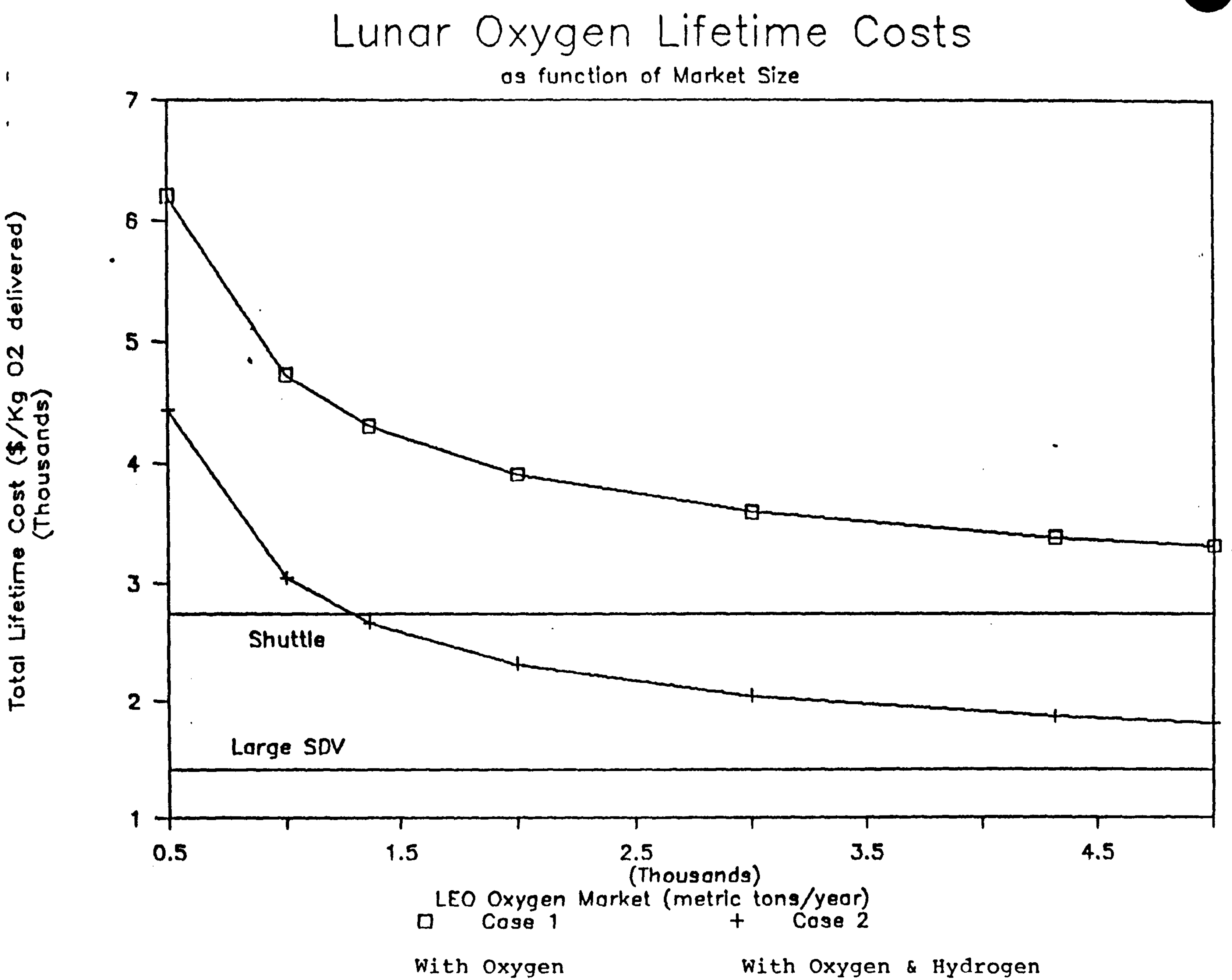


Fig. 4 Launch Cost Ratios

Oxygen and Hydrogen Production




Lunar Oxygen Lifetime Costs
as function of Market Size
Total Lifetime Cost ($/Kg O2 delivered)
(Thousands)
7
6
5
4
3
2
1
0.5
1.5
2.5
3.5
4.5
(Thousands)
LEO Oxygen Market (metric tons/year)
Shuttle
Large SDV
□ Case 1
+ Case 2
With Oxygen
With Oxygen & Hydrogen

References

- Babb, Stump, The Effect of Mars Surface and Phobos Propellant Production on Earth Launch Mass, Manned Mars Mission Working Group Papers, Vol. 1, NASA M002, June 1986.
- Babb, Stump, The Use of Lunar Produced Propellants for Manned Mars Missions, Manned Mars Mission Working Group Papers, Vol. 1, NASA M002, June 1986.
- Stump, Christiansen, Babb, Analysis of Lunar Propellant Production, Eagle Report No. 85-103B, Dec. 9, 1985

C-11

Conceptual Design of a Lunar Lander

NASA Participants: Stecklein, Petro
Eagle Participants: Stump, Morris, Varner, Rawlings, Yodzis, Chambers, Hirasaki, D'Onofrio, Nudd, and Zimprich

NASA Contract NAS9-17878, Lunar Base Systems Study
Reference - Eagle Report No. 88-181

Eagle Engineering, Houston
April 5, 1988

EAGLE

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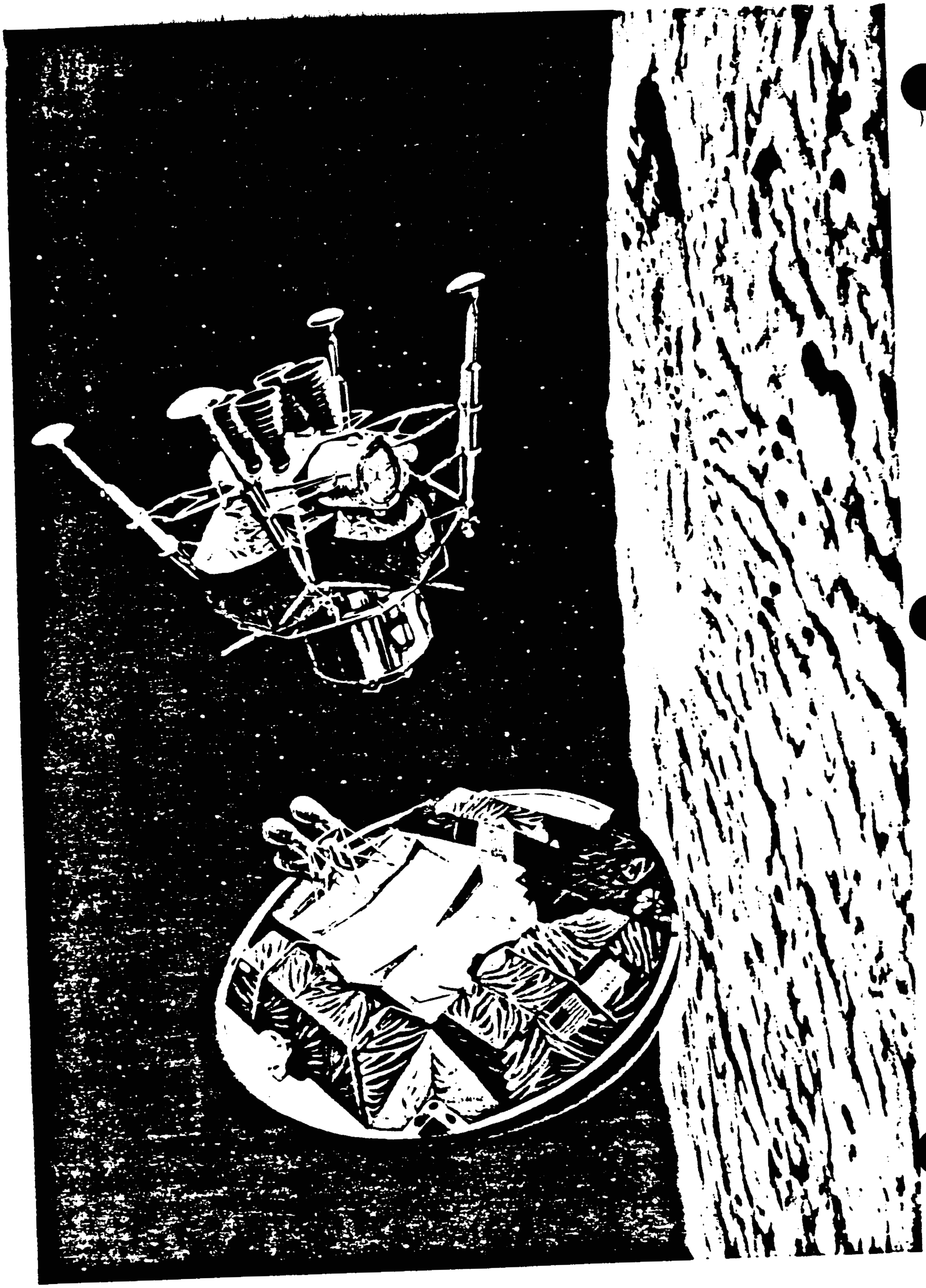


Key Results:

- Single stage is possible from low lunar orbit
  - Penalty = 15 to 30% of deorbit mass over 2 stage
  - Single stage desired for reusable, space maintainable vehicle

- Single Stage OTV to deliver single stage lander is possible
  - 5 to 15% more massive than a two stage stack
  - Single stage preferable for reusable, space based
  - Sized for lunar delivery, not GEO delivery

- Lander refurb. and propellant loading at Space Station recommended
  - OTV returns lander to Space Station after every mission
  - Penalty of 10 to 20 % over lander remaining in lunar orbit

Key Results, Continued

- A multi-purpose lander is possible from low lunar orbit
  - 25 m ton one way cargo down
  - 6 m ton crew module round trip
  - 17 m ton down, inert back to orbit
  - Penalty is on the order of 5 to 10 % over dedicated design

- The engine is the key long term development item
  - Pump fed due to need for regen. cooling and throttling
  - Must be maintainable/replaceable in space
  - Studies should continue until this engine is defined
  - Two to Four engines
  - Total thrust - 35,000 lbf, 15 or 20 to 1 throttling

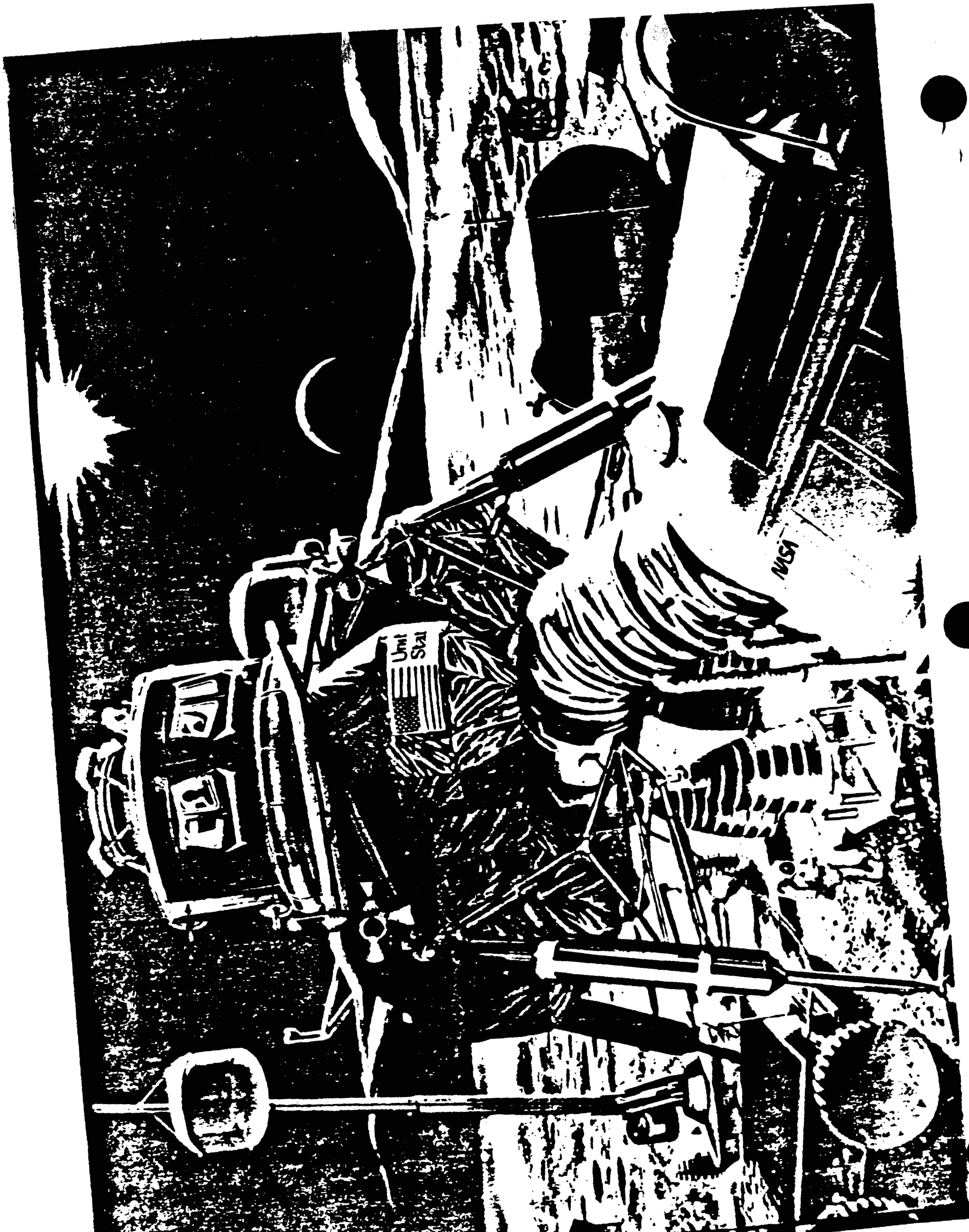
Key Results, Continued

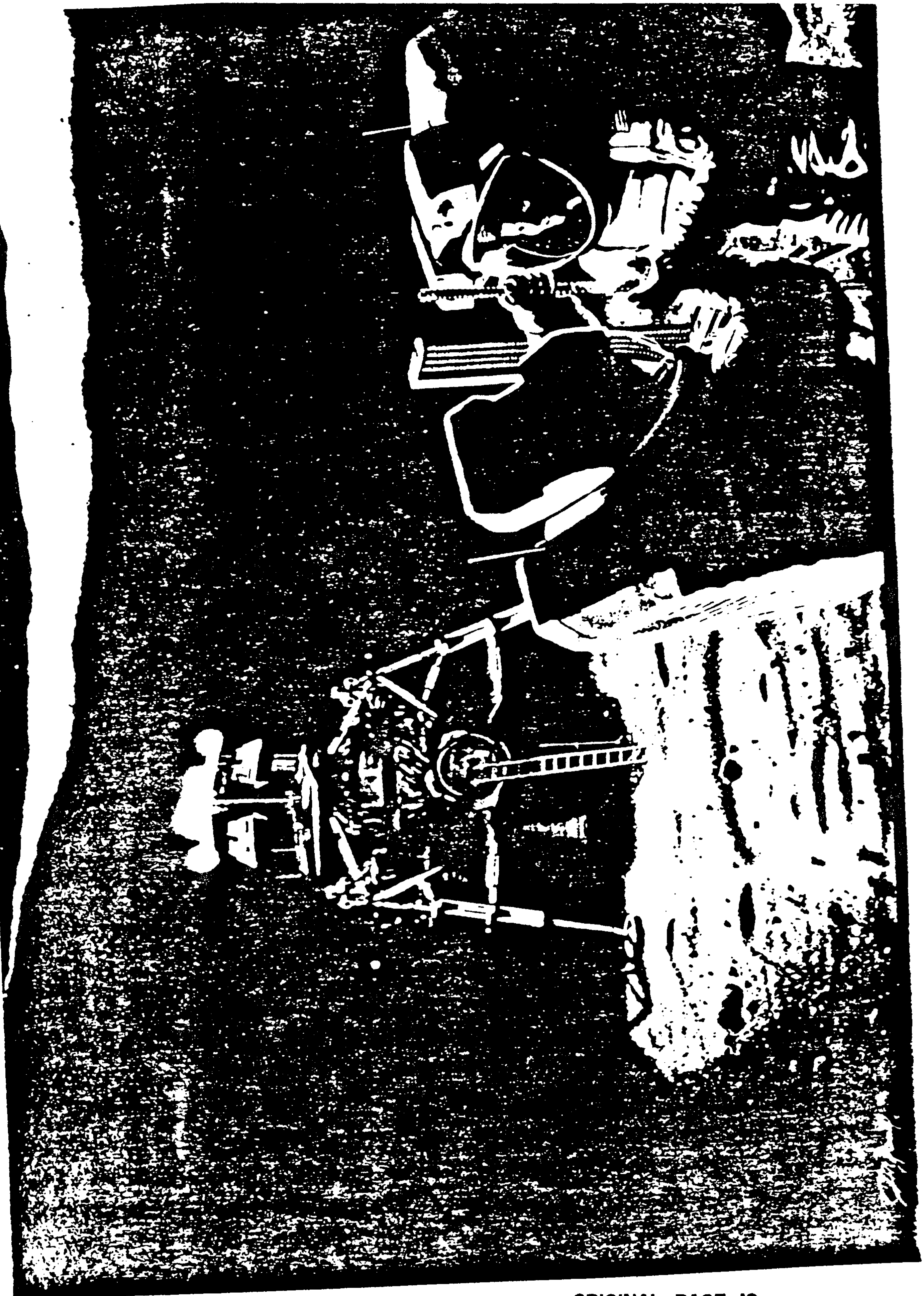
- Maintenance and Reusability must be made priorities
  - Without reusability, entirely different design needed
  - Money and effort typically not expended on these problems

- Liquid Oxygen/Hydrogen propellants favored, but:
  - Hydrogen storage for 6 month stay on surface needs look
  - Good point design needed to make sure performance gain is real
  - Inert mass is higher

- MMH/Nitrogen Tetroxide and Oxygen/hydrocarbon not ruled out

- Low lunar parking orbits result in minimum LEO stack mass
  - Question of stability for lower orbits

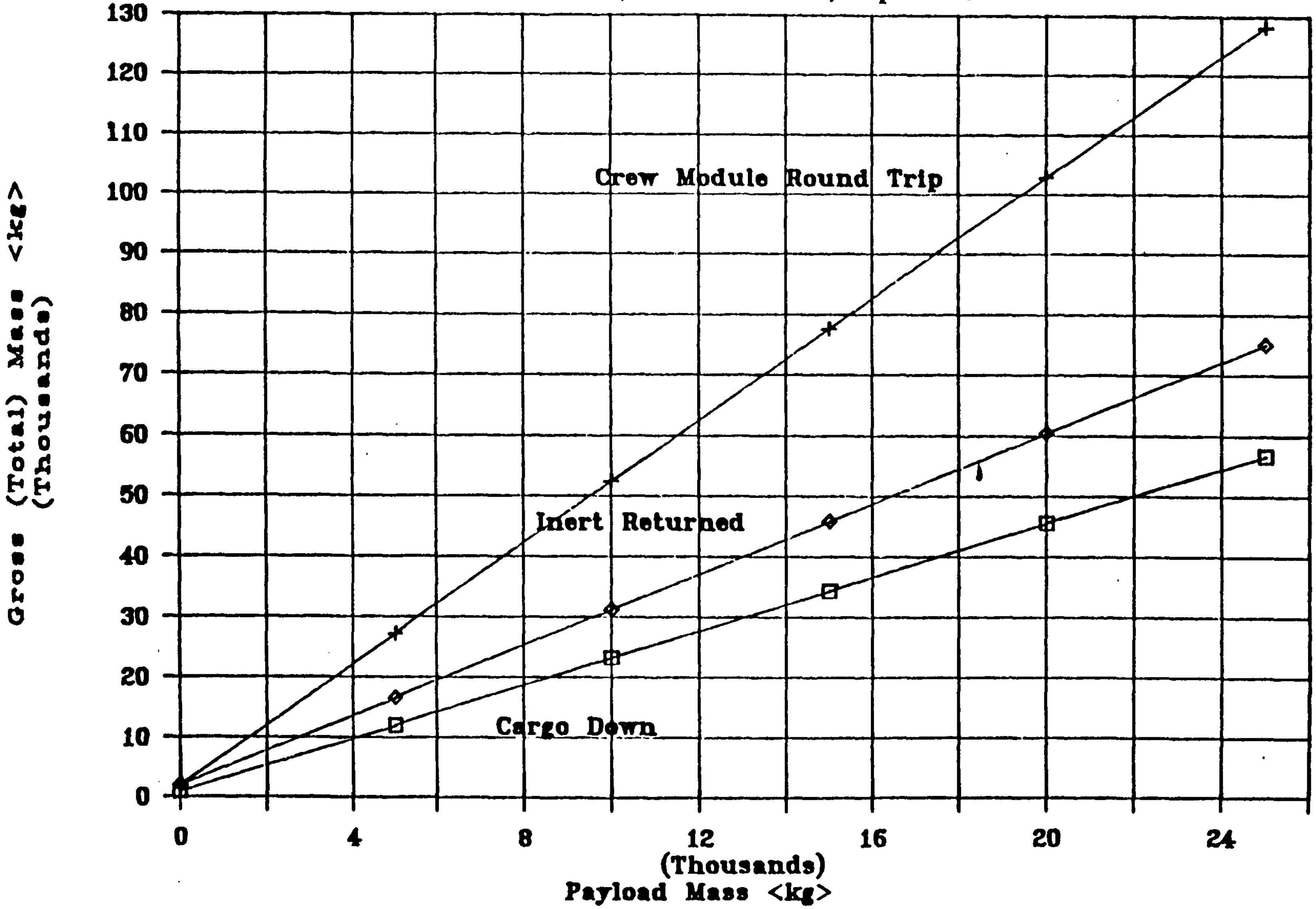

NASA



- Tools Developed and Collected
  - Scaling equations for inert mass based on Apollo lander developed
  - Plots produced
  - Detailed weight statements of Apollo lander
  - Annotated bibliography of Apollo literature on subject
- Open Loop ECLSS recommended
- GN&C system conceptually designed
  - Automated docking and rendezvous needed
  - Cruise missile terrain following radar navigation updates

Single Stage Crew/Cargo Lander

Orbit = 93 km;MR = 6.0 O/H; Isp = 450



Recommendations:

- Continue work on subject until the engine is well defined
- Relationship between throttling and regenerative cooling needs work
- Need good point design for O_2/H_2 vehicle to verify performance gain
- Thermal analysis of lander for long surface stay needed

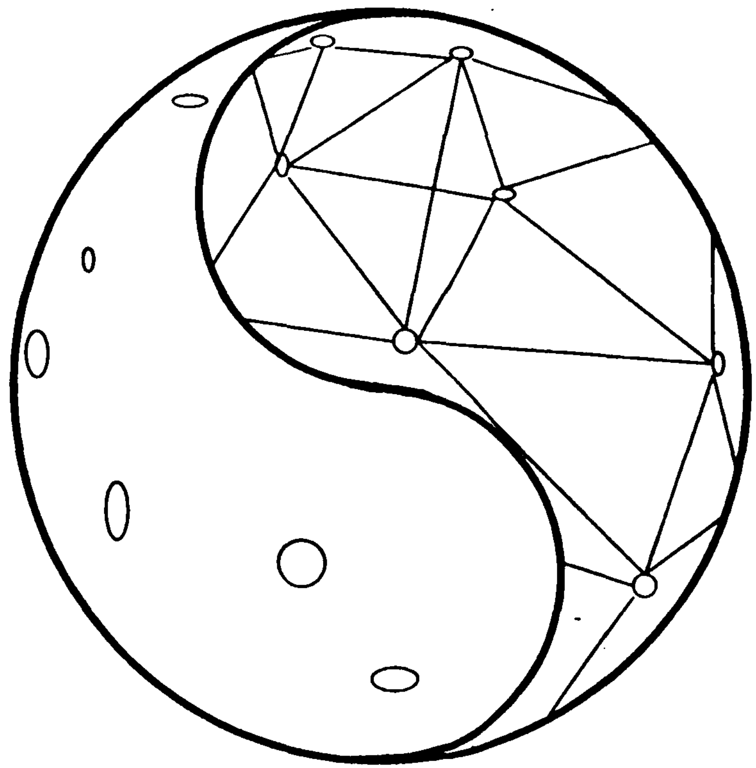
Lunar Stepping-Stones to a Manned Mars Exploration Scenario

W.L. Davidson, W.R. Stump

Eagle Engineering/Lockheed, Houston

April 5, 1988

COMBINED LUNAR-MARS EXPLORATION SCENARIO



Lunar Stepping-stones to Mars

April 7, 1988
Eagle Engineering, Inc.
W.L. Davidson

COMBINED LUNAR-MARS EXPLORATION SCENARIO

THE CHALLENGE

- o The Mars Trip is Measured in Years
- o We Do Not Have Space Systems Which Can Sustain Life Away From Our Home Planet For Years
- o First Irreversible Severing of Human Dependence on Earth's Environment
- o Communications Takes 15 to 20 Minutes to Receive a Response

April 7, 1988
Eagle Engineering, Inc.
W.L. Davidson

COMBINED LUNAR-MARS EXPLORATION SCENARIO

STRATEGY

- Go to Mars With Humans; Return to the Moon With Emphasis on Science and Resources Applications
- Develop Credible Planetary Systems Proven With Space Experience
- Use Moon as The Proving Ground Which Enables a Safe Journey to Mars
- Learn Planetary Operations in Safe Lunar Environment
- Using Lunar Experience, Separate From Earth for Extended Exploration at Mars
- Continue Planetary Systems Development and Establish a Human Mars Colony

April 7, 1988
Eagle Engineering, Inc.
W.L. Davidson

COMBINED LUNAR-MARS EXPLORATION SCENARIO

LUNAR STEPPING-STONES

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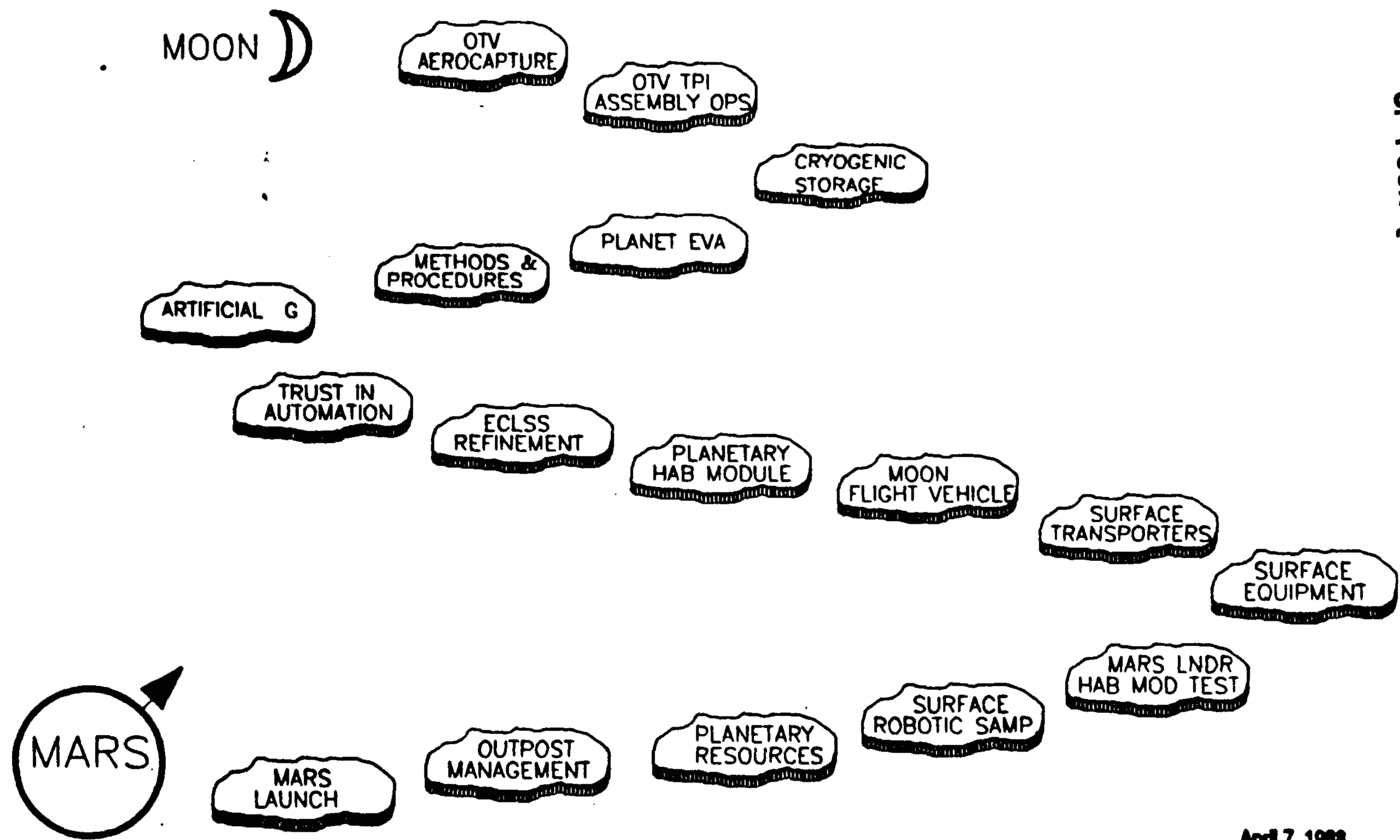

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W.L. Davidson

COMBINED LUNAR-MARS EXPLORATION SCENARIO

COMMON OPERATIONS, METHODS, AND PROCEDURES (STEPPING-STONES)

- In-space OTV Assembly and Transplanetary Injection (TPI) Operations
- EVA-based Planetary Field Operations
- On-site Planetary Science and Sample Analysis Methods and Procedures
- Planetary Construction Methods and Procedures
- Artificial Gravity Environment Verification and Operations Adaptation
- Trusting Dependence on Automated Systems
- Proven Planetary Outpost Site Management

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COMBINED LUNAR-MARS EXPLORATION SCENARIO

COMMON SYSTEMS DESIGN (STEPPING-STONES)

- OTV Aerocapture Aerobrake Structures, Materials, and Operations
- Cryogenic Propellant In-space Storage and Transfer Systems
- Artificial Gravity Systems
- Life Support Systems Refinement
- Planetary Habitation Module (PHM)
- Moon Flight Vehicle (MFV)
- Local Surface Transportation Vehicle (LOTRAN)
- Mobile Surface Applications Traverse Vehicle (MOSAP)
- Surface Equipment for Science and Construction
- Mars Lander Habitation Module
- Planetary Surface Robotic Sampler
- Planetary Resources Plant Equipment

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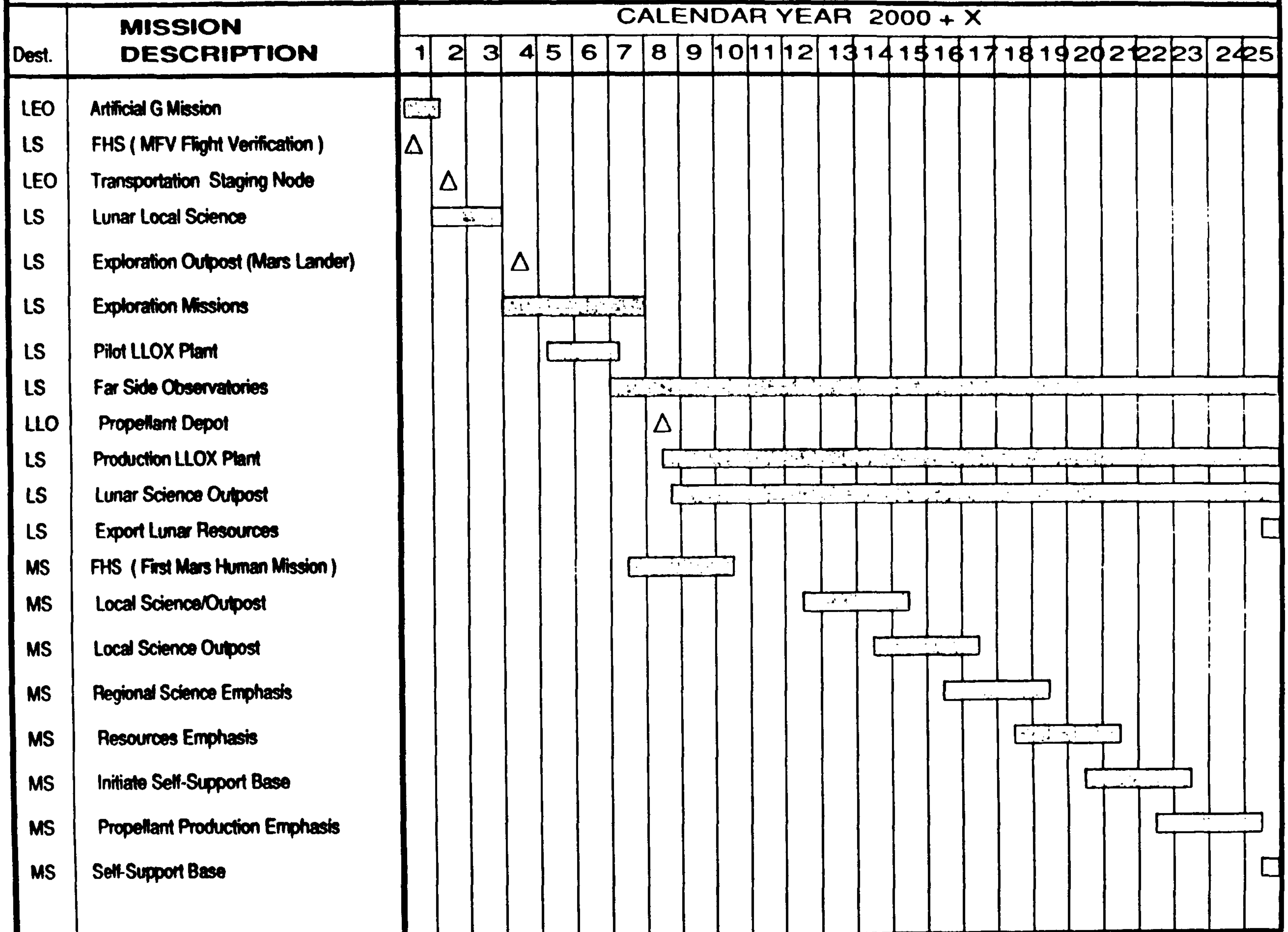
W.L. Davidson

March 28, 1988

Flight Schedule: Combined Lunar-Mars Exploration Scenario



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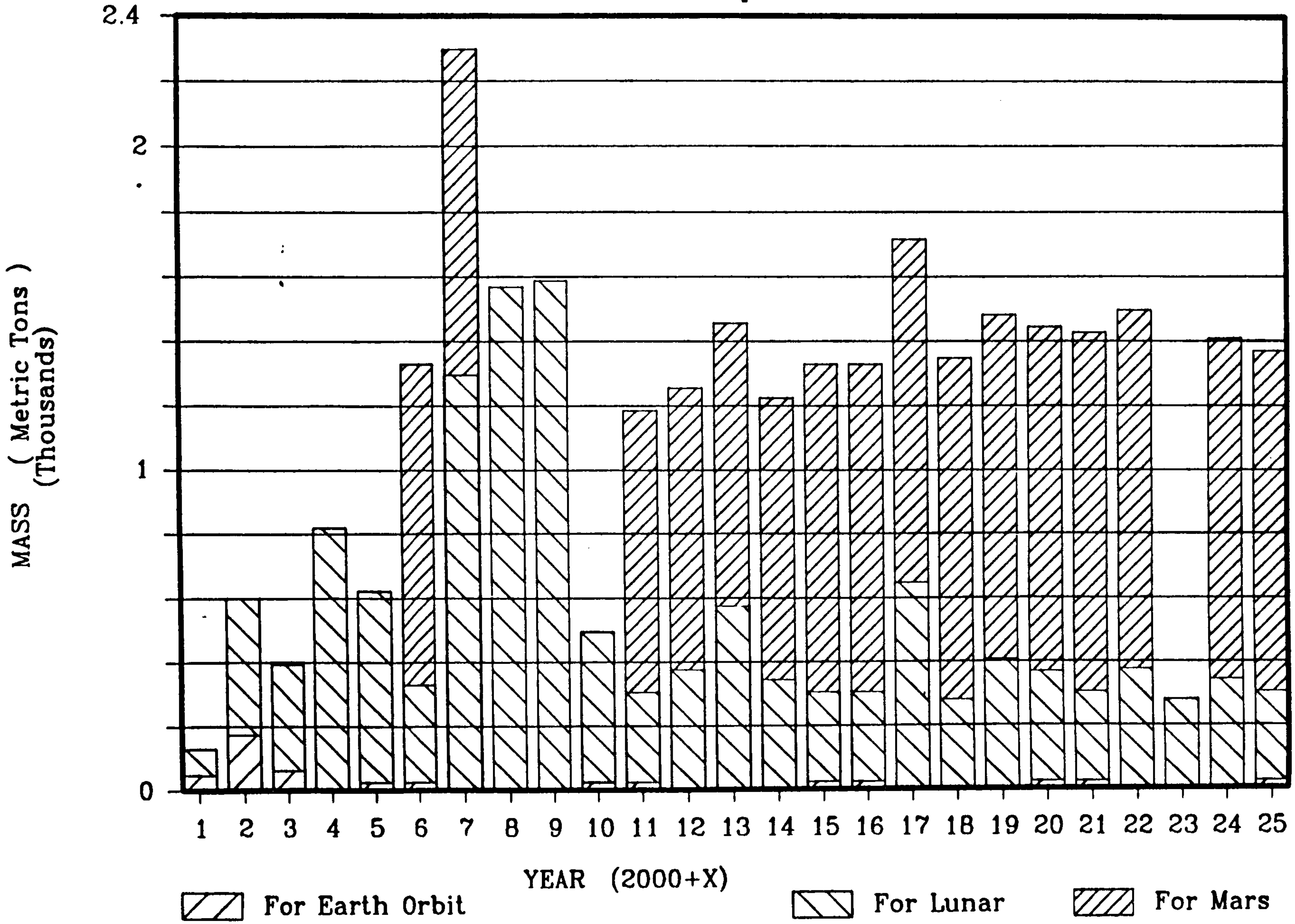
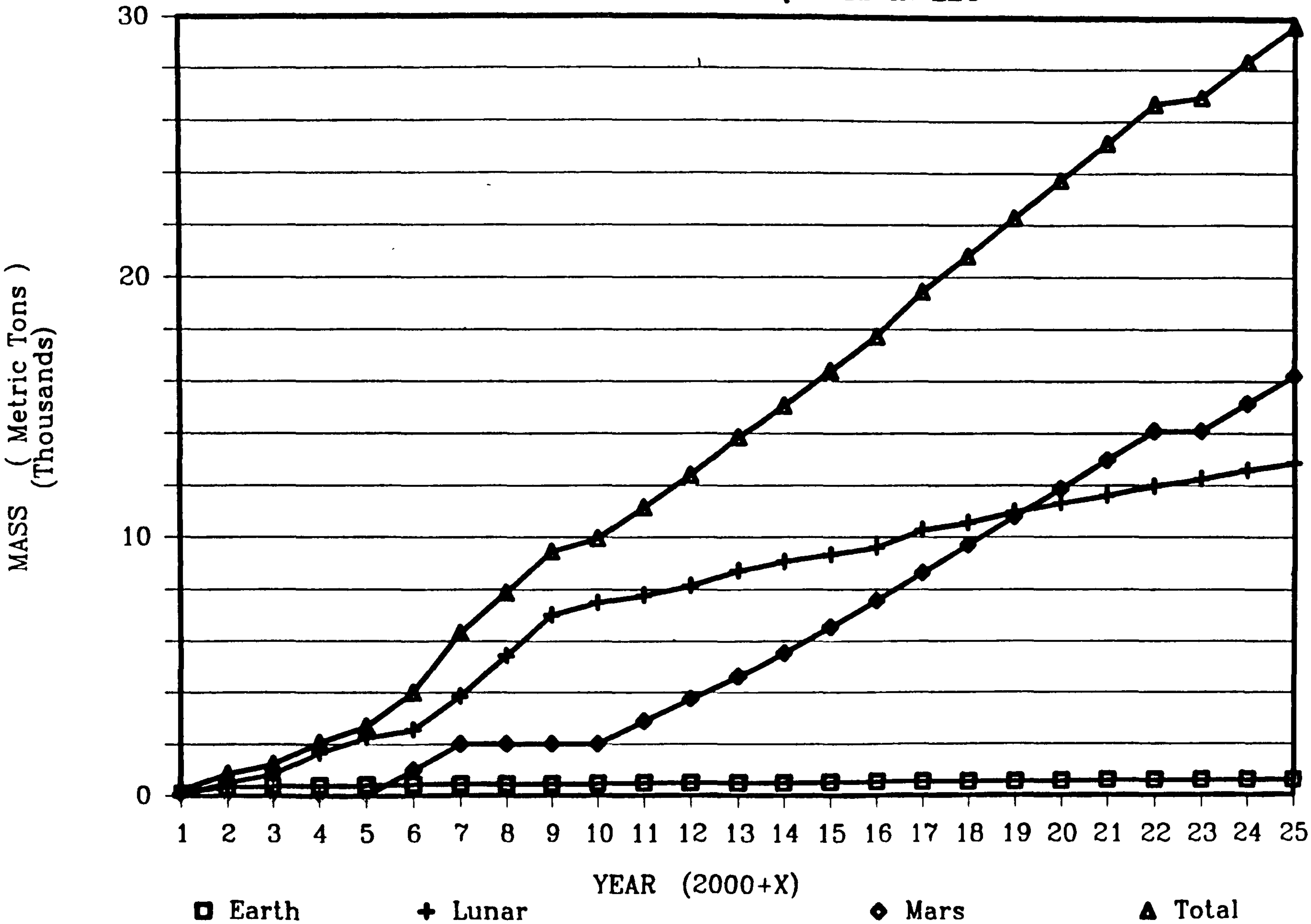
COMBINED LUNAR—MARS SCENARIO

ANNUAL MASS REQUIRED IN LEO



COMBINED LUNAR–MARS SCENARIO

CUMULATIVE MASS REQUIRED IN LEO



COMBINED LUNAR-MARS EXPLORATION SCENARIO

APPROACHES TO REDUCING REQUIRED MASS IN LEO

- o Lunar Oxygen Use at the Moon
- o Lunar Oxygen Delivered to Earth Orbit
- o Phobos Oxygen
- o Mars Oxygen
- o Mars Departure From Lunar Vicinity
- o Other Propulsion Systems

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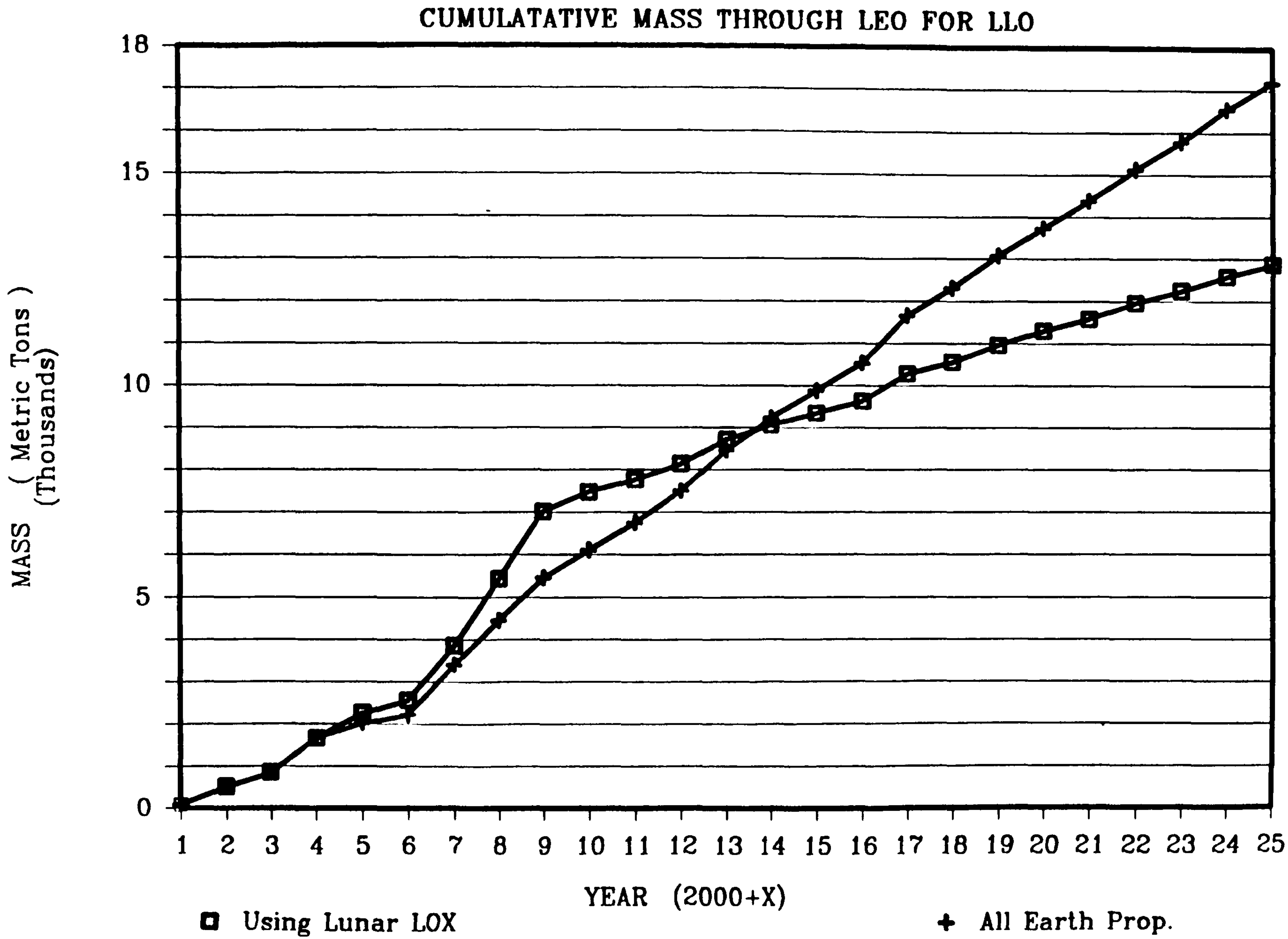
COMBINED LUNAR–MARS SCENARIO



COMBINED LUNAR-MARS EXPLORATION SCENARIO

SUMMARY

- o Combined Lunar-Mars Scenario Has Been Developed

  Science (Lunar), Resources (Lunar), Human Presence (Mars)

- o Common Links Between Lunar and Mars Exploration Described

- o Scenario Implementation Synthesis Methodology Developed

- o Implementation Impact Displayed

April 7, 1988
Eagle Engineering, Inc.
W.L. Davidson

LUNAR SURFACE BASE PROPULSION SYSTEMS STUDY

FINAL
PROGRESS REVIEW

24 OCTOBER 1986

CONTRACT NO. NAS 9-17468

ASTRONAUTICS CORPORATION OF AMERICA
ASTRONAUTICS TECHNOLOGY CENTER

AEROJET TECHSYSTEMS
UNIVERSITY OF WISCONSIN

This represents documentation for the Final Progress Review of the Lunar Surface Base Propulsion System Study, Contract No. NAS 9-17468. The contract effort was initiated on 15 January 1986 and will continue through 15 December 1986.

The study is being conducted by the Astronautics Corporation of America - Technology Center in Madison, Wisconsin. Aerojet TechSystems of Sacramento, California is a subcontractor contributing propulsion and propellant analyses. Additional contributions have been made by the Engineering Mechanics, Nuclear Engineering, and Chemistry Departments of the University of Wisconsin .

Comments or questions concerning this study effort should be directed to the NASA Technical Monitor, Leo R. Johnson at the Johnson Space Center; or to the Astronautics Project Manager, Ronald R. Teeter; or to the Deputy Project Manager, Thomas M. Crabb:

Leo R. Johnson	Ronald R. Teeter
Mail Code: EP4	Thomas M. Crabb
NASA / Johnson Space Center	Astronautics Technology Center
Houston, TX 77058	5800 Cottage Grove Road
	Madison, WI 53716
(713)483-5495	(608)221-9001

AGENDA

INTRODUCTION	LEO JOHNSON
EXECUTIVE SUMMARY	RON TEETER
RESULTS	
TASK 1	TOM CRABB
TASK 2	RON TEETER
TASK 3	TOM CRABB
TASK 4	RON TEETER
CONCLUSIONS / RECOMMENDATIONS	RON TEETER
RECOMMENDATIONS	RON TEETER
INFORMAL DISCUSSION	ALL

STUDY OBJECTIVE

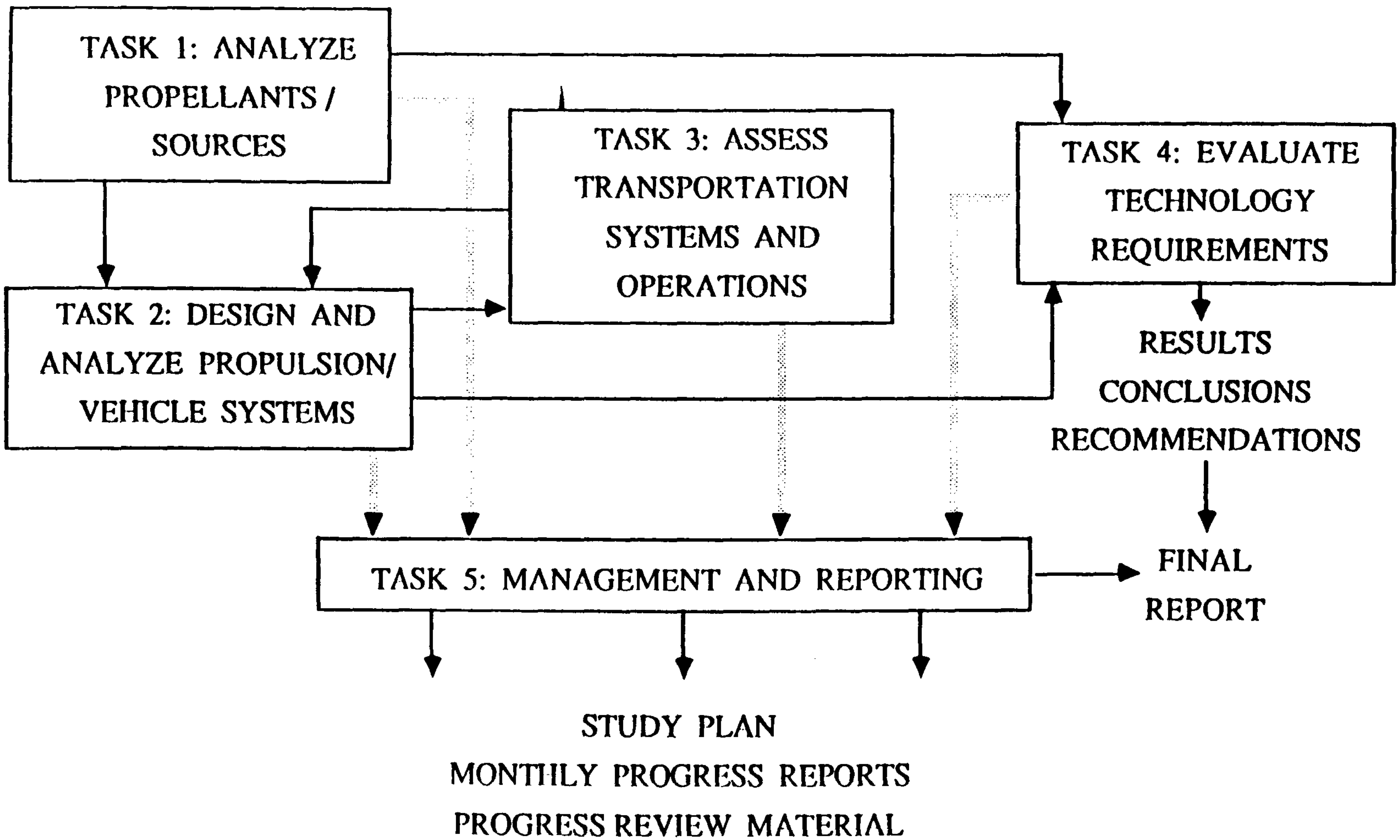
IDENTIFY ECONOMICAL ALTERNATIVES FOR TRANSPORTATION BETWEEN LOW EARTH ORBIT AND A LUNAR SURFACE BASE. SPECIFIC OBJECTIVES INCLUDE:

- IDENTIFY VALUABLE PROPELLANT ALTERNATIVES - VALUE DEPENDENT ON SOURCE LOCATION AND PROCESSING REQUIREMENTS
- DEFINE PROPULSION/VEHICLE ALTERNATIVES
- ASSESS TRANSPORTATION ALTERNATIVES AND TRADEOFFS
- RECOMMEND TECHNOLOGY DEVELOPMENT OPTIONS

APPROACH

SPACE SYSTEMS

ACA — Astronautics Corporation of America - Technology Center — ATC



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EARTH-MOON TRADES AND OPTIONS

SPACE SYSTEMS

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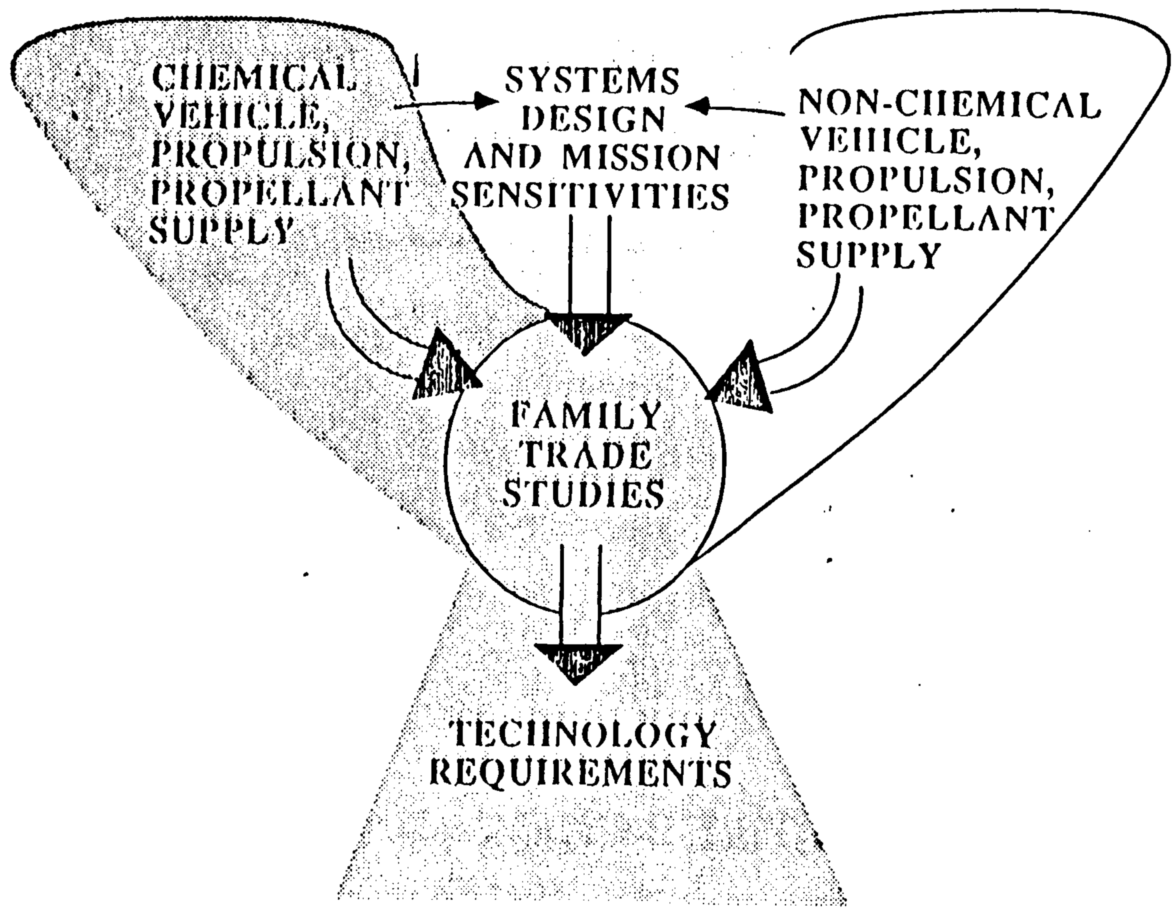

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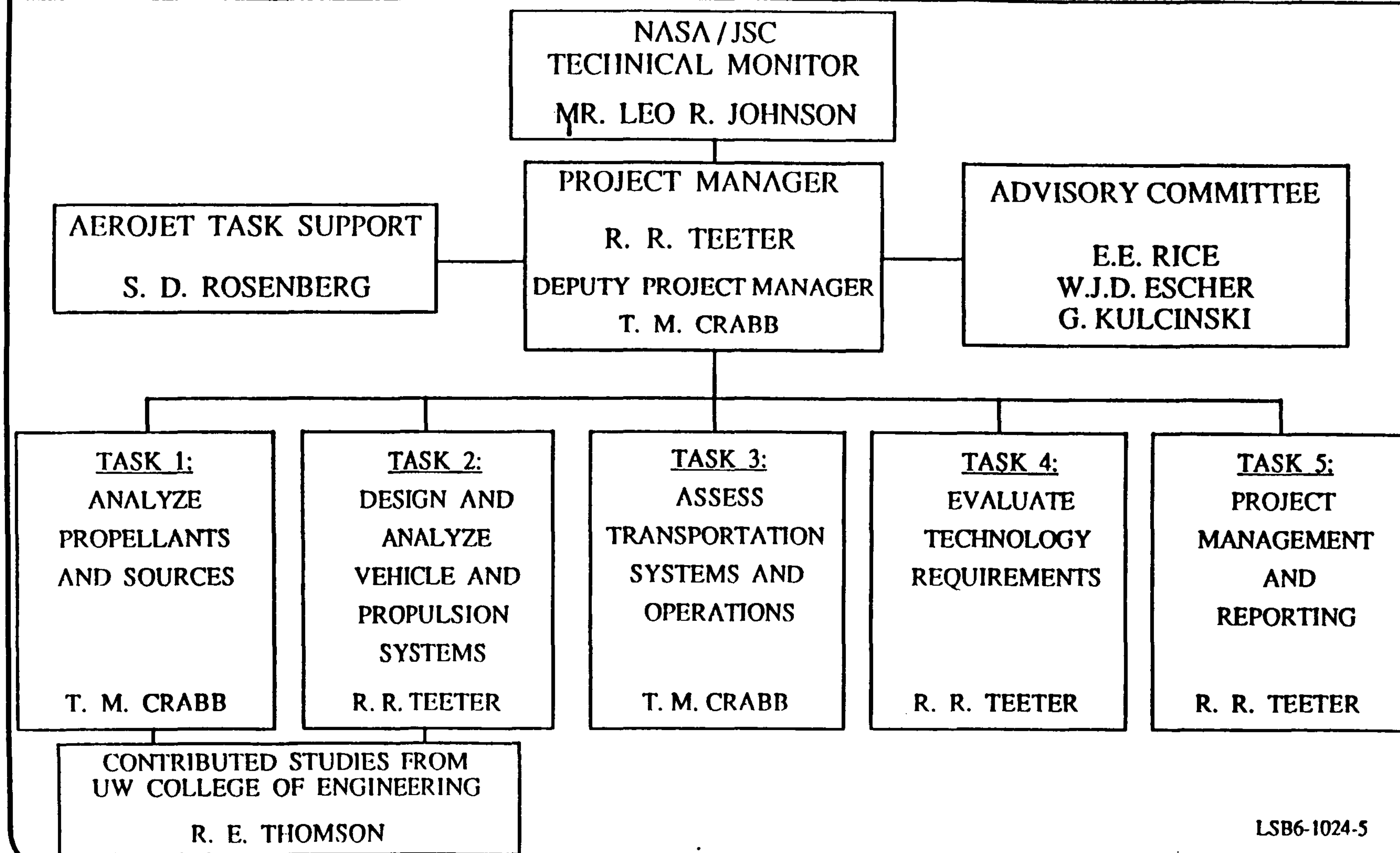
PROJECT ORGANIZATION

ACA — Astronautics Corporation of America - Technology Center — ATC

SPACE SYSTEMS



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TASK 1: OBJECTIVE
ANALYZE PROPELLANTS/SOURCES

SPACE SYSTEMS

TO IDENTIFY THE FULL RANGE OF PROPELLANT SOURCES FROM SPACE SHUTTLE SCAVENGING TO LUNAR RESOURCES, AND DEFINE THE PROCESSING TECHNIQUES AND REQUIREMENTS FOR PRODUCING SUCH PROPELLANTS

- IDENTIFY CANDIDATE PROPELLANTS/SOURCES
- DETERMINE PROCESSING/STORAGE REQUIREMENTS
- DEVELOP PROPELLANT DATA BASE
- EVALUATE/SELECT PROPELLANTS/SOURCES
- REFINE PROPELLANT SUPPLY OPERATIONS

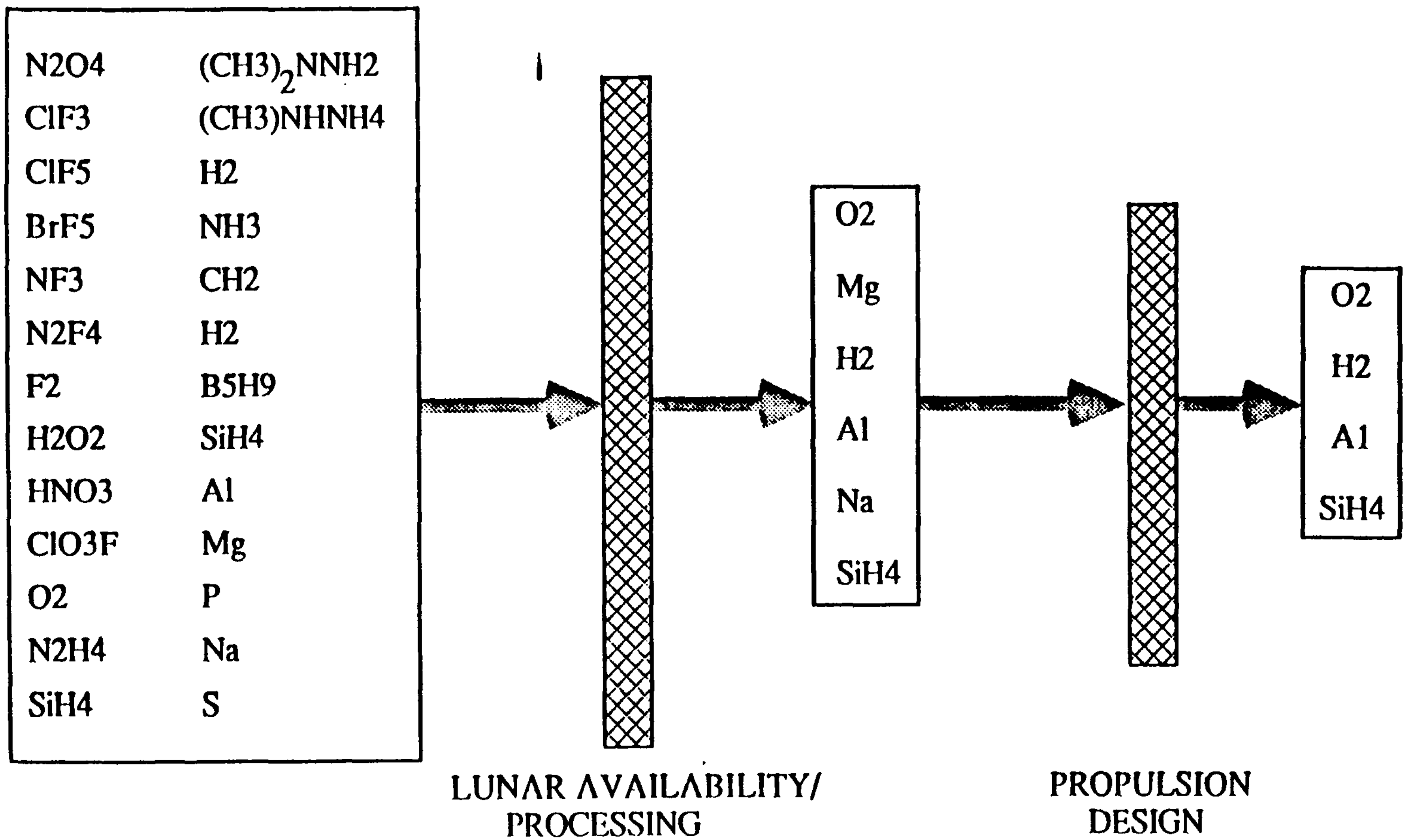
ACA

PROMISING CHEMICAL PROPELLANT CANDIDATES

Astronautics Corporation of America - Technology Center

ATC

SPACE SYSTEMS



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PROPELLANT PROCESSES
RESOURCE REQUIREMENTS

NOTE: All estimates based on 10 MT O2 per month

SPACE SYSTEMS

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	HYDROGEN REDUCTION O2 ONLY	ELECTROLYSIS O2 ONLY	CARBOCHLORINATION O2 & Al	HF LEACH O2 & Al	VAPOR-ION SEPARATION O2 & Al	CARBO-THERMAL LOX & Si
ELECTRIC ENERGY	43,200 kWhr	92,900 kWhr	78,500 kWhr	119,800 kWhr	67,000 kWhr	43,200kWhr
THERMAL ENERGY	7,600 kWhr	62,700 kWhr	83,700 kWhr	41,600 kWhr	501,000 kWhr	136,000kWhr
SYSTEM WEIGHT EST.	1800 kg	980 kg	13,400 kg	11,700 kg	10,000 kg	12,000kg (c)
SYNERGISTIC POTENTIAL	POOR	POOR - FAIR	GOOD	GOOD	EXCELLENT	FAIR
RESOURCES CONSUMED (kg/10MT - O2) • MARE SOIL	662,000	1,325,000	238,000	87,000	39,700	131,700,000
• ADDITIVES/CONSUMABLES	300	250 (a)	112,000 (b)	53,700 (a)	500 (b)	2,000
REACTANT RECOVERY POTENTIAL (%)	95	NA	70	70	NA	80
ACQUISITION EFFICIENCY [(O2 RECOVERED/ O2 AVAILABLE IN MARE) X 100]	3.7	1.7	10.2	27.8	61.0	0.016 (d)

(a) Caustic fluids may induce additional equipment replacement

(b) High temperatures may induce additional equipment replacement, also plasma sustenance required for selective ionization vapor-ion separation (e.g. Argon)

(c) Estimated using research conducted by Rosenberg, et. al.

(d) Acquisition efficiency for Si

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MOST VALUABLE CHEMICAL PROPELLANTS / PROCESSING

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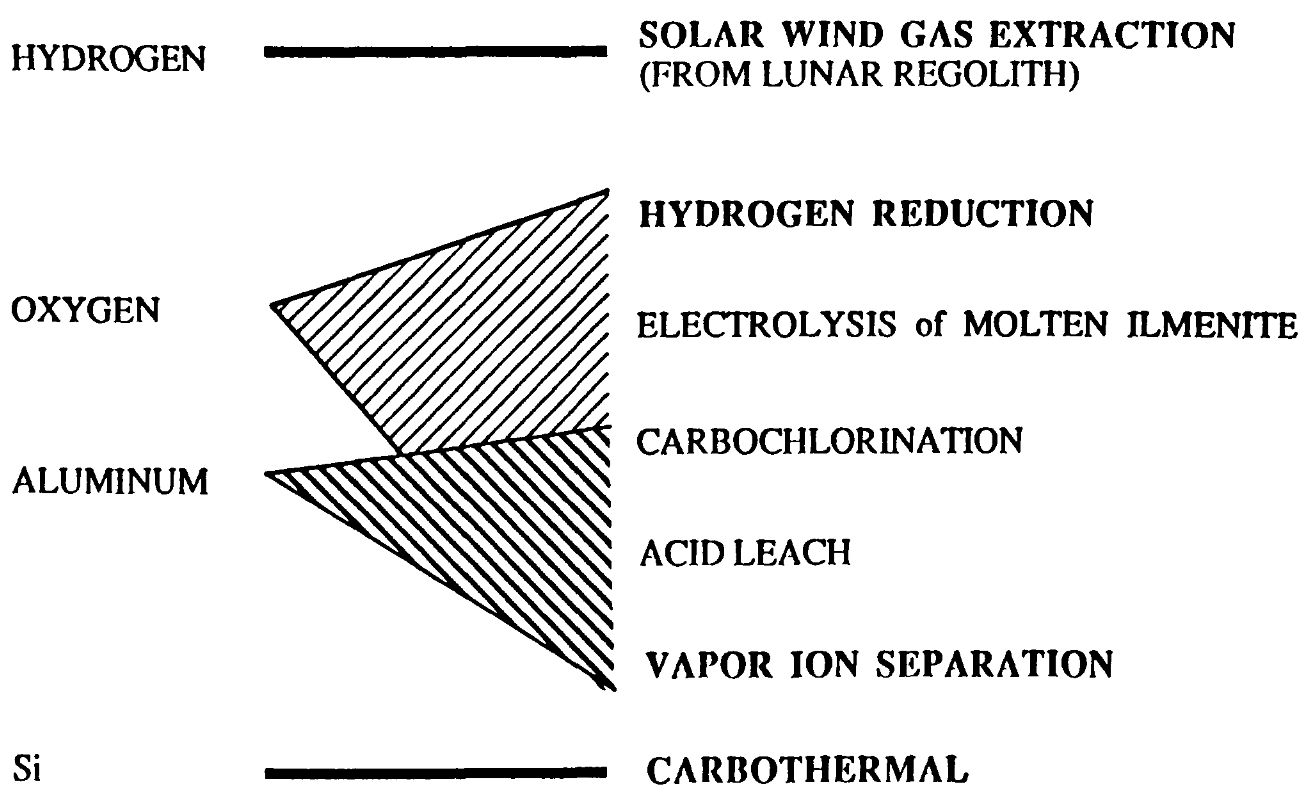
SPACE SYSTEMS



TASK 2: OBJECTIVE
DESIGN AND ANALYZE VEHICLE / PROPULSION SYSTEMS

TO IDENTIFY AND SELECT PROPULSION SYSTEMS CAPABLE OF SUPPORTING THE LUNAR MISSION TRANSPORTATION MODEL USING THE PUMP-FED LOX/LH2 SYSTEM AS A BASELINE FOR COMPARISON.

- DEFINE PROPULSION AND VEHICLE SYSTEMS REQUIREMENTS TO ACCOMPLISH MISSION MODEL
- IDENTIFY CANDIDATE VEHICLE / PROPULSION SYSTEMS
- CONCEPTUALLY DESIGN PROMISING CANDIDATES
- REFINE PROPULSION/VEHICLE SYSTEMS DESIGNS

ACA

TRADEOFFS AND SENSITIVITIES

SPACE SYSTEMS

Astronautics Corporation of America - Technology Center

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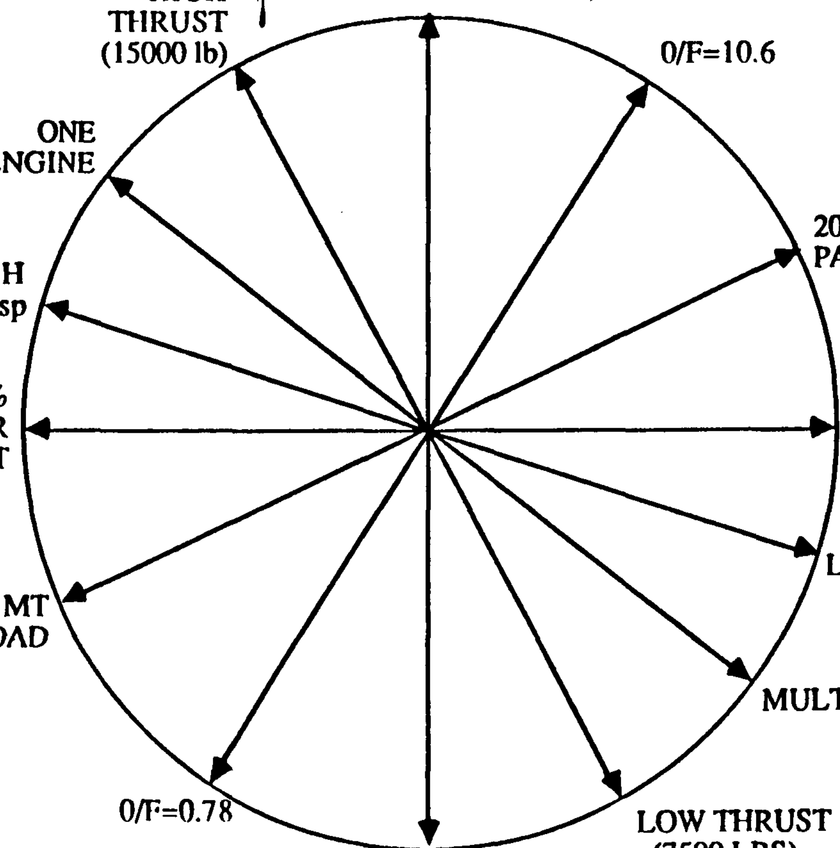

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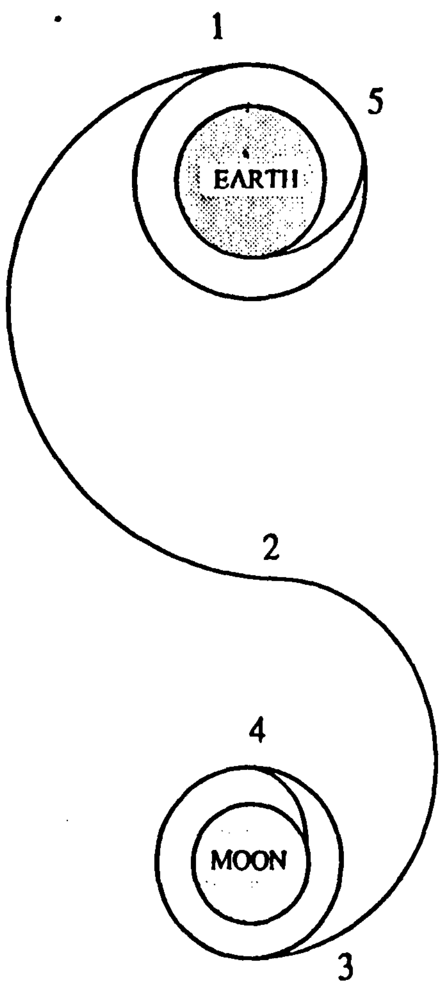
ASSUMED EARTH - MOON TRANSPORTATION REQUIREMENTS

SPACE SYSTEMS

ACA Astronautics Corporation of America - Technology Center ATC



1 3.15 km/sec LEO TO TRANSLUNAR

2 0.05 km/sec MIDCOURSE CORRECTION

3 0.97 km/sec TRANSLUNAR TO LLO

4 2.10 km/sec LLO TO LUNAR SURFACE

5 0.09 km/sec TRANSLUNAR TO LEO WITH AEROBRAKE

TOTAL TRIP FROM LUNAR SURFACE TO LEO

3.21 km/sec WITH AEROBRAKE

6.27 km/sec WITHOUT AEROBRAKE

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BASELINE STAGE DESIGN ASSUMPTIONS

- H/O STAGES MODELED AFTER AEROJET DUAL PROPELLANT EXPANDER CYCLE
- ENGINE
  - Isp = 470 sec
  - H/O O/F = 5.5
  - Pc = 2000 psi
  - Tc = 6515 R, 3346 C, 3619 K
- TANKAGE
  - 301 CRES TANKS WITH 0.014 in MINIMUM GAGING
  - ELLIPTICAL END TANKS WITH ELLIPSE RATIO 1.38
  - PRESSURIZED AUTOGENOUSLY
- AEROBRAKE MASS ASSUMED 15% OF REENTRY MASS
- MISCELLANEOUS WEIGHTS MODELED AFTER CENTAUR D-1

VEHICLE / PROPULSION WEIGHTS AND PERFORMANCE SUMMARY

SPACE SYSTEMS

ACA — Astronautics Corporation of America - Technology Center — ATC

NAME		I_{sp} (sec)	O/F	MASS FRACTION	DRY MASS (MT)	LUNAR LOX (MT)[c]	PROPELLANT SOURCE LEO	PROPELLANT SOURCE LSB
BASELINE H/O OTV & LANDER	(a) (b)	470 470	5.5 5.5	0.929 0.881	0.84 0.84		O,H	-
H/O OTV & LANDER WITH LLOX	(a) (b)	470 470	5.5 5.5	0.886 0.881	5.25 3.33	13.5	O,H	O
H/O OTV & LANDER WITH LLOX & LH_2	(a) (b)	470 470	5.5 5.5	0.872 0.811	4.37 3.33	13.0	O,H	O,H
H/O OTV & Al/LLOX LANDER	(a) (b)	470 260	5.5 2.18	0.871 0.912	3.70 10.29	9.7	O,H,Al	O,Al
SiH_4 / LLOX OTV & LANDER	(a) (b)	366 366	0.78 0.78	0.937 0.906	3.33 4.22	7.1	O,SiH4	O,SiH4
Al-H/LLOX OTV & & LANDER	(a) (b)	400 400	3.1 3.1	0.918 0.889	4.59 4.71	9.2	O,H,Al	O,Al

(a) OTV
(b) LANDER
(c) LLOX RETURNED TO LEO PER UNMANNED TRIP

ALL OTVS BASELINED WITH AEROBRAKE MASS OF 15% REENTRY MASS; UNMANNED/MANNED PAYLOAD CAPABILITY OF 15.9 MT AND 8.97 MT, RESPECTIVELY. EACH VEHICLE HAS A COROLLARY DESIGN TO OPERATE WITHOUT LUNAR PROPELLANTS.

LSB6-1024-12

TASK 3: OBJECTIVE
ASSESS TRANSPORTATION SYSTEMS AND OPERATIONS

SPACE SYSTEMS

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TO CONDUCT TRADE STUDIES AND RELATED ANALYSES TO DETERMINE WHICH PROPELLANTS AND PROPULSION / VEHICLE SYSTEMS ARE MOST EFFICIENT BASED ON ESTABLISHED CRITERIA

- ASSESS MISSION OPERATIONS
- ANALYZE TRANSPORTATION/OPERATIONS TRADEOFFS
- EVALUATE / SELECT RECOMMENDED SYSTEMS
- DEFINE INTEGRATED MISSION OPERATIONS AND PERFORMANCE

FAMILY STUDY FLOW

SPACE SYSTEMS

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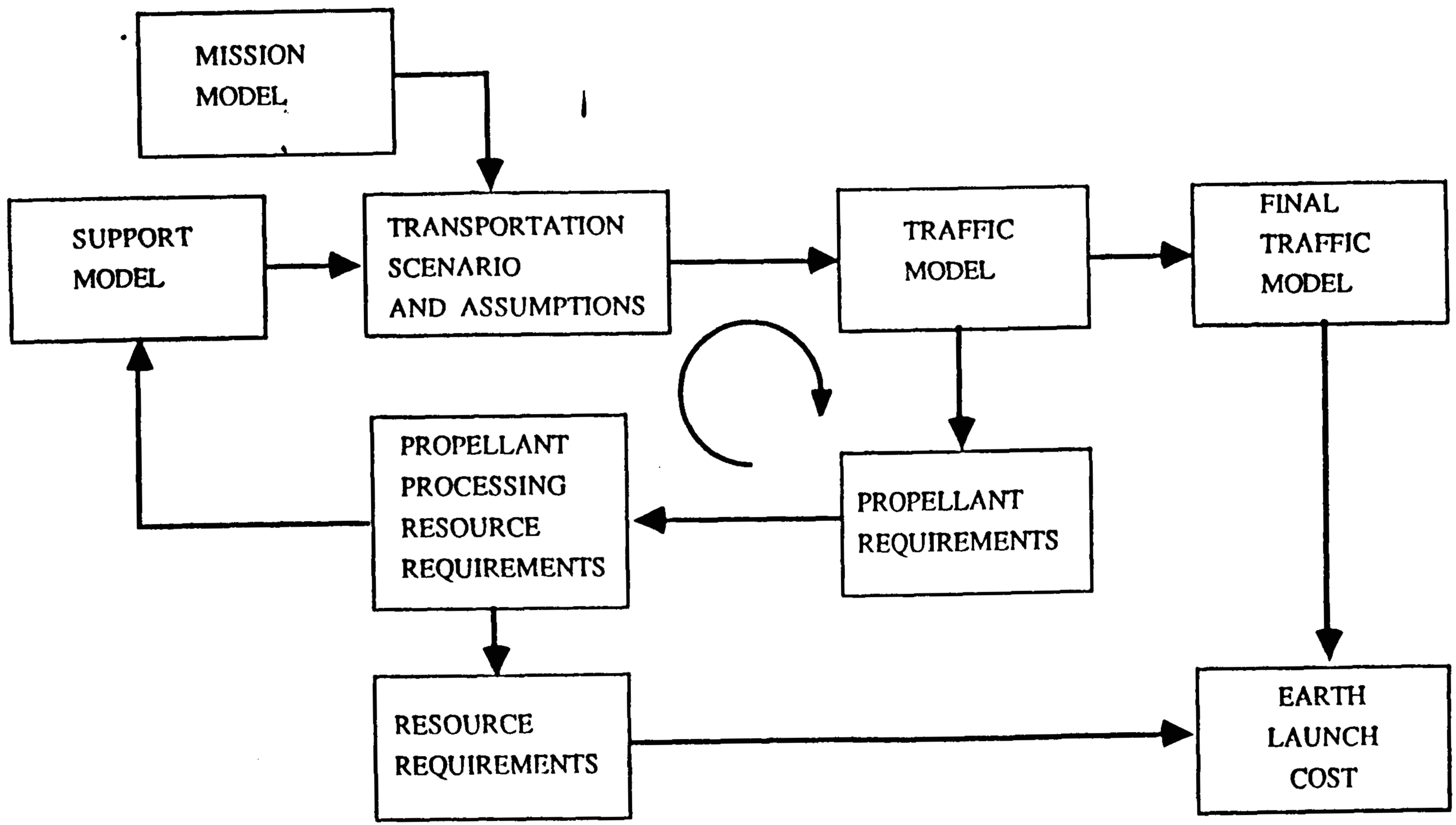

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LEGEND FOR VEHICLE SUMMARY

SPACE SYSTEMS

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(0) BASELINE, H/O OTV & LANDER (NO LUNAR PROPELLANT)

(1) H/O OTV & LANDER (LLOX AVAILABLE), DRY WEIGHT REDUCTION

(2) H/O OTV & LANDER (LLOX & LH2 AVAILABLE), DRY WEIGHT REDUCTION

(3) H/O OTV & Al/ LOX LANDER; LANDER Isp = 260, O/F = 12.1

(4) H/O OTV & LANDER; O/F = 8.7 (Isp = 421)

(5) H/O OTV & LANDER; O/F = 10.6 (Isp = 384)

(6) H/O OTV & LANDER; I_{sp}= 460

(7) H/O OTV & LANDER; I_{sp}= 490

(8) H/O OTV & LANDER; PAYLOAD = 10MT

(9) H/O OTV & LANDER; PAYLOAD = 20MT

(10) H/O OTV & LANDER; NO AEROBRAKE

(11) H/O OTV & LANDER; AEROBRAKE MASS = 18% OF REENTRY MASS

(12) H/O OTV & LANDER; AEROBRAKE MASS = 20% OF REENTRY MASS

(13) H/O OTV & LANDER; AEROBRAKE MASS = 25% OF REENTRY MASS

(14) H/O OTV & LANDER; AEROBRAKE MASS = 30% OF REENTRY MASS

(15) SiH_4 OTV & LANDER

(16) Al-H/LOX OTV & LANDER

LSB MISSION MODEL

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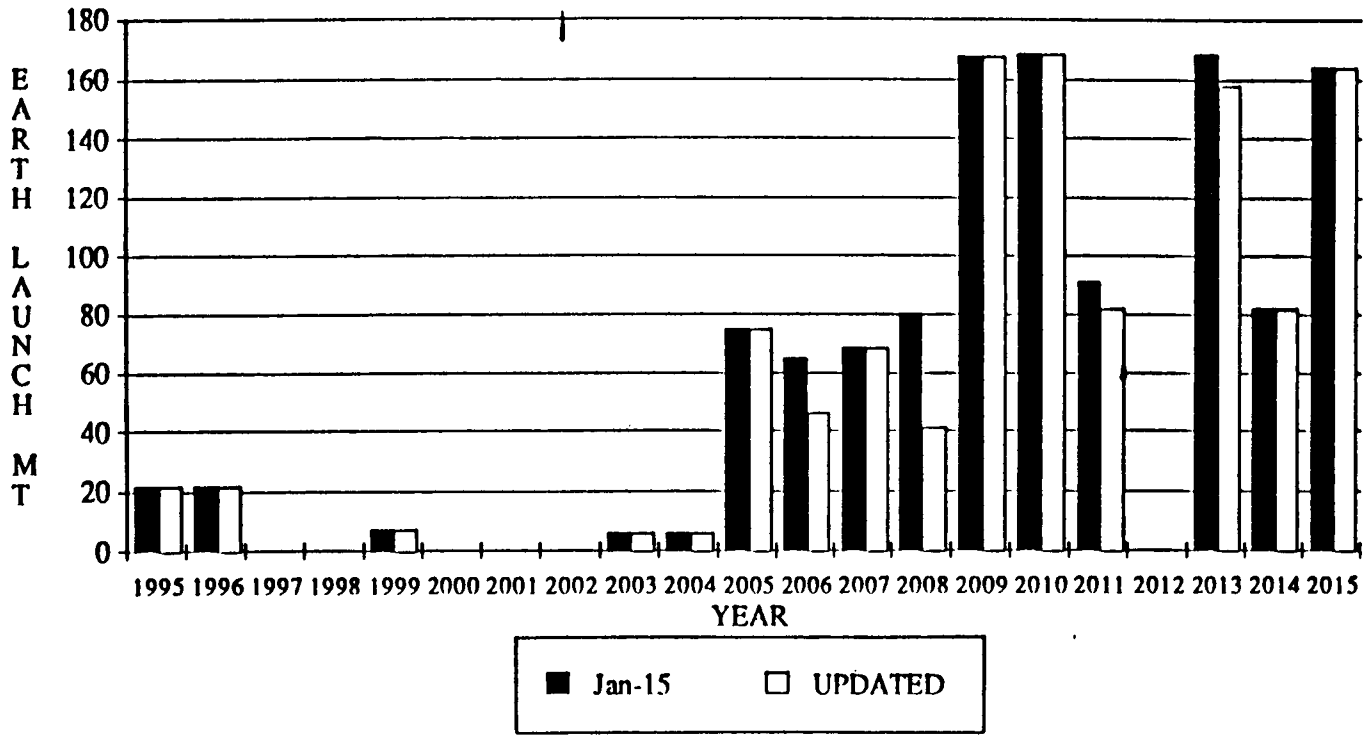
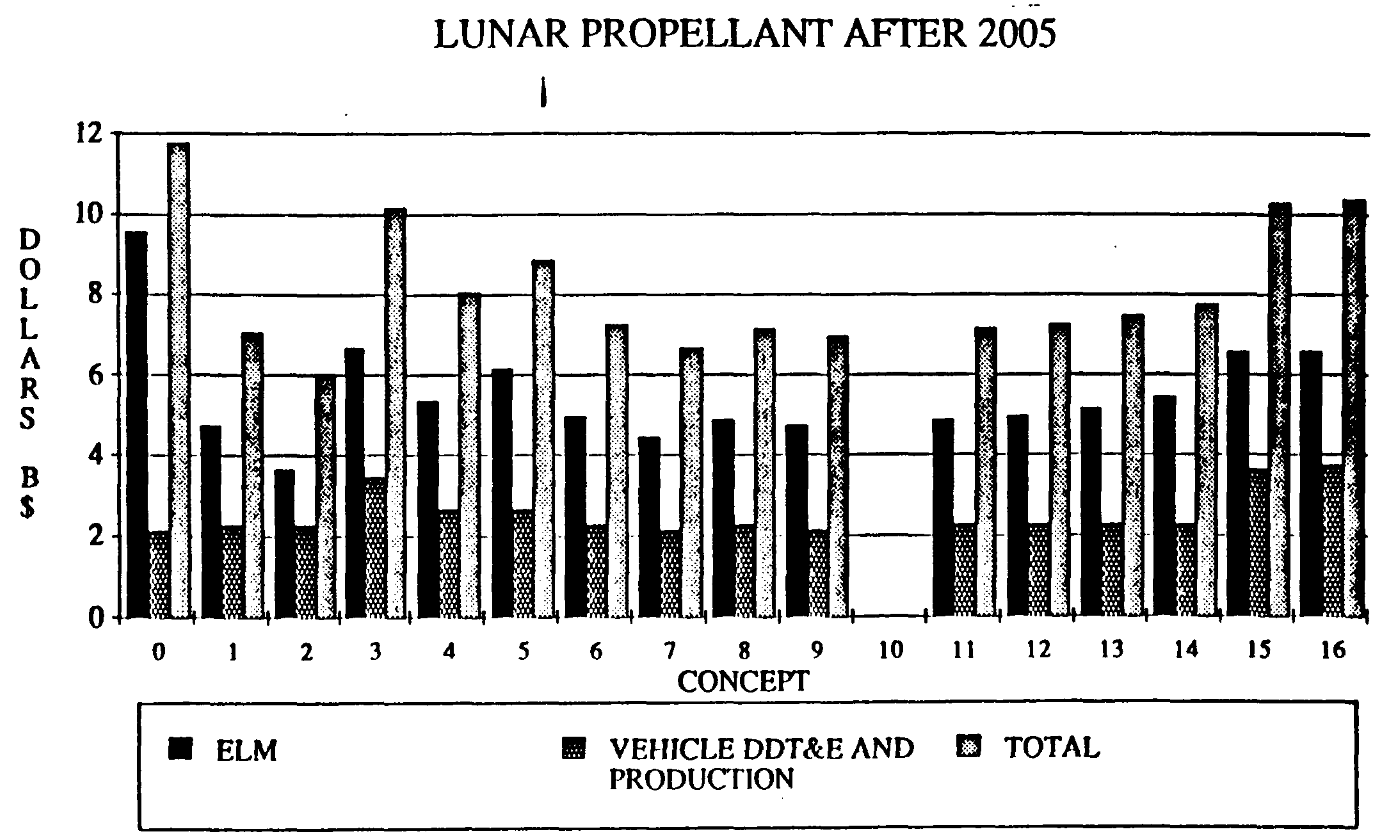


TOTAL TRANSPORTATION COST

SPACE SYSTEMS



TOTAL TRANSPORTATION COST

SPACE SYSTEMS

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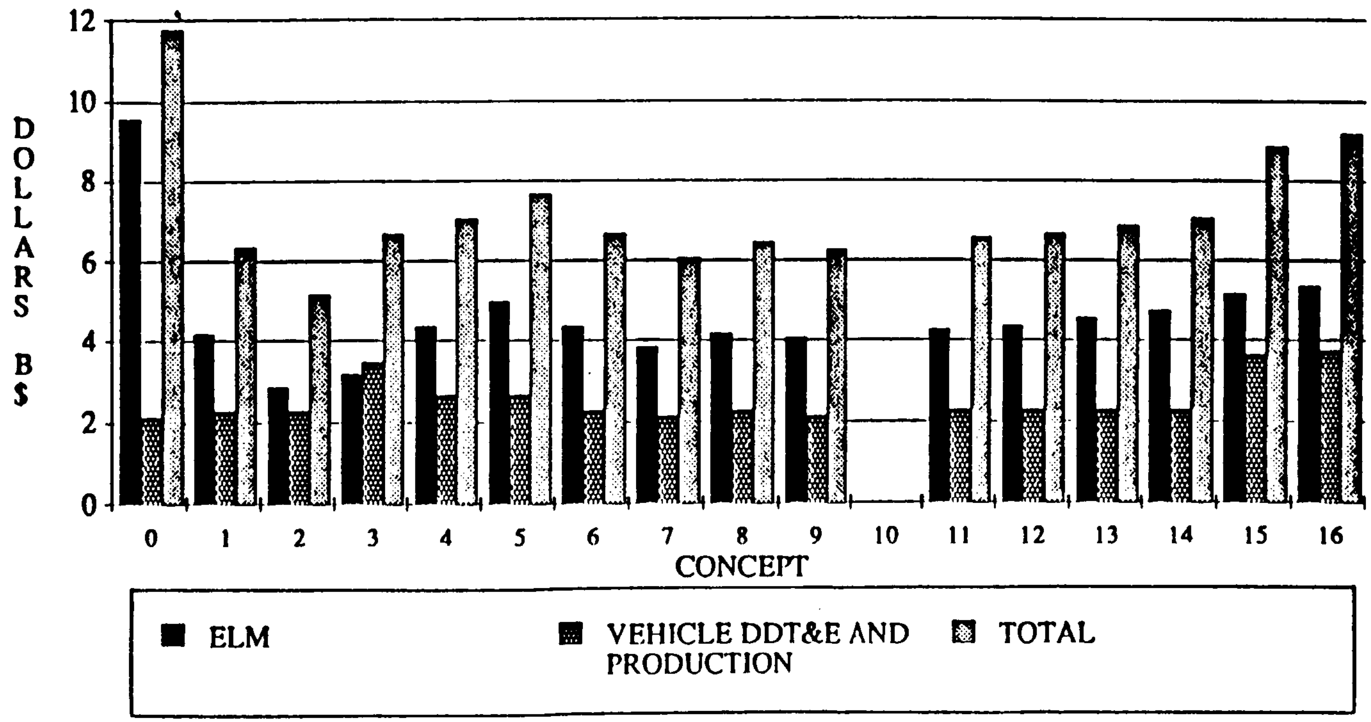
LUNAR PROPELLANT IN 1995



VEHICLE FAMILY RESULTS SUMMARY
(Based on entire Mission Model)

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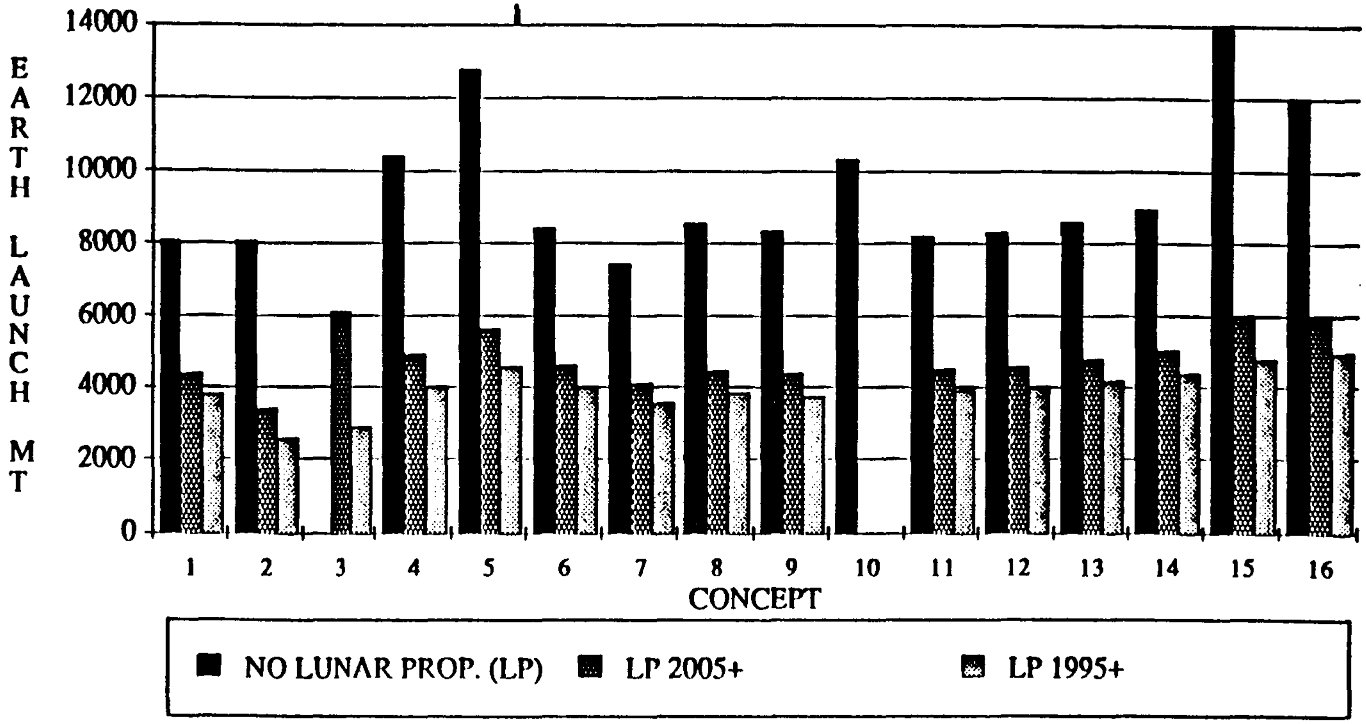
SPACE SYSTEMS



EARTH LAUNCH MASS RESULTS

SPACE SYSTEMS

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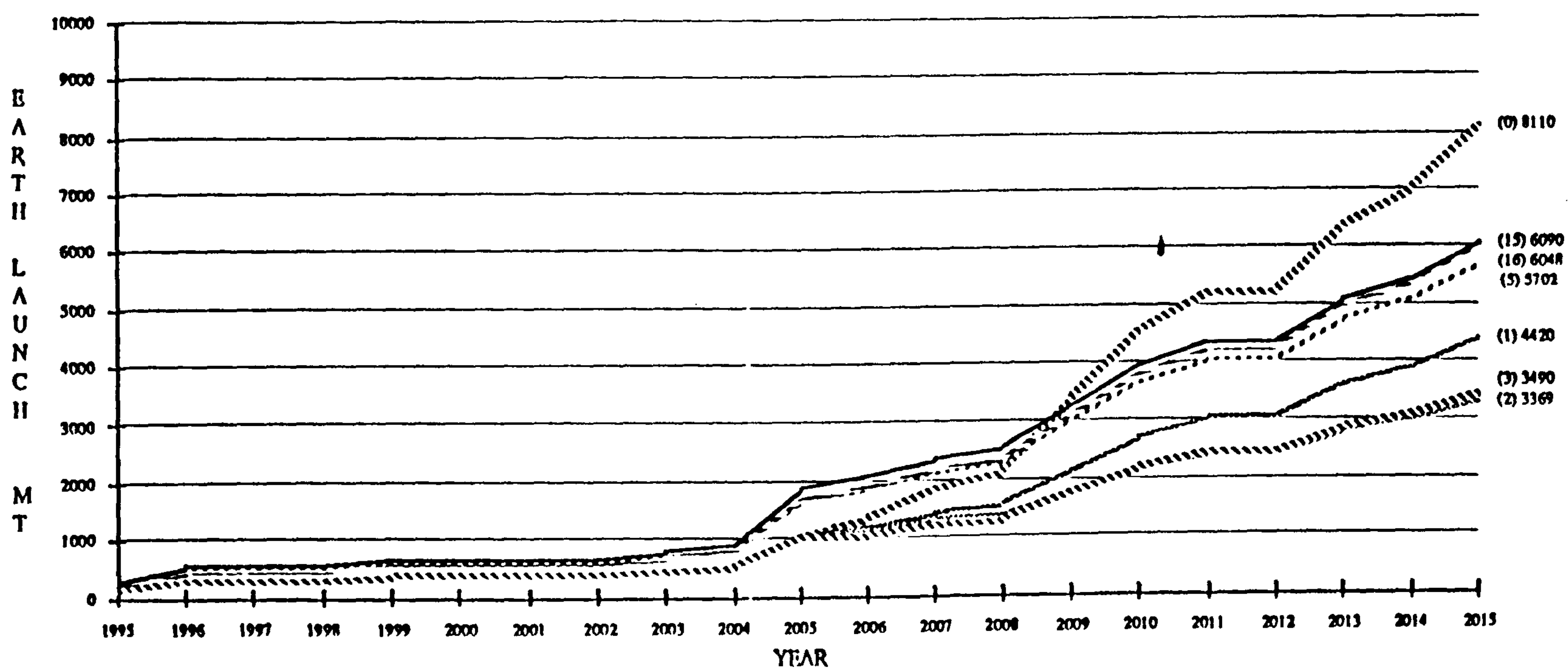
CONCEPT COMPARISONS
LUNAR PROPELLANT AFTER 2005



EARTH LAUNCH MASS RESULTS

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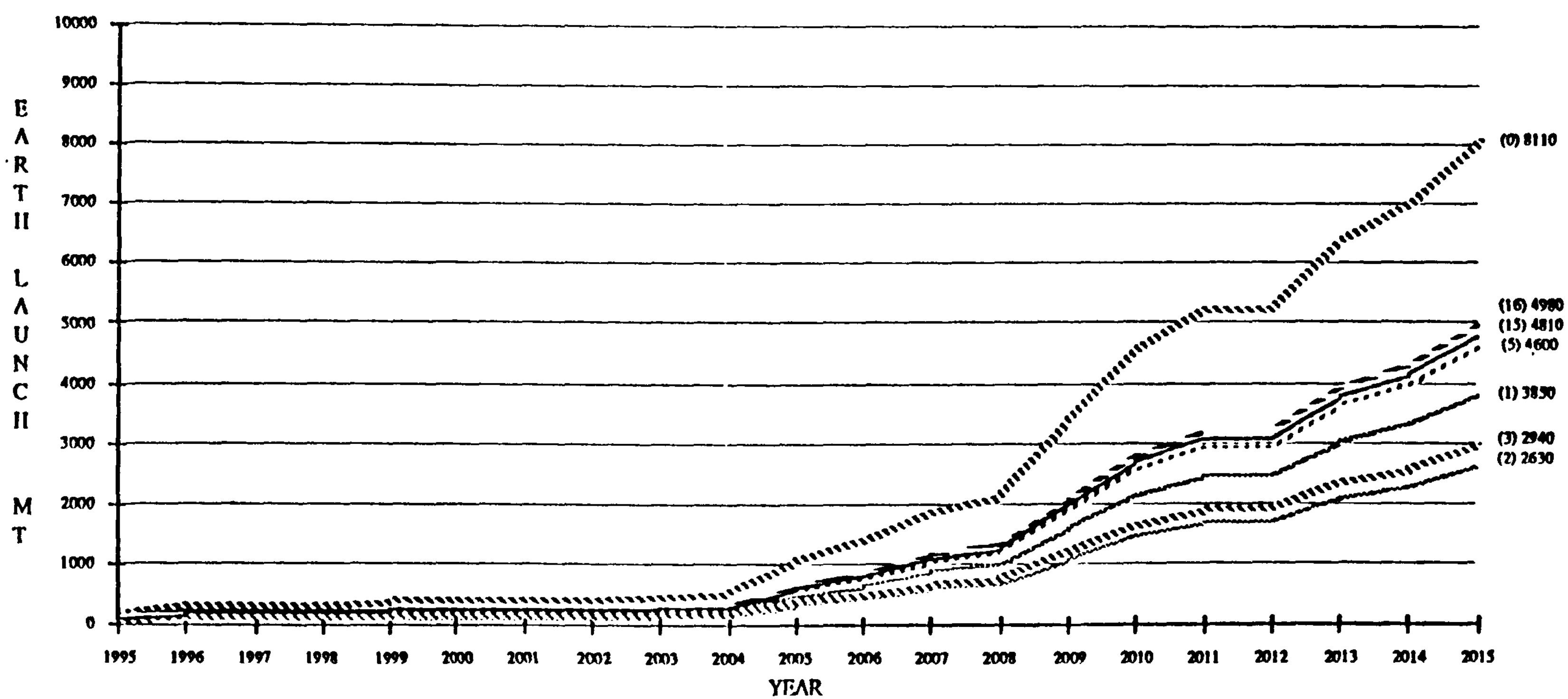
SPACE SYSTEMS

CONCEPT COMPARISONS
LUNAR PROPELLANT IN 1995



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SUMMARY OF RESULTS

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- LLOX/LH2 IS BEST PROPULSION SYSTEM SCENARIO
- LLOX REDUCES EARTH LAUNCH MASS BY MORE THAN 50% IN THE H/O SYSTEM
- LUNAR HYDROGEN REDUCES ELM 30% OVER LLOX H/O FAMILY
- AEROBRAKE SAVES OVER 60% ELM IN H/O FAMILY
- SHUTTLE SCAVENGING COULD POTENTIALLY SAVE 2000 MT OVER THE 20 - YEAR MISSION MODEL PERIOD - ABOUT 1/2 OF THE H/O SYSTEM REQUIREMENT
- A 20 SECOND INCREASE IN H/O Isp REDUCES EARTH LAUNCH MASS (ELM) BY 7%

SPACE SYSTEMS

SUMMARY OF RESULTS

(CONTINUED)

- SENSITIVITY OF H/O O/F RATIO ON EARTH LAUNCH MASS IS REDUCED WITH LLOX AVAILABILITY
  - WITHOUT LLOX AN INCREASE IN O/F RATIO FROM 5.5 TO 8.7 INCREASES ELM BY 29%
  - WITH LLOX AN INCREASE IN O/F RATIO FROM 5.5 TO 8.7 INCREASES ELM BY 5%
  - OPTIMUM O/F NOT YET IDENTIFIED (BETWEEN 6 - 8 O/F RATIOS)
- ELM IS NOT EXTREMELY SENSITIVE TO OTV PAYLOAD CAPABILITY
- INCREASING SPECIFIC AEROBRAKE MASS FROM 15% TO 30% OF REENTRY MASS INCREASES ELM BY 103% FOR NON-LLOX SCENARIOS AND ONLY 14% WITH LLOX
- PROPELLANT PRODUCTION CONSUMABLE REQUIREMENTS GREATLY AFFECT THE ELM
  - INCREASING CONSUMABLE REQUIREMENTS BY 12% FOR Al AND LOX INCREASES ELM BY 63%
  - INCREASING CONSUMABLE REQUIREMENTS BY 12% FOR LOX INCREASES ELM BY 37%
- CONSUMABLE REQUIREMENTS AFFECT COST GREATER THAN LAUNCH MASS BECAUSE OF INCREASED VEHICLE REQUIREMENTS

TASK 4: OBJECTIVE
EVALUATE TECHNOLOGY REQUIREMENTS

SPACE SYSTEMS

TO IDENTIFY MAJOR TECHNOLOGY ISSUES THROUGHOUT THE MISSION, VEHICLE, AND PROPULSION SYSTEMS, AND TO RECOMMEND RESEARCH NEEDS

- DEFINE PROPELLANT SUPPLY TECHNOLOGY NEEDS
- DEFINE PROPULSION SYSTEM TECHNOLOGY NEEDS
- DEFINE VEHICLE SYSTEMS TECHNOLOGY NEEDS
- DEVELOP INTEGRATED TECHNOLOGY PLAN

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KEY PROPULSION SYSTEM TECHNOLOGIES (CHEMICAL)

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ENABLING	ENHANCING
ALUMINUM FEED IN Al/LO2 ENGINE	SiH4 COMBUSTION
Al/LO2 COOLING	Al/LO2 COMBUSTION
Al-LH2/LO2 COOLING	Al-LH2/LO2 COMBUSTION
HYBRID PROPELLANT CHARACTERIZATION	COMBUSTION / EXHAUST COMPOSITION
SPACE BASING CAPABILITIES	HIGH MR LH2/LO2 ENGINE
AUTOMATION / ROBOTICS	COOLING OF HIGH MR ENGINES
ENGINE HEALTH MONITORING	HIGH EXPANSION RATIO NOZZLE
	HIGH SPEED TURBOMACHINERY

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KEY PROPELLANT SUPPLY TECHNOLOGIES

SPACE SYSTEMS

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ENABLING	ENHANCING
PLASMA PROCESSING	SYNERGISTIC PROCESSING TECHNIQUES
ELECTRODE LIFETIME AND DURABILITY IMPROVEMENT	HIGH TEMPERATURE MATERIALS PROCESSING
CONSUMABLE RECYCLING IN PROCESSING OPERATION	SEPARATION OF METAL OXIDES FROM SILICON OXIDES
SILICATE SEPARATION	SEPARATION OF ILMENITE
HYDROGEN EXTRACTION	RAW MATERIAL COLLECTION, TRANSPORTATION
CRYOGENIC STORAGE	RESOURCE AND MINING SITE LOCATION

KEY VEHICLE TECHNOLOGIES

ENABLING	ENHANCING
AEROBRAKING / CAPTURE	LIGHT WEIGHT STRUCTURES, TANKAGE
PROPELLANT TRANSFER / HANDLING	EXTENDED LIFE
SPACE BASING / SERVICING / REFURBISHMENT	AUTOMATION / ROBOTICS
	HEALTH MONITORING

TECHNOLOGY STUDIES AND TRADEOFF ANALYSES PLAN

SPACE SYSTEMS

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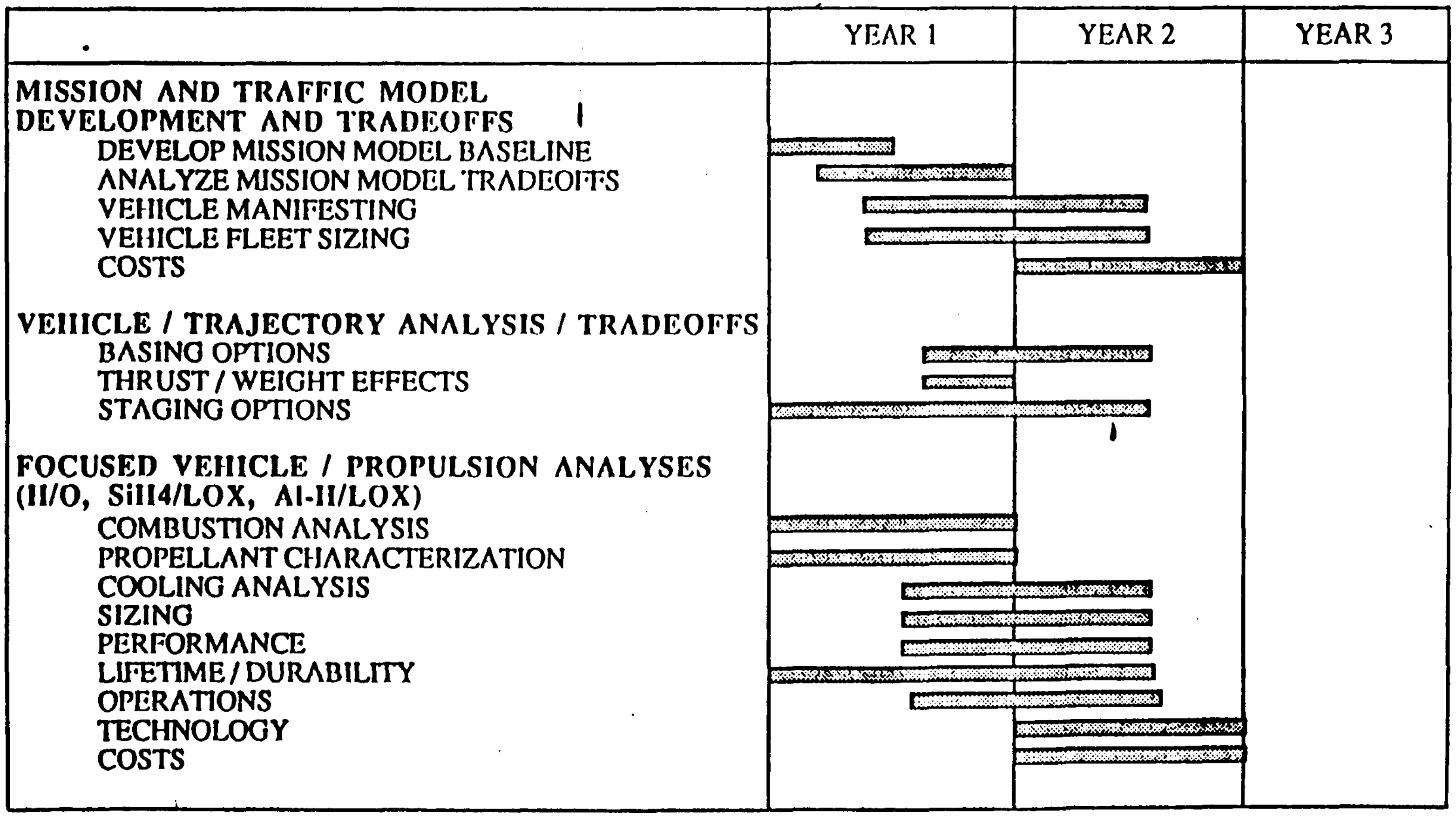
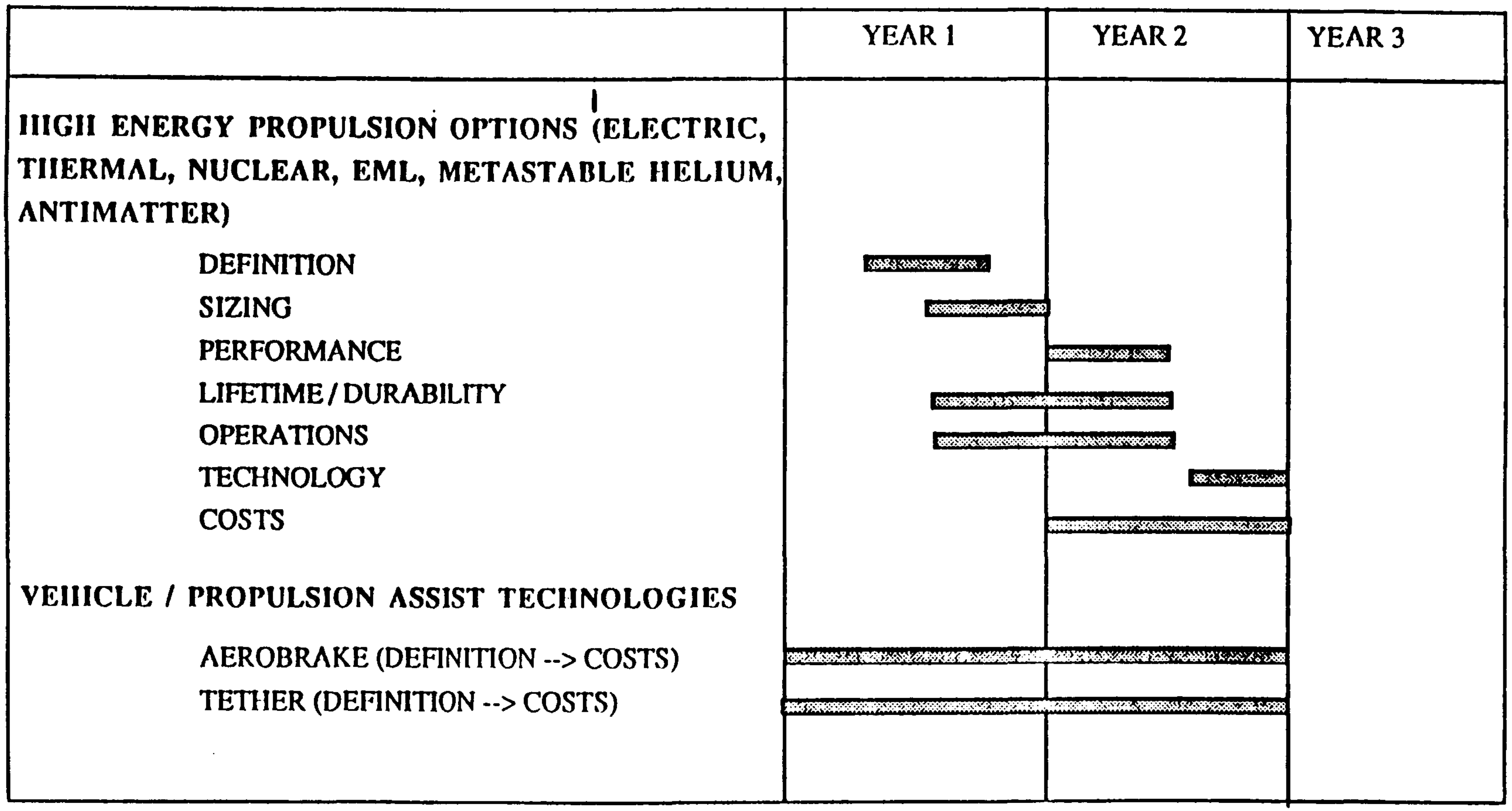


TECHNOLOGY STUDIES AND TRADEOFF ANALYSES PLAN

(CONTINUED)

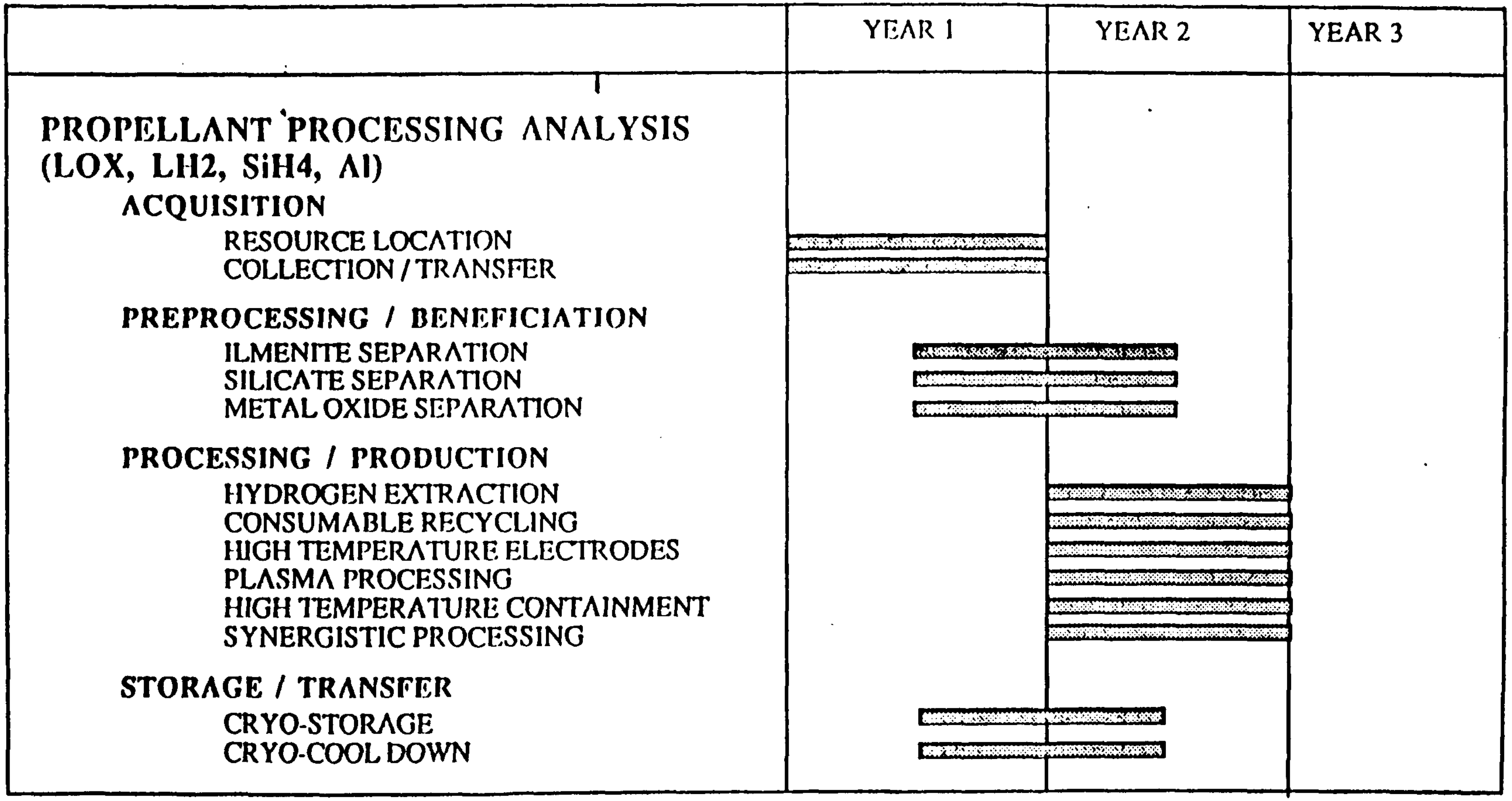


TECHNOLOGY STUDIES AND TRADEOFF ANALYSES PLAN

(CONTINUED)

ACA — Astronautics Corporation of America - Technology Center — ATC

SPACE SYSTEMS



TECHNOLOGY EXPERIMENT PLAN

SPACE SYSTEMS

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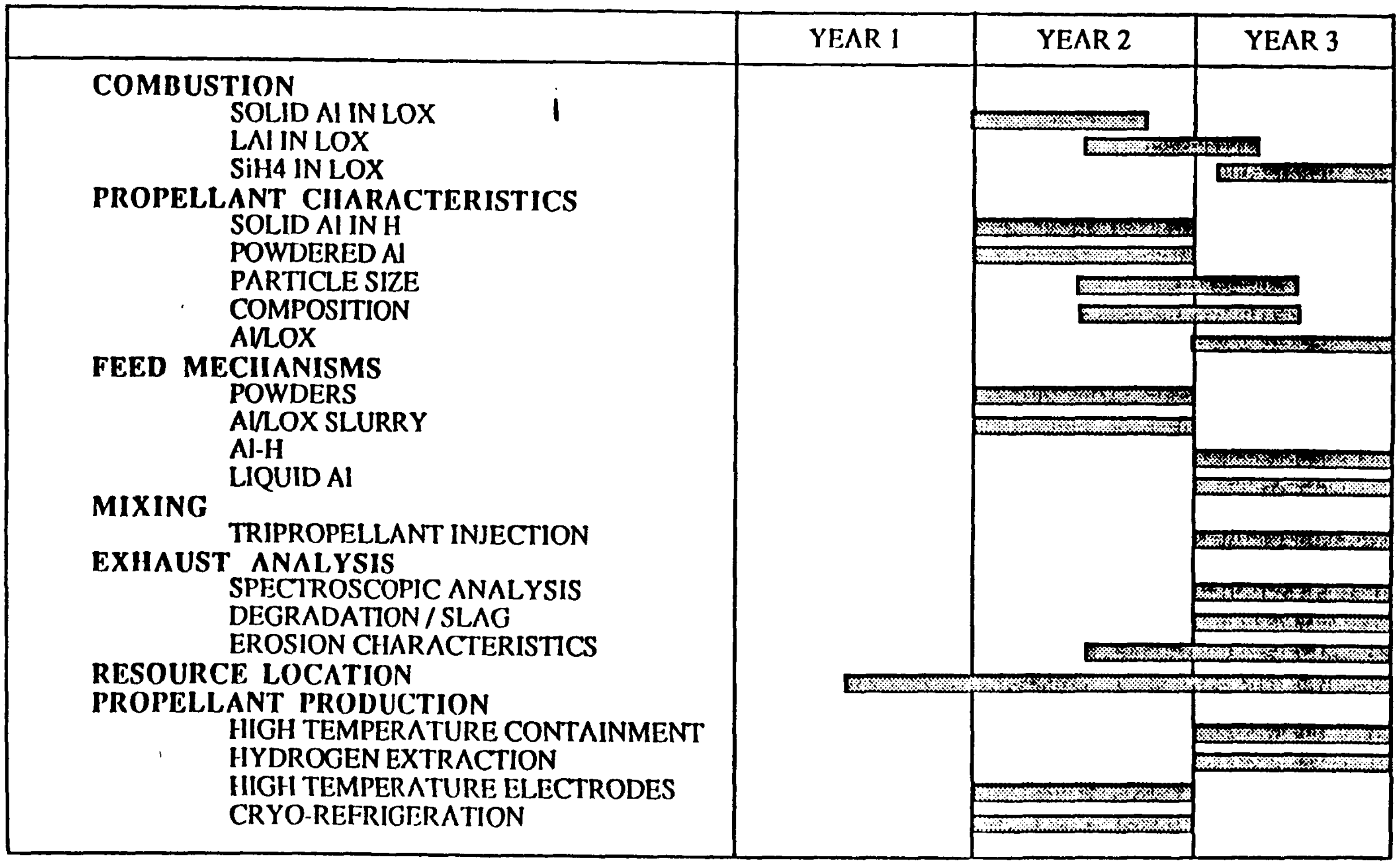


CONCLUSIONS

- LLOX REDUCES EARTH LAUNCH MASS AND COST BY MORE THAN 50%
- AEROBRAKE PRODUCES APPROXIMATELY 60% REDUCTION IN EARTH LAUNCH MASS
- ELM SIGNIFICANTLY AFFECTED BY SPECIFIC AEROBRAKE MASS WITHOUT LUNAR PROPELLANT AVAILABILITY BUT AFFECTED ONLY MARGINALLY WITH LUNAR PROPELLANT AVAILABLE
- LLOX AVAILABILITY REDUCES BURDEN ON ENGINE PERFORMANCE
  - ELM SENSITIVITY TO H/O Isp IN RANGE 470 TO 490 IS LOW
  - O/F RATIOS AT OR HIGHER THAN 8.7 NOT A BENEFIT
  - O/F RATIO BETWEEN 5.5 AND 8 MAY SHOW BENEFIT OF REDUCED FUEL REQUIREMENT
- LUNAR PROPULSION SYSTEM PERFORMANCE VERY SENSITIVE TO COMBINED FACTORS OF Isp AND PERCENTAGE OF LUNAR-DERIVED PROPELLANT
- PRELIMINARY ESTIMATES OF PROPELLANT PROCESSING REQUIREMENTS MAY BE OPTIMISTIC AND AFFECT TOTAL ELM A GREAT DEAL
- LUNAR HYDROGEN VERY VALUABLE FUEL (SAVES 30% ELM IN H/O SYSTEM OVER LLOX SCENARIO)
- LUNAR FUELS RANK: H, Al, SiH_4, Al-H SCENARIOS IN DECREASING ORDER OF BENEFIT TO LUNAR TRANSPORTATION

CONCLUSIONS

(CONTINUED)

- O/F RATIO OF 8.7 INCREASES TOTAL EARTH LAUNCH MASS BY 29% WITHOUT LLOX AND 5% WITH LLOX: O/F RATIO HIGHER THAN 6 AND LOWER THAN 8.7 MAY BE OPTIMAL

- LARGE CARGO VEHICLE SUCH AS SEPS NEPS IS POTENTIALLY VALUABLE IN REDUCING EARTH LAUNCH MASS BY TRANSFERING PROPELLANT MISSION MASS TO AND FROM THE MOON

- PROPELLANT TANKS REPRESENT THE LARGEST SINGLE MASS OF THE VEHICLE AND THUS AFFECT ELM A GREAT DEAL

- SHUTTLE SCAVENGING COULD POTENTIALLY PROVIDE A SIGNIFICANT AMOUNT (20 - 50%) OF THE LUNAR HYDROGEN AND OXYGEN REQUIREMENTS OVER THE CURRENT MISSION MODEL

- WIDE VARIATION IN MISSION FLIGHT RATES SEEMS UNREALISTIC

RECOMMENDATIONS

- UPDATE AND DEVELOP MISSION MODEL FOR TRADE-OFF CONSIDERATIONS
- ANALYZE EFFECTS OF LLOX RETURN TO LEO USING CHEMICAL AND OTHER MEANS
- DETERMINE OPTIMUM O/F RATIO FOR ALL RECOMMENDED PROPULSION SYSTEMS (O/H MAY OCCUR BETWEEN 6 AND 8)
- ANALYZE EQUIPMENT AND CONSUMABLES FOR Al, Si, O, AND H PRODUCTION IN GREATER DETAIL TO INCLUDE CREW NEEDS, LIFETIME ESTIMATES, MINING REQUIREMENTS AND CONSUMABLE RECYCLING
- INCLUDE BASING REQUIREMENTS INTO SUPPORT MODELING
- ASSESS ENGINE PERFORMANCE VERSES LIFETIME TRADEOFF
- ANALYZE AND DETERMINE POTENTIAL BENEFITS OF A LARGE CARGO VEHICLES AND HIGH-ENERGY PROPULSION SYSTEM
- ANALYZE VEHICLE AND PROPULSION SYSTEM SIZES AND WEIGHTS
- COMPLETE (IN MORE DETAIL) AND ACTIVATE TECHNOLOGY PLAN
- INVESTIGATE AND DEFINE MORE CLEARLY PERFORMANCE AND LIFETIME OF SILICON- AND ALUMINUM- FUELED ENGINES
- ANALYZE EFFECTS OF VEHICLE SIZING AND ALTERNATIVE MISSION TRAJECTORIES

A LUNAR ELECTROMAGNETIC LAUNCHER

by

Curt Bilby, Hubert Davis, and Stewart Nozette

Center for Space Research
The University of Texas at Austin

and

Mircea Driga and Randy Kamm

Center for Electromechanics
The University of Texas at Austin

LUNAR BASES & SPACE ACTIVITIES
IN THE 21ST CENTURY

April 5 - 7, 1988
Westin Galleria Hotel
Houston, Texas

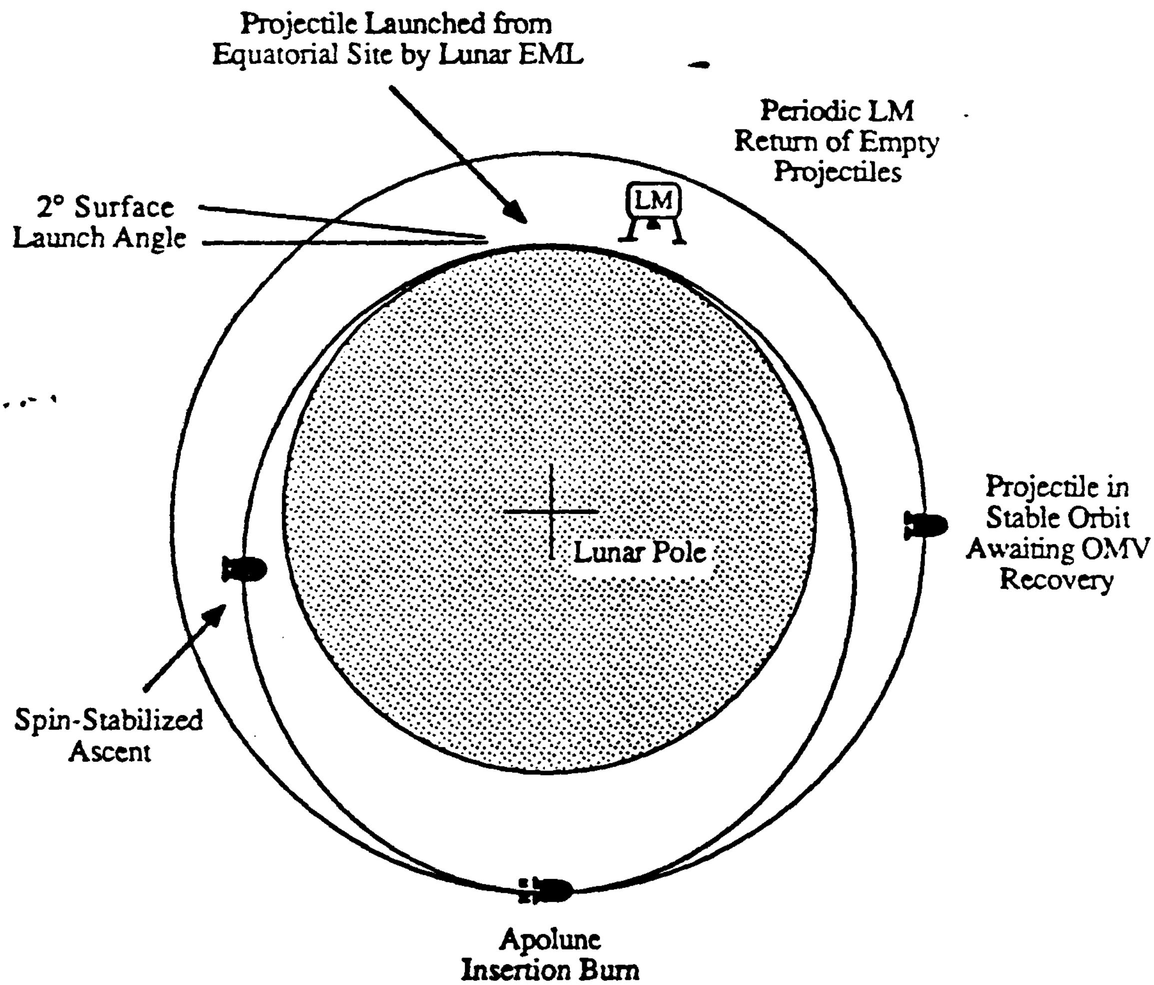
PROBLEM STATEMENT

Identify the Design, System Integration, and Logistics Requirements Associated with Utilizing Electromagnetic Launch Technology on the Lunar Surface and Determine the Benefit of such a System

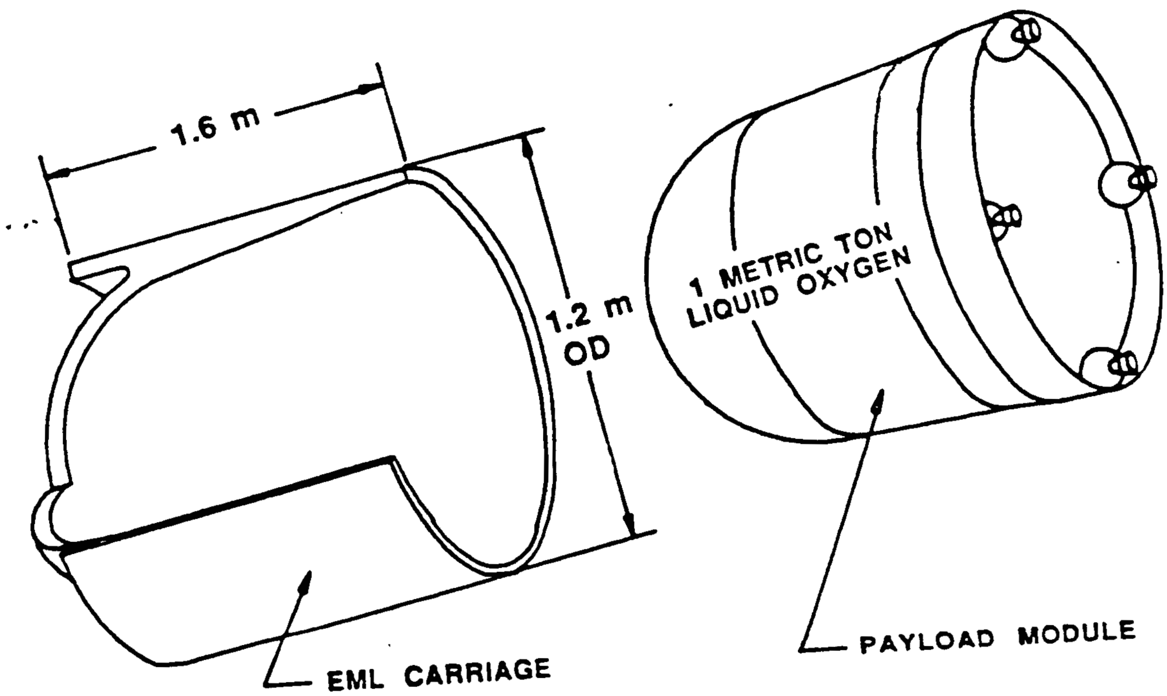
ASSUMPTIONS

- A Permanent Manned Lunar Base is Operational
- The Requisite Number of Space Transportation Vehicles (Launch Vehicles, OTVs, and LMs) are in Place and Operational
- A Lunar-Derived Liquid Oxygen Facility is Operational on the Moon
- Enough Liquid Oxygen is Being Produced on the Lunar Surface to Allow Export to a Low Earth Orbit Market

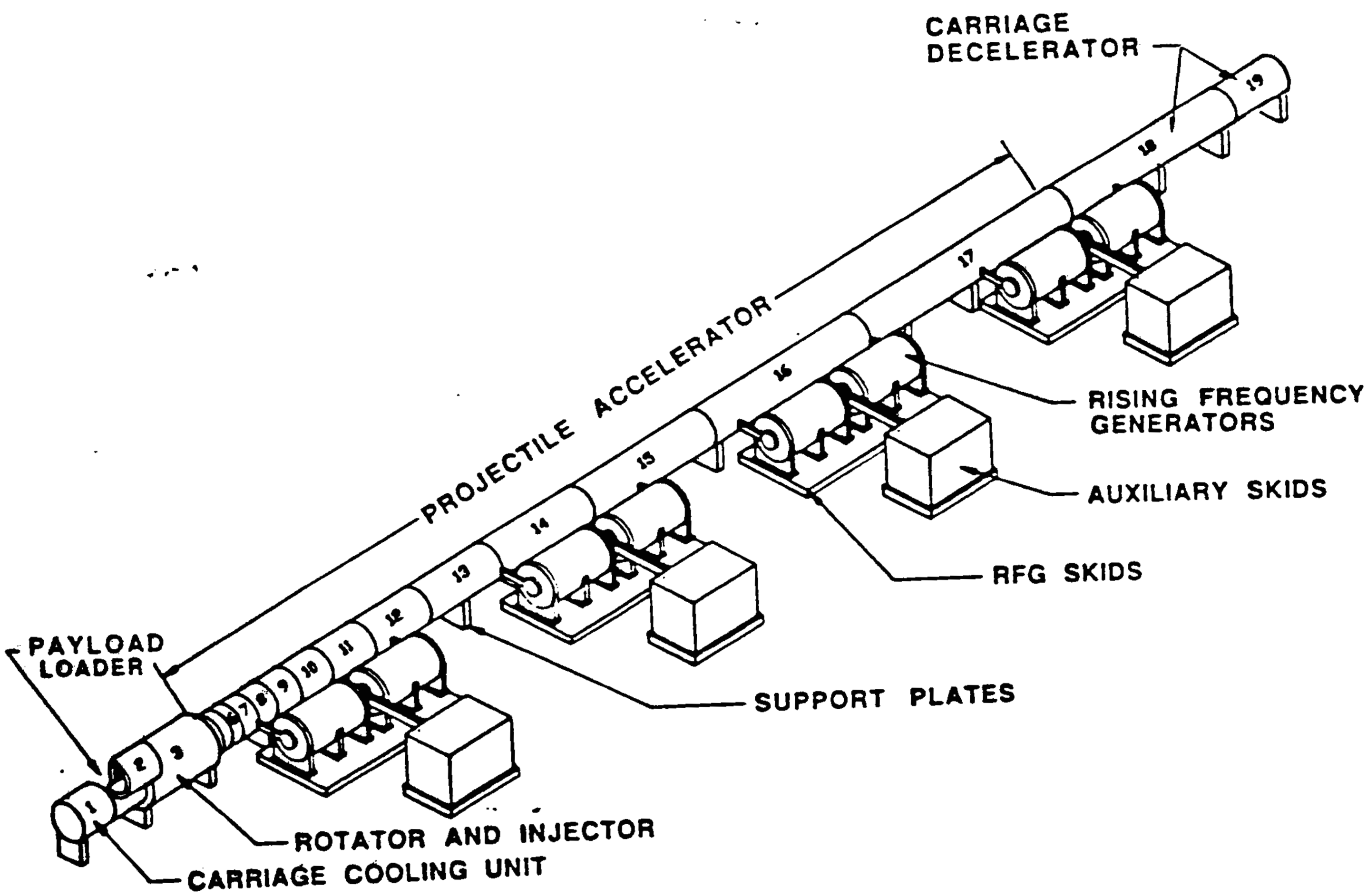
LUNAR PROXIMITY SCENARIO



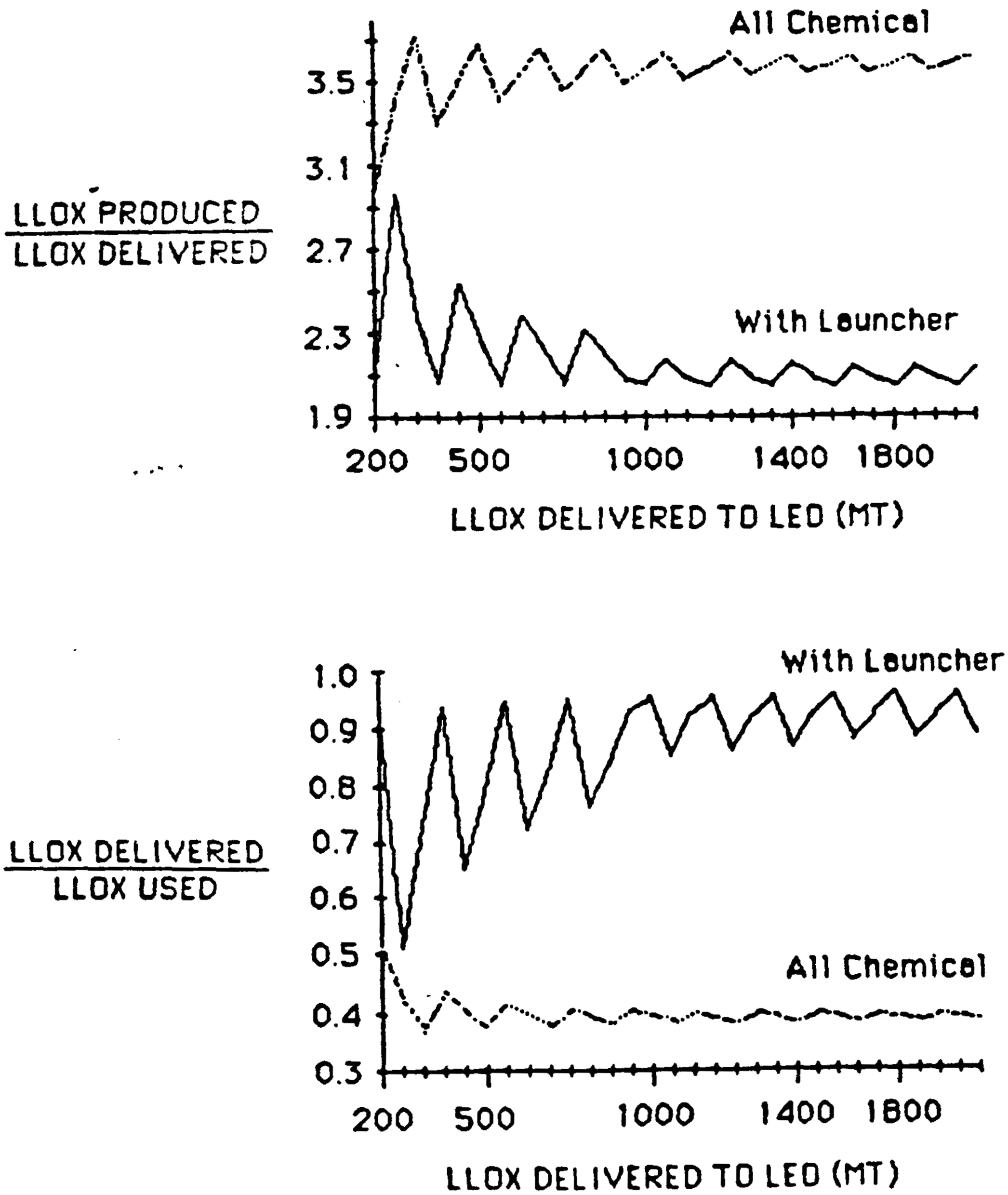
PROJECTILE DESIGN



LUNAR LAUNCHER CONCEPT

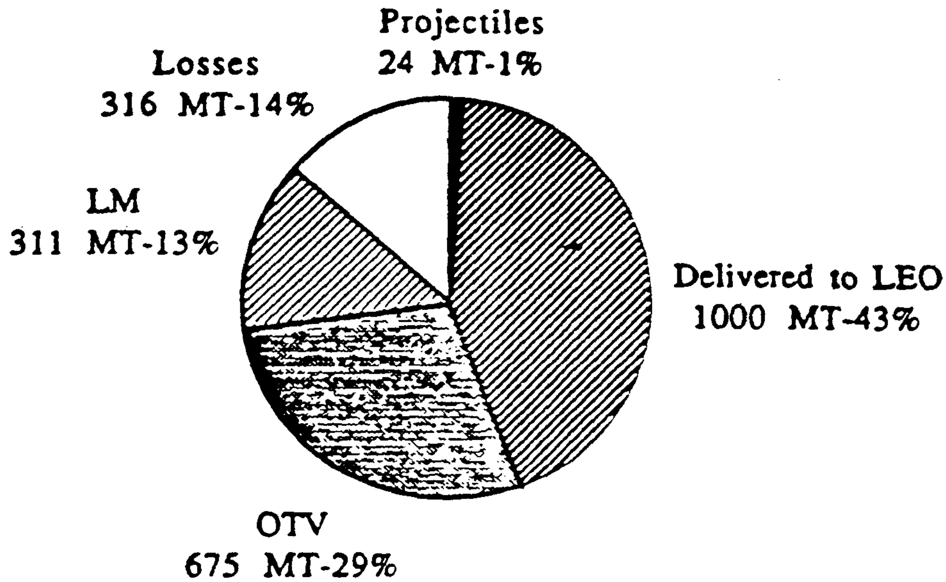


LLOX DELIVERY EFFICIENCY



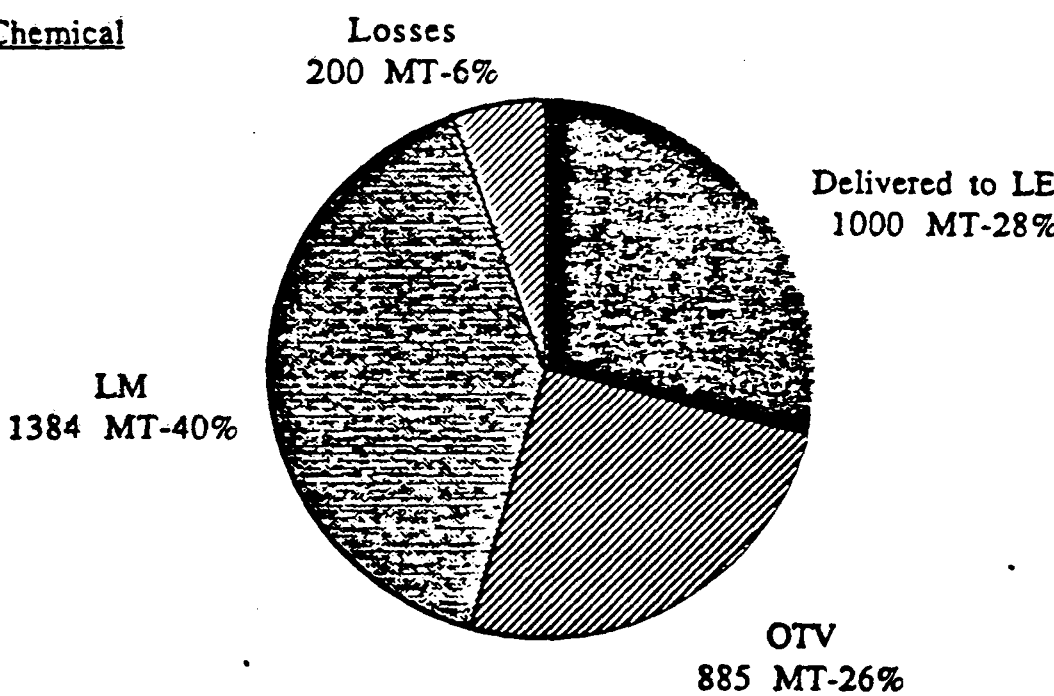
UTILIZATION OF LLOX

with EML



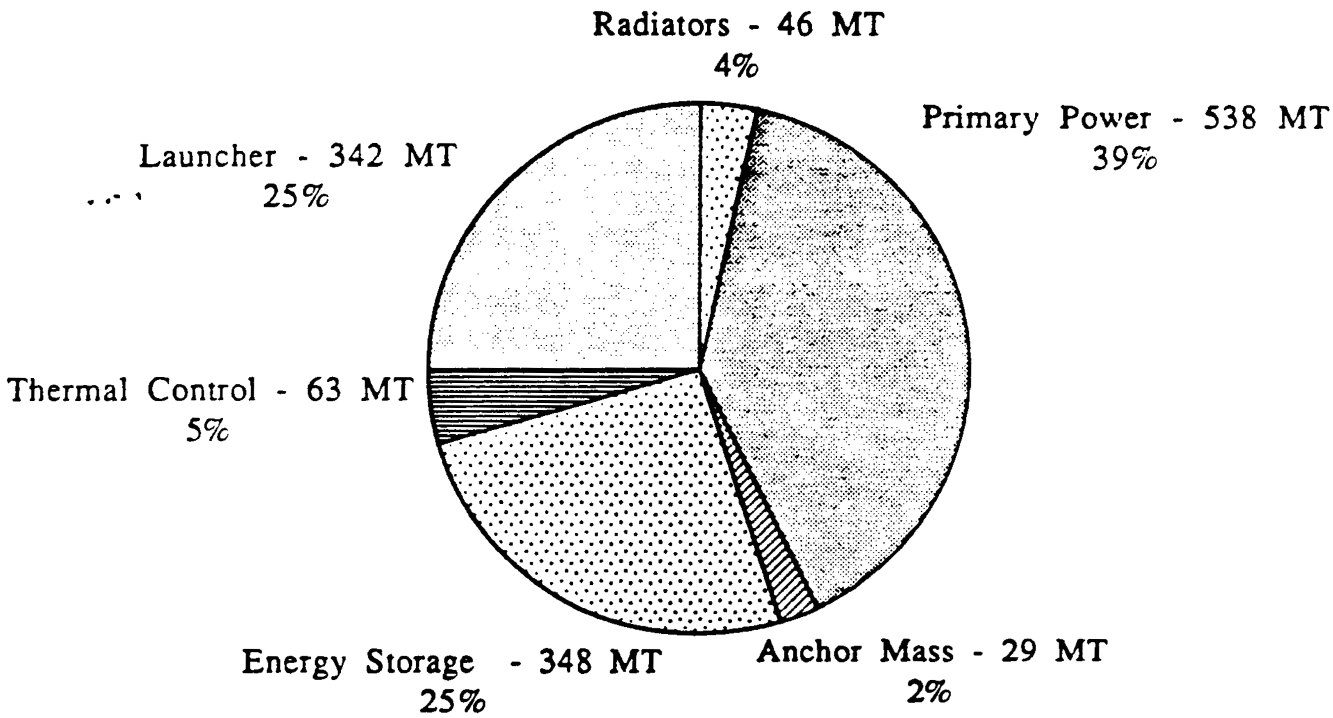
Total LLOX Produced on Lunar Surface - 2326 MT

All Chemical



Total LLOX Produced on Lunar Surface - 3469 MT

EMPLACEMENT MASS DISTRIBUTION FOR LAUNCHER AND SUPPORTING SYSTEMS



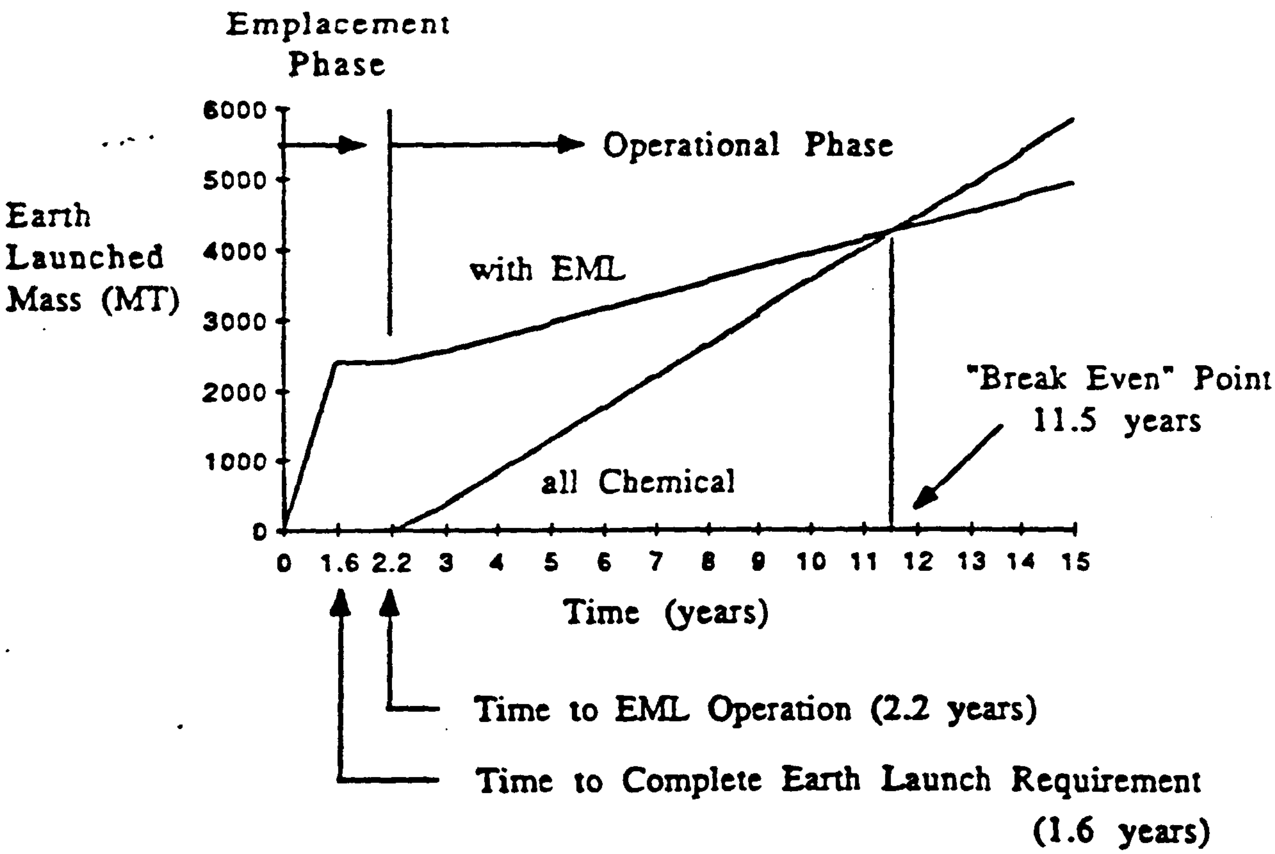
SUPPORT REQUIREMENTS

Item	Mass (MT)	Destination
O T V fuel (LH2)	371	LEO
L L O docking facility	5	LLO
Lunar OMVs (3)	14	LLO
1 mo. supply O M V fuel	4	LLO
O T V and L M fuel (LH2)	513	LLO
Projectiles (191)	48	Lunar Surface
Launcher and systems	1366	Lunar Surface
Operation facility	52	Lunar Surface
Construction equipment	17	Lunar Surface

TRANSPORTATION REQUIREMENTS FOR SYSTEM EMPLACEMENT

Vehicle	Flight Origination	Flight Destination	Number Required
HLLV	Earth	LEO	20
OTV	LEO	LLO	43
LM	LLO	Lunar Surface	38
LM	Lunar Surface	LLO	76
OTV	LLO	LEO	11

TIME TO BREAK EVEN



ADVANCED PROPULSION FOR LOW EARTH ORBIT - MOON TRANSPORTATION

Mark Henley

GENERAL DYNAMICS

Space Systems Division

AIM OF STUDY:

To identify and evaluate advanced propulsion methods capable of yielding substantial benefits over conventional chemical propulsion for the transportation of materials between LEO and the Moon.

TASKS

1. Select Configurations
   a) Reference Case & Std. Mission
   b) Tethers
   c) Others

2. Select Performance Criteria
   a) Quantitative
   b) Qualitative

3. Describe ("model") the configurations

4. "Model" the performance

5. Evaluate the configurations

ADVANCED CONFIGURATIONS

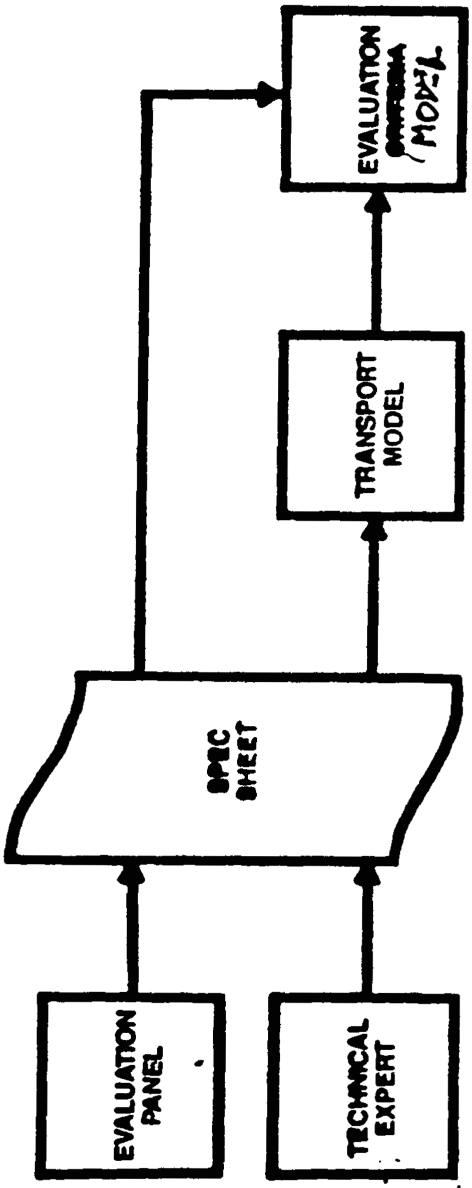
A. TETHERS

1. Hanging tether in LLO
2. Spinning tether in LLO or LEO
3. Spinning tether in EEO -- throw and catch

3a. Spinning tether in EEO -- throw only

B. OTHER

1. Laser thermal propulsion
2. Ion engine propulsion
3. Mass driver launch from Moon



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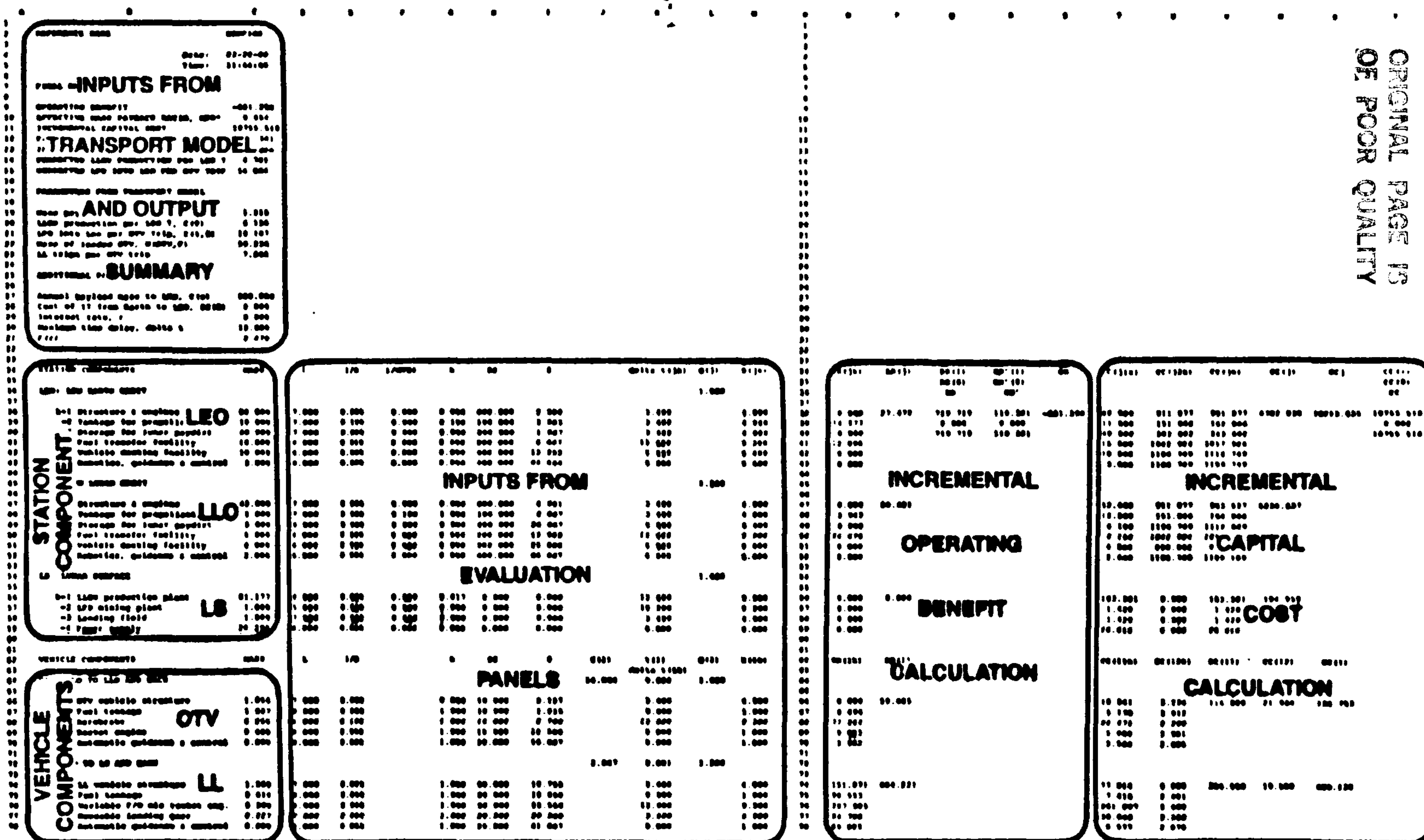
INPUTS FROM
TRANSPORT MODEL
AND OUTPUT
SUMMARY
STATION
COMPONENT
LEO
LLO
LS
VEHICLE
COMPONENTS
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LL
INPUTS FROM
EVALUATION
PANELS
INCREMENTAL
OPERATING
BENEFIT
CALCULATION
INCREMENTAL
CAPITAL
COST
CALCULATION

NEXT STEPS

1. Evaluate additional "pure" configurations (e. g., solar thermal propulsion).

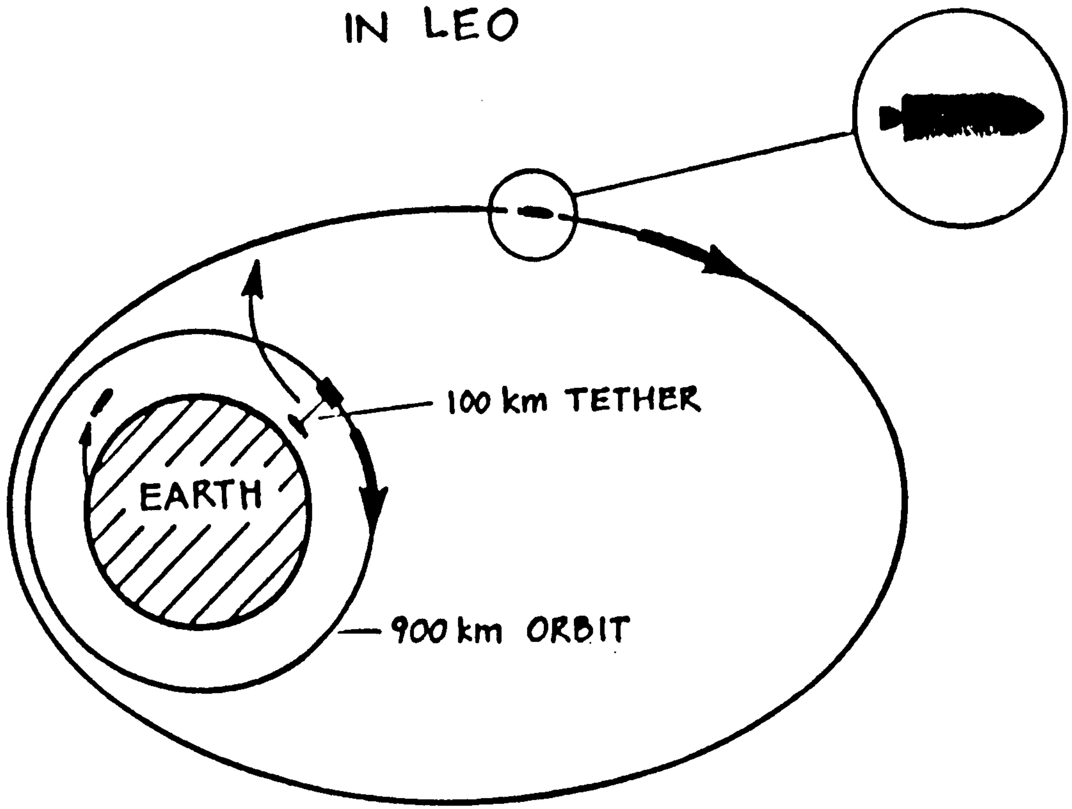
2. Develop and define engineering concepts for all promising "pure" configurations.

3. Determine orbital windows, schedule and vehicle fleet requirements for above.

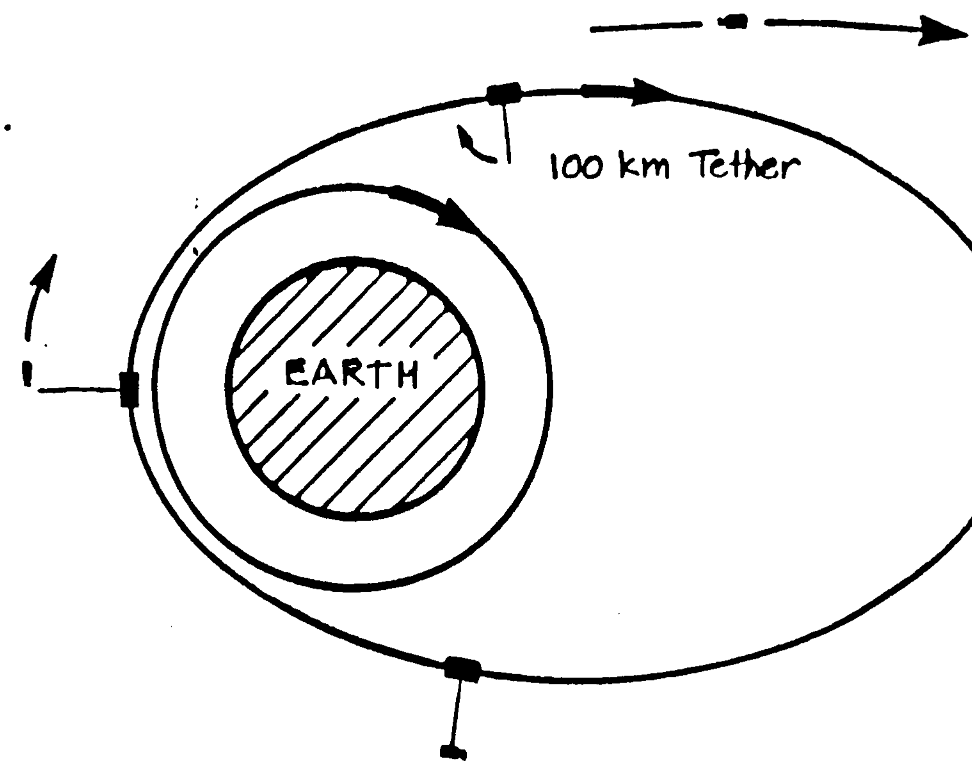
4. Indentify and evaluate most promising "hybrid" configurations, and provide engineering concept definitions and orbital characteristics for them.

5. Identify R & D priorities and needed resources for most promising advanced configurations.

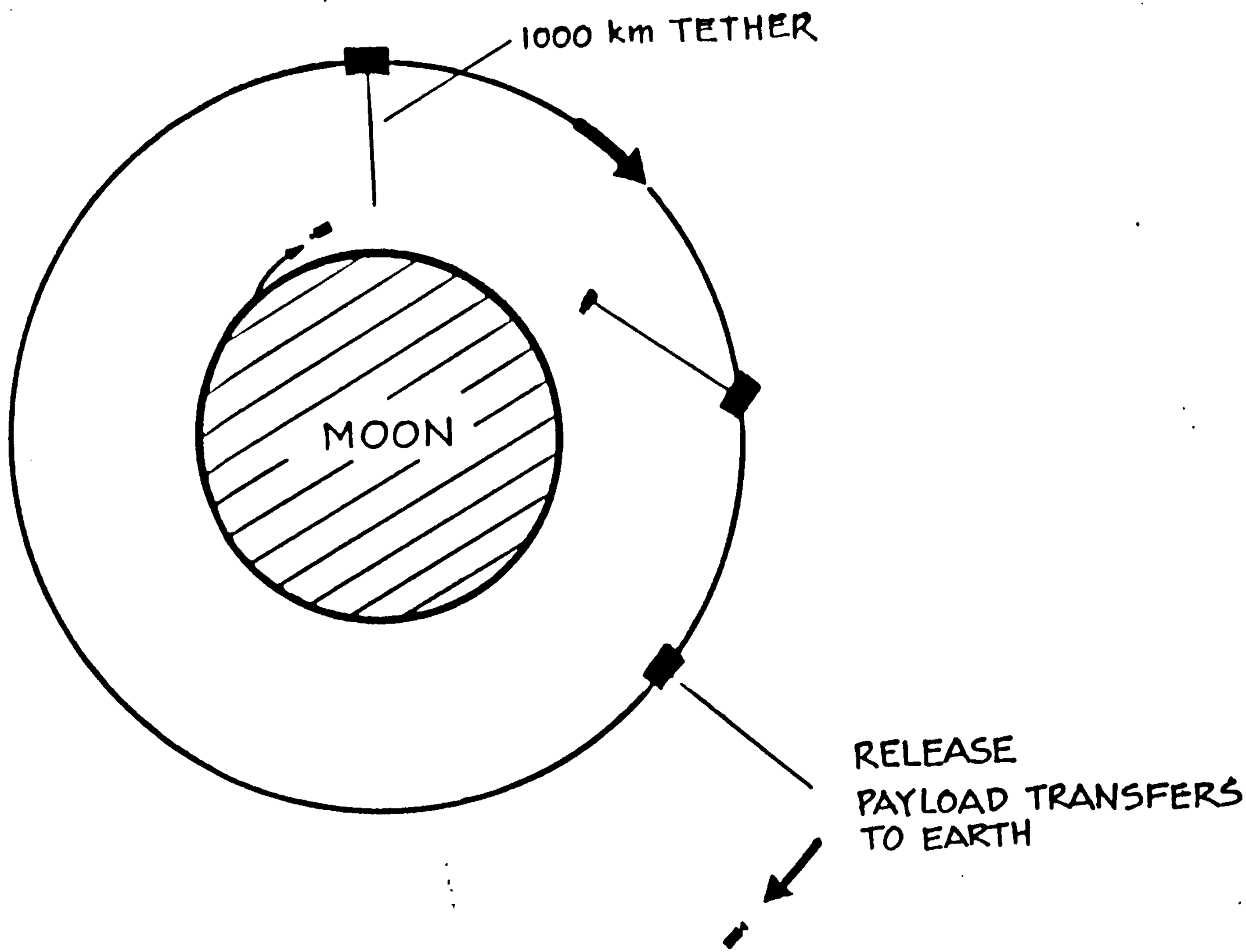
SPINNING TETHER
IN LEO



TETHER TRANSFER ORBIT



HANGING TETHER IN LLO



Advanced Propulsion for Low Earth Orbit - Moon Transportation: IV. Transportation Model

Mark W. Henley
General Dynamics Space Systems Division
P.O. Box 85990; San Diego, CA 92138

Abstract

A simplified computational model of low Earth orbit - Moon transportation systems has been developed to provide insight into the benefits of new transportation technologies. A reference transportation infrastructure, based upon near-term technology developments, is used as a departure point for assessing other, more advanced technology alternatives. Comparison of the benefits of technology application, measured in terms of a mass payback ratio, suggests that several of the advanced technology alternatives could substantially improve the efficiency of low Earth orbit - Moon transportation.

INTRODUCTION

A computer model has been constructed to assess new technology alternatives as implemented in a reference Earth - Moon transportation infrastructure. This transportation model was developed as part of an Advanced Propulsion for Low Earth Orbit - Moon Transportation study performed by the California Space Institute at the University of California, San Diego. Input for the transportation model has been developed through interaction with participants in this study, and this model's output, in turn, is being used in the study as an input to a separate economic model of infrastructure alternatives (Stern, 1988).

The reference transportation infrastructure employs Orbit Transfer Vehicles (OTVs) for orbit to orbit transfer, OTV-derived lunar landers for transportation between the lunar surface and low lunar orbit (LLO), and Orbital Transfer and Staging Facilities (OTSFs) in low Earth orbit (LEO) and LLO. Technology needed for the reference infrastructure is already in the planning and early development stages (Bialla and Ketchum, 1987).

Advanced propulsion systems that have been compared through the transportation model have included mass drivers, tethers for momentum transfer, ion engines, and laser propulsion. These alternatives were analyzed separately as incremental modifications of the reference configuration and selected promising options were combined. System parameters for configurations using these technologies were determined through the interaction of a team of academic, government, and industry representatives participating in the Advanced Propulsion for LEO-Moon Transportation study, resulting in representative alternative configurations that could be analyzed in the transportation model.

Alternative systems, which use more advanced technology, are compared with the reference transportation infrastructure in terms of mass payback ratio (MPR), the net mass of lunar material delivered to LEO per unit mass of terrestrial material used in the system (Frisbee and Jones, 1983). An MPR greater than one is required for the export of lunar material (such as lunar oxygen) to LEO to be preferred to the transport of similar material up from Earth. The reference transportation system can achieve an MPR slightly greater than one (the system can deliver more lunar mass to LEO than the terrestrial mass needed to produce and transport this lunar mass). MPRs for some of the more advanced system alternatives are high enough to suggest that these technologies should play a major role in future lunar operations.

REFERENCE TRANSPORTATION INFRASTRUCTURE

The reference infrastructure is based upon recommendations of recent internally funded studies at General Dynamics Space Systems Division (Bialla, 1986; Bialla and Henley, 1987), with minor modifications to optimize the system for utilization of lunar oxygen. Figure 1 provides an overview of this reference infrastructure, illustrating the Orbit Transfer Vehicle (OTV), Orbital Transportation and Staging Facilities in low Earth orbit (LEO) and low lunar orbit (LLO), and an OTV-derived lunar lander.

OTV Concept

The OTV concept chosen for this reference infrastructure is modelled after the modular S-4C concept recommended in recent OTV studies (Ketchum, 1985). This space-based, reusable, aerobraked vehicle is illustrated in Figure 2 in its single tankset configuration. The only significant modification of the S-4C for this lunar application is an increase in the aerobrake mass in order to accommodate the large masses of lunar material brought to LEO each time the OTV returns.

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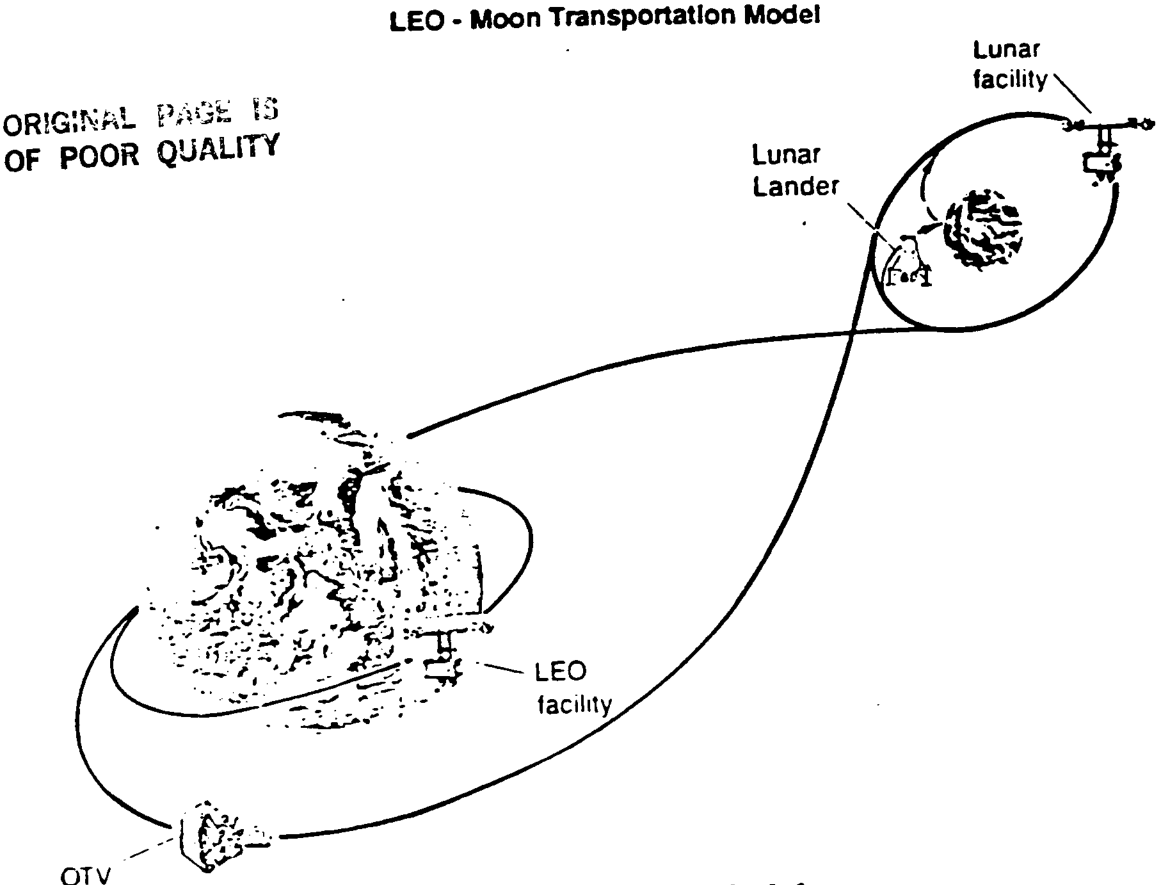

Figure 1. Reference Orbital Transfer Infrastructure.

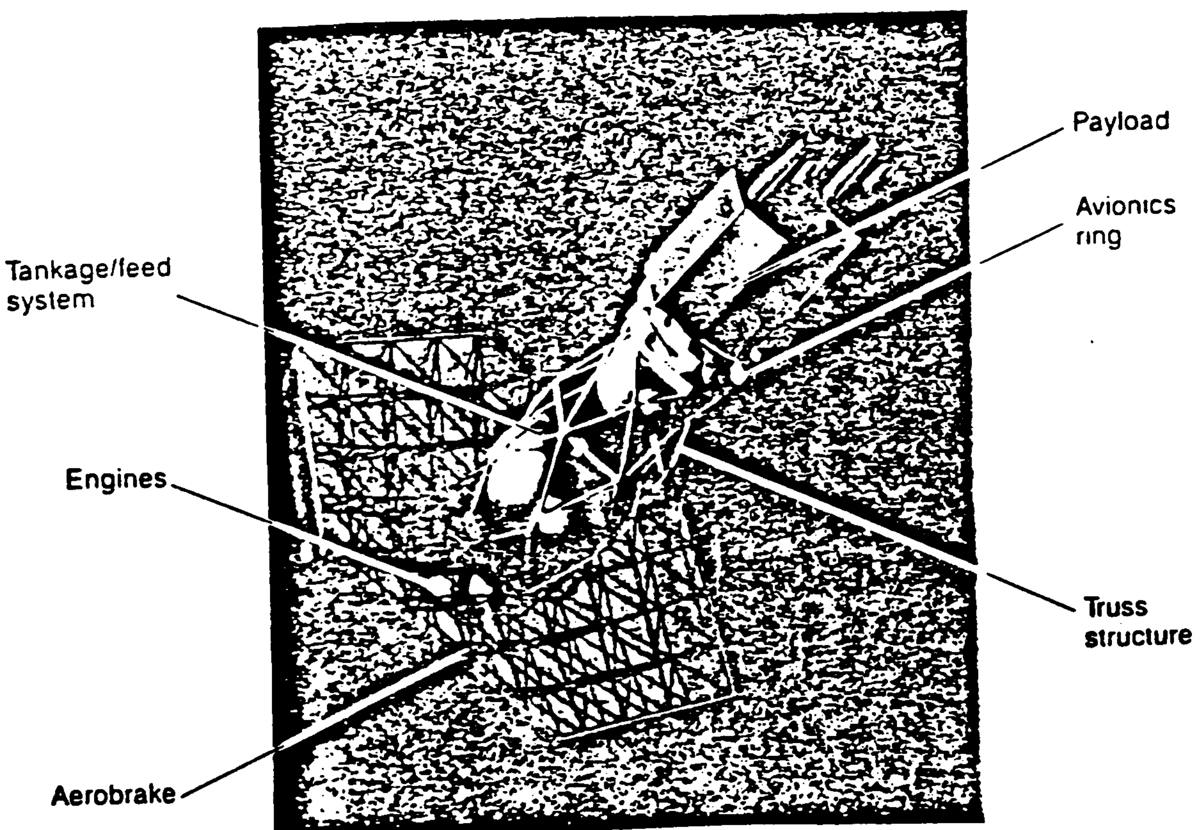

Figure 2. Reference Orbital Transfer Vehicle - Single Tankset Configuration.

The OTV is propelled by two advanced oxygen/hydrogen (LOX/H2) engines of 22,000 N (5,000 lbf) thrust each, with an oxidizer to fuel (O:F) ratio of 6:1 and a specific impulse of 485 seconds. This relatively low thrust level minimizes engine mass, but requires a multiple perigee burn trajectory to reduce gravity losses upon departure from LEO. A modification of this OTV engine for lunar lander applications would make use of a significantly higher mixture ratio (well past the stoichiometric ratio of 7.8:1).

The S-4C OTV concept allows variation of the number of tanksets (sets of individual tanks for LOX, LH2, pressurant, and RCS propellants), with combinations of 1, 3, 4, 5, and 7 tanksets giving the vehicle a wide range of propellant capacity. For the reference OTV, different tankset options have been considered in the analytical model, and the three tankset configuration has been chosen for the reference OTV. The less efficient one tankset configuration might be reasonable for use in early, low mass transport operations required to set up an initial infrastructure, and the most efficient seven tankset configuration might be preferred for eventual, high mass transport operations.

The reference OTV utilizes a fully reusable aerobrake which is sized as a function of the mass brought back to LEO. The aerobrake is specified to be 13% of the total mass entering the earth's atmosphere, a factor which is typical of previous OTV designs for return from geosynchronous Earth orbit (GEO).

Modular avionics on the OTV allow modification of guidance and control systems with advances in the state of the art. The modular avionics approach also allows easy modification of guidance as required for an OTV-derived lunar lander.

Orbital Transportation and Staging Facilities

Two Orbital Transportation and Staging Facilities (OTSFs) are utilized in the reference infrastructure, one in LEO, and one in LLO. OTSF functions include spare vehicle parts storage, meteoroid and debris shelter, and propellant storage. In the transportation model, these facilities are repositories for lunar oxygen and terrestrial hydrogen. With an OTSF present in LLO, the lunar lander can deliver lunar oxygen to LLO while the OTV is in transit between LLO and LEO.

A representative LEO OTSF is illustrated in Figure 3. Its subsystems are derived from Space Station hardware, and, in this reference infrastructure, it co-orbits with the Station at 28.5° inclination. Telerobotic operations are expected to be the normal means of maintenance, propellant transfer, and payload processing.

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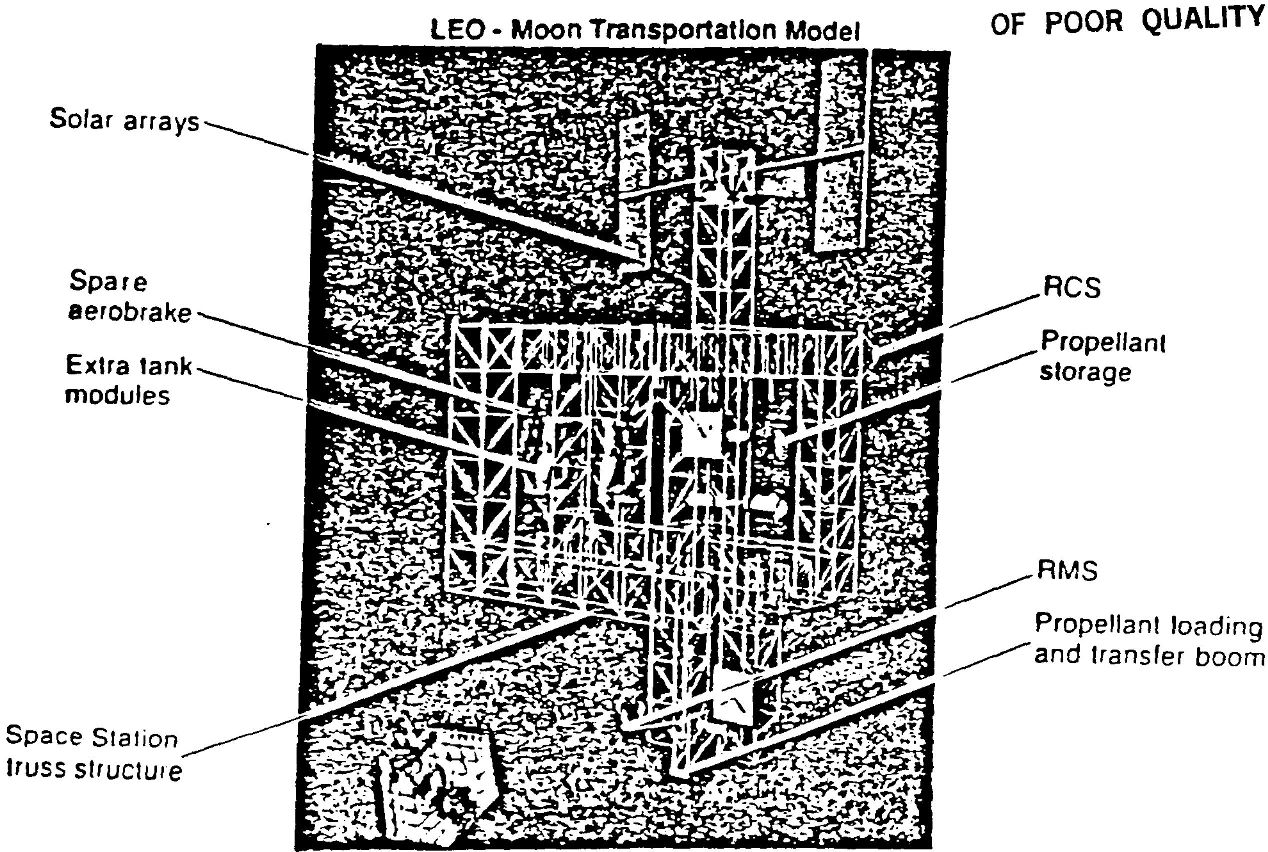

Figure 3. Representative Orbital Transfer and Staging Facility in Low Earth Orbit.

The representative LLO OTSF shown in Figure 4 is similar to the LEO facility in most respects. The lunar facility uses a more advanced solar power system (also derived from Space Station hardware), and has a larger OTV hangar for multiple vehicles. This facility contains several manned modules, and is expected to evolve with time, to eventually serve as a staging base for Mars missions using LOX (Bialla, 1986, Cordell and Wagner, 1986). OTSF concepts illustrated here result from internal General Dynamics studies. More detailed definition of such systems is needed, including design adaptable to later modification by more advanced technology.

OTV - Derived Lunar Lander

The reference lunar lander is illustrated in Figure 5. This configuration is derived from the OTV by substituting landing gear in place of the aerobrake, and thus has common subsystems and interfaces for propellant handling. More sophisticated avionics packages are substituted for the additional requirements of launch and landing. A single tankset derivative of the OTV is used for the reference lunar lander, as the thrust from its two engines would be insufficient to lift a larger lander (with full LOX tanks) from the Moon's surface. The most significant feature of the lander selected for the reference configuration is the modification of the basic OTV engine for operation at a higher mixture ratio. The purpose of this vehicle is the transport of LOX from the Moon's surface to LLO, and the return to the surface with logistic supplies and enough hydrogen for the next trip up to LLO.

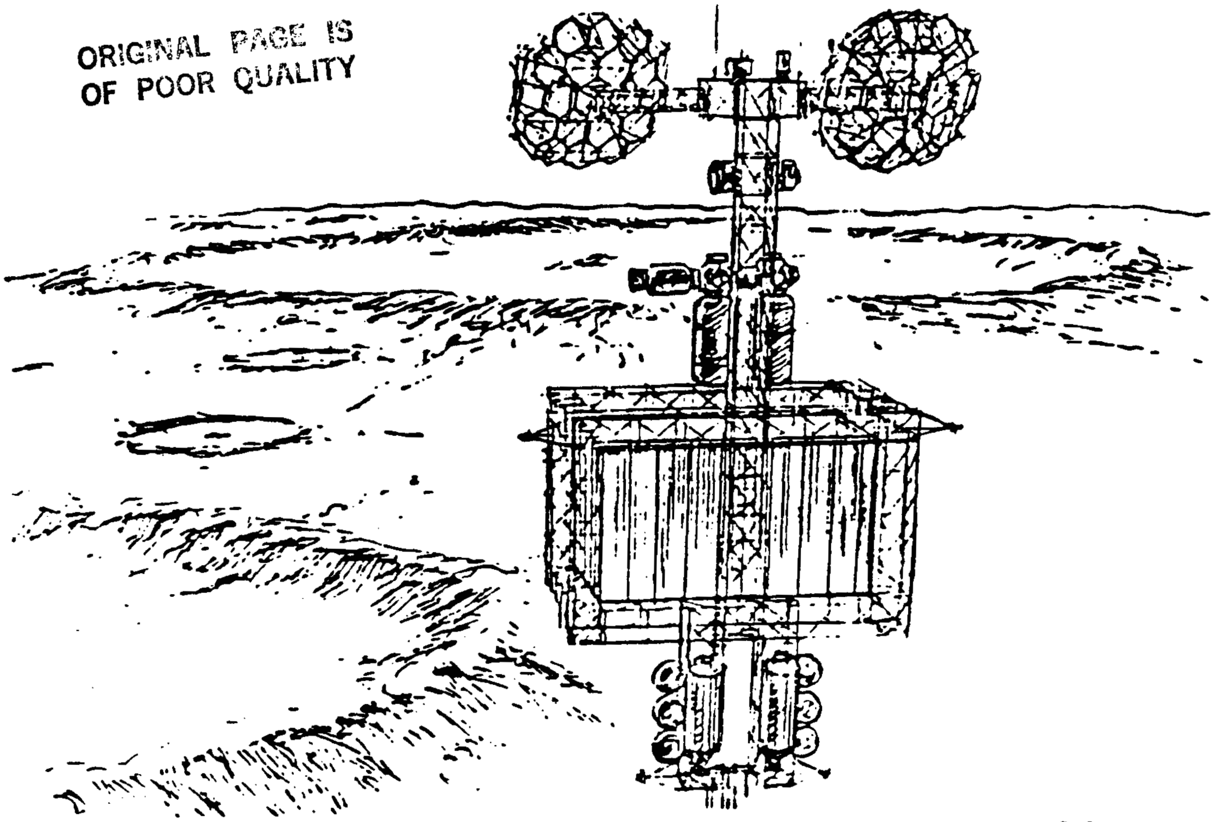

Figure 4. Representative Orbital Transfer and Staging Facility in Low Lunar Orbit.

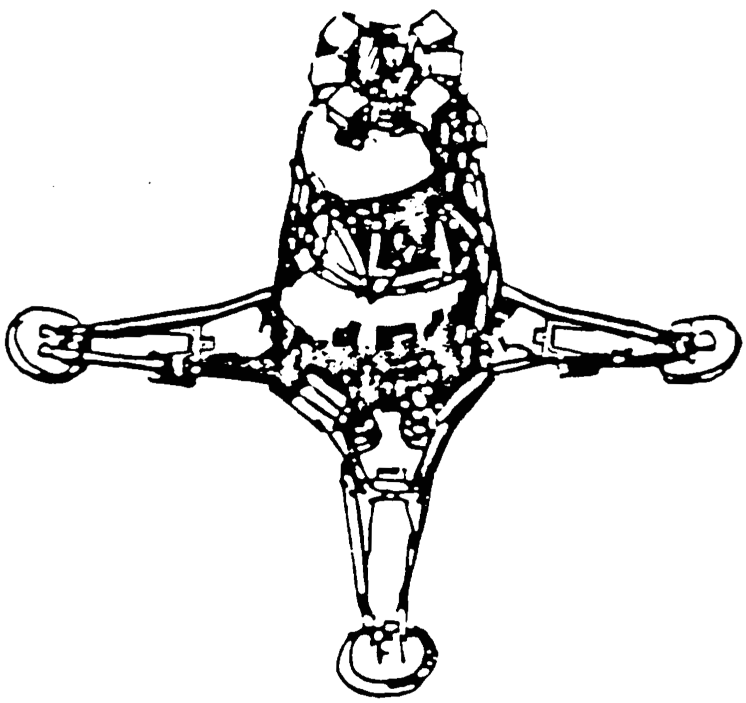

Figure 5. Reference Lunar Lander Derived from Orbital Transfer Vehicle Subsystems.

Engine performance as a function of O:F ratio (the ratio of oxygen used to hydrogen used) follows the trend of the curve in Figure 6. This curve is based upon the output of a General Dynamics computer program, for one dimensional equilibrium LOX/H2 combustion in an engine with a 100 bar (1500 psi) chamber pressure and an area ratio of 400. Higher chamber pressures and area ratios would generally increase the engine's specific impulse. (Optimal area ratios may actually be lower due to factors such as increased weight and radiative energy losses associated with large engine nozzles.) As the mixture ratio increases beyond the maximum region (around 6:1), the specific impulse (force divided by mass flow rate) decreases. Lunar lander applications can achieve higher mass payback ratios at higher mixture ratios in spite of this decrease in specific impulse, as the oxygen used is nearly free, while hydrogen must be imported from Earth. O:F ratios selected for the OTV and the lander were arrived at by iterative trial of mixture ratio (and corresponding Isp) parameters in the transportation model. The indicated lander O:F ratio of 12 was a compromise; slightly better mass payback ratios would result if the lander engine were operated at a higher O:F ratio (>12) for liftoff, and at a lower ratio (<12) for landing.

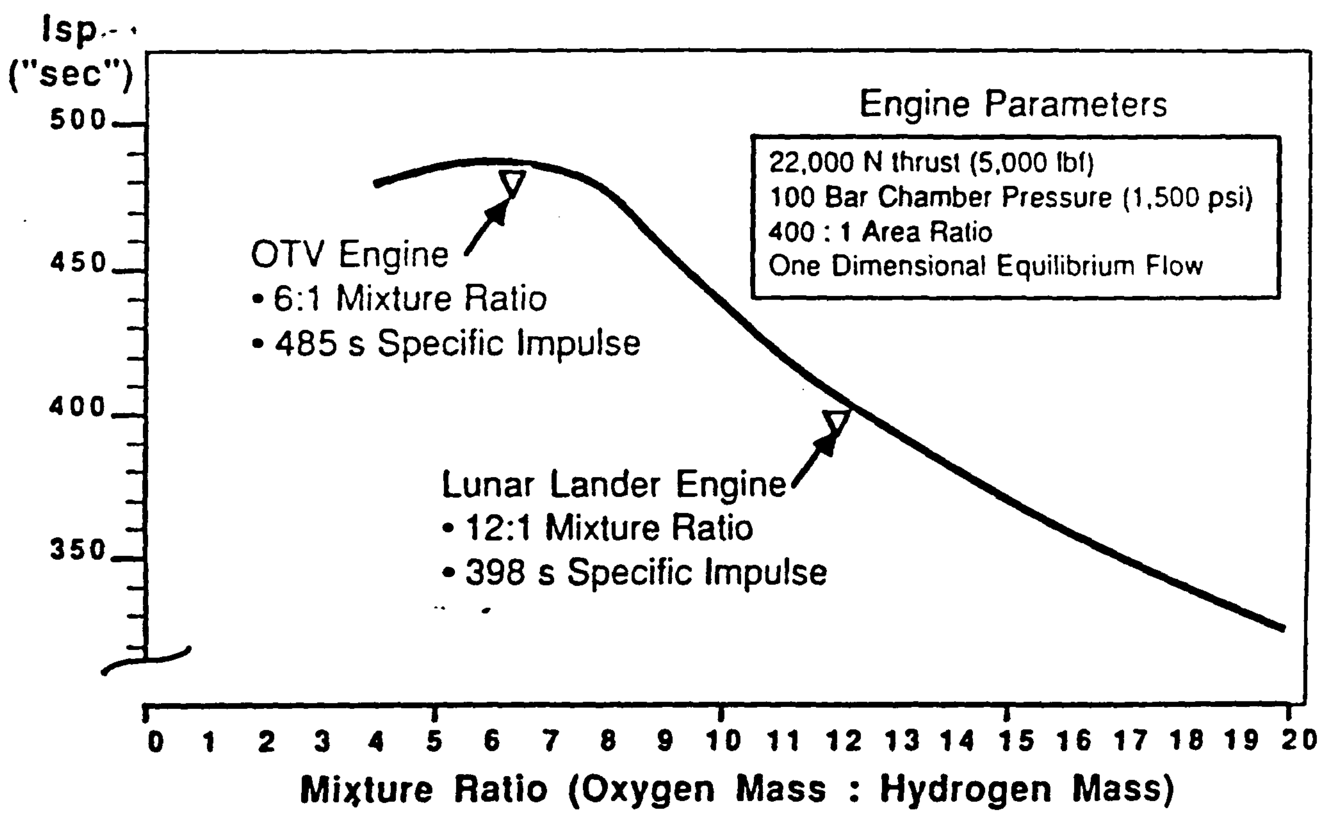

Figure 6. Engine Performance as a Function of Oxygen :Hydrogen Mixture Ratio.

Technology Development Requirements

The reference transportation infrastructure in this model presumes fruition of certain technology developments for reusable Orbit Transfer Vehicles (OTVs), OTV-derived lunar landers, space-based OTV accommodations, and the lunar surface base. Key OTV technology in the reference case includes aerobraking, advanced oxygen/hydrogen engines, advanced avionics, and lightweight structures. Technology for space-based OTV servicing at an Orbital Transfer and Staging Facility (OTSF) includes telerobotic maintenance, zero-g propellant transfer, and automated rendezvous and docking. New technology is also needed for lunar materials processing to produce liquid oxygen propellant for the OTV and lunar lander. In order to utilize this lunar oxygen most effectively, the lunar lander uses an engine with a high O:F ratio.

Modification of a basic OTV engine to operate at a higher mixture ratiosfor lunar lander applications is considered to be a reasonable evolutionary step for an engine that is still in the early stages of technology development. Engine technology development activities sponsored by Lewis Research Center (such as the use of gaseous oxygen to drive LOX turbopumps), are relevent to such an increase in O:F ratio. Similar high O:F ratio and variable O:F ratio engines are being studied for Earth-to-orbit applications, where the increase in O:F ratio can reduce launch vehicle dry mass (Martin, 1987). Small oxygen/hydrogen engines at the stoichiometric (7.8:1) ratio have aready been developed for use on satellites (Stechman and Campbell, 1973) and on the Space Station (Aerospace America, Sept, 1986).

ANALYTICAL MODELLING OF TRANSPORTATION INFRASTRUCTURES

An analytical model has been developed to compare advanced technology alternatives against this reference architecture. This model uses Excel spreadsheet software to apply an iterative series of equations to alternative transportation systems. This relatively simple model can easily be modified to consider variations of input parameters, and can be run rapidly on a personal computer.

The analytical model of the lunar transportation infrastructure considers separate loops for LEO-LLO and LLO-lunar surface transportation as illustrated in Figure 7. The lunar lander: a) leaves the surface with a full load of LOX (35,000 lbm) and enough hydrogen to reach LLO; b) transfers excess LOX to the lunar OTSF (retaining enough to return to the surface) and receives hydrogen and logistics mass to make the next round trip and produce the next load of LOX; and c) returns to the surface to complete this loop. For the reference case, the lander must make approximately 7 round trips to the lunar OTSF to transport the LOX that will be transferred later from the OTSF to fill the three tanksets of the OTV. The OTV loop: a) leaves LEO with enough hydrogen to make the round trip, enough oxygen to reach LLO, and the payload (hydrogen and logistics mass) required to support the ~7 lander loops; b) delivers the payload to LLO and refills LOX tanks at the lunar OTSF; and c) returns to LEO with excess LOX. The ratio of this excess LOX (beyond that required for the next trip up) to hydrogen and logistic mass is termed the mass payback ratio. This ratio (1.31:1 for the reference infrastructure) is a basis for assessing new technology alternatives to the reference system.

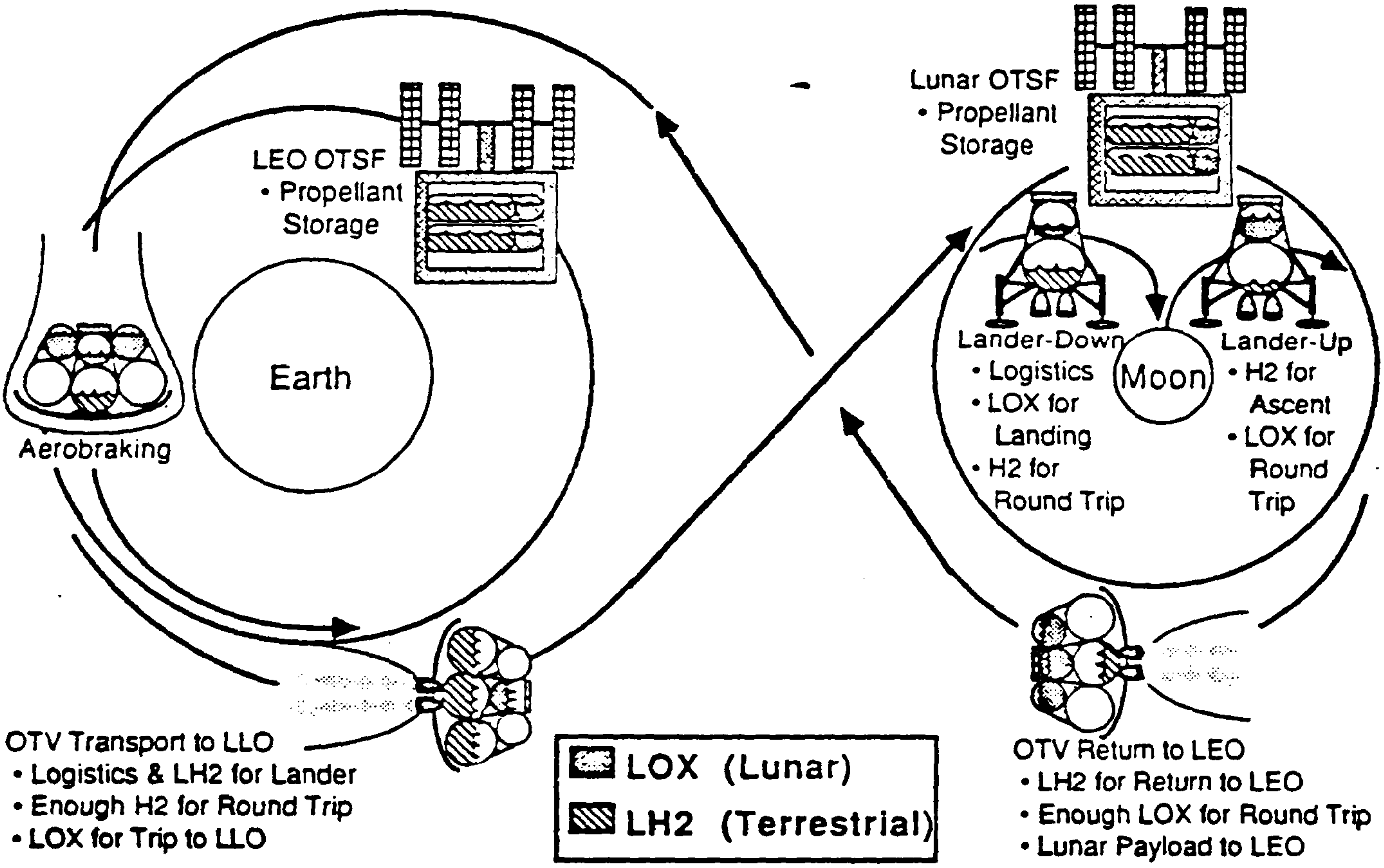

Figure 7. Transportation Model Overview.

Material on the surface of the moon is at a higher potential energy level than the same mass in LEO, as is illustrated in Figure 8. If we could construct a "siphon" between the moon's surface and LEO, mass would flow freely, and if we placed a "turbine" in this mass flow, a tremendous amount of energy would be released. In the reference system, we construct such a "siphon, although it is not very efficient in mass transfer (requiring an input of mass from Earth), or in energy conversion (dissipating energy by aerobraking). Alternative systems that supplement the reference configuration by more advanced technology are generally more efficient in mass transfer and/or energy conversion.

LEO - Moon Transportation Model

Velocity increments used in the transportation model are listed here. These "Δ V" requirements, based on Appollo-type free return trajectories, are greater than those that might be expected in practice, as a free return option (in case of problems) will be less critical when supplies and assistance are available in LLO and at a lunar base. For an unmanned OTV, much longer flight times might be reasonable, with attendant reduction in its mission ΔV requirements. The altitude and eccentricity of "low" lunar orbit have not been optimized (with corresponding changes in the individual velocity increments) for the reference or alternative infrastructure, but such an analysis would probably result in greater mass payback ratios. Gravity losses for the lander (which transports more mass upward than downward) could be higher in ascent than in descent, tending to exchange the delta Vs attributed to these mission phases.

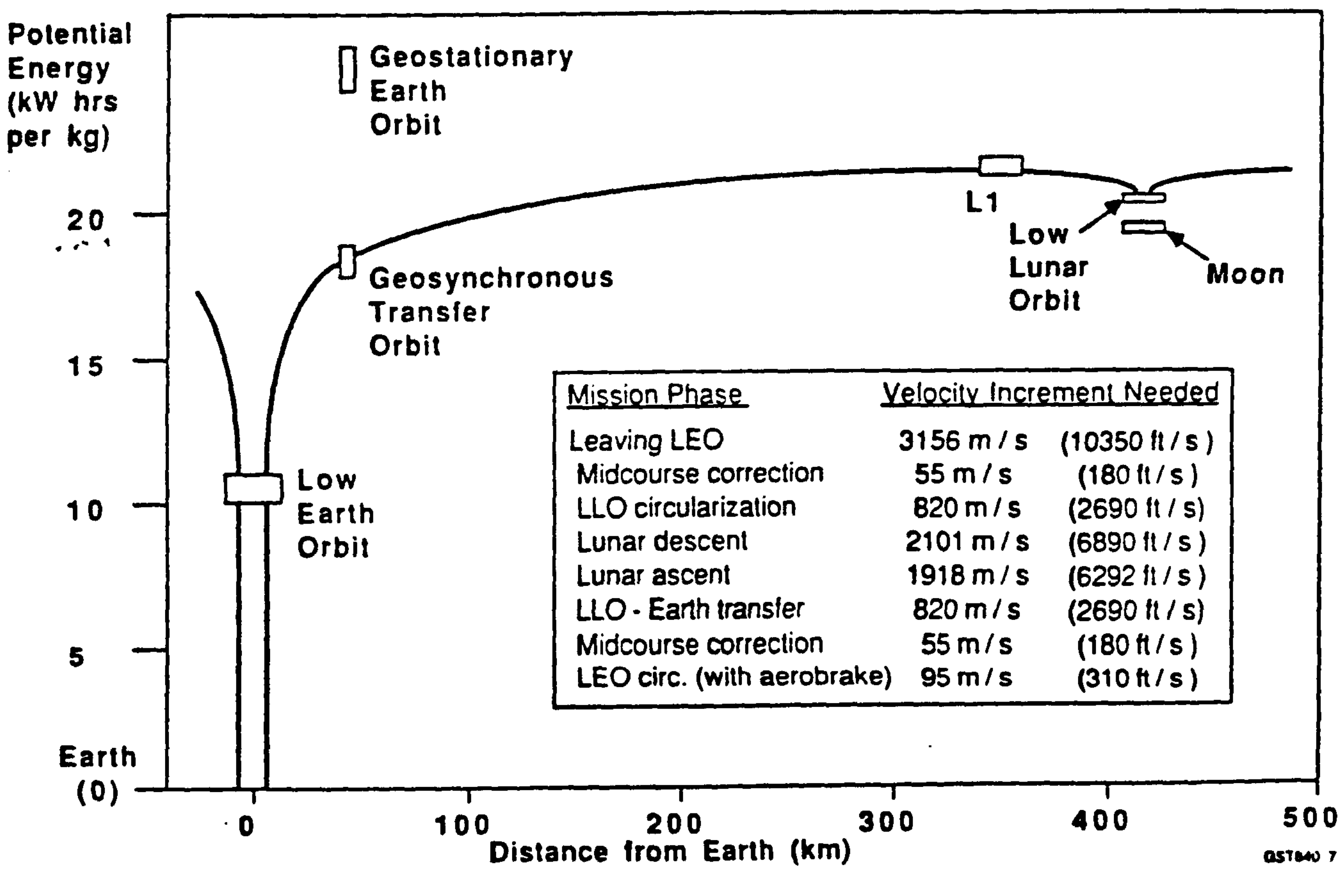

Figure 8. Potential Energy of Lunar Materials.

Hydrogen is the major component of the OTV's payload from to LLO. For cases in which hydrogen usage exceeds OTV capacity, additional tankage, weighing 10% of the contained propellant, is presumed to be carried to LLO (and left there). The OTV's hydrogen tankage is actually oversized for most mission propellant requirements, and thus, <u>if the logistic mass is hydrogen</u>, it might be carried directly within OTV tanks. For example, production of oxygen by reduction of ilmenite and subsequent water electrolysis (Gibson and Knudson, 1985) would use hydrogen as a principal reagent:

$$H_2 + Fe\,Ti\,O_3 = H_20 + Fe + TiO_2$$

$$2\,H_2O = 2\,H_2 + O_2$$

If all of the hydrogen used in this reaction is not recovered, hydrogen might comprise a substantial portion of the logistics mass required for lunar oxygen production. The transportation model assumes that one unit of terrestrial mass must be delivered to the moon's surface for every one hundred units of lunar mass produced on the moon (oxygen or other useful lunar products). Spare parts for OTV, OTSF, and LLOX production facility maintenance are not separated from other logistics in this transportation model, however both their unit cost and transportation cost are included in an economic model (Stern, 1988), which uses the output of this transportation model.

This LEO - Moon transportation model describes steady state operations, assuming that the lunar base, including a LLOX production plant, is already established for reasons other than transport of lunar material to LEO (e.g., scientific exploration). The reference infrastructure would initially transport men and supplies for a manned lunar base, and thus "bootstrapping" of the system (to provide for its own development) is not considered. Expansion of the system for higher LLOX production and transportation rates would require a temporary increase in the flow of mass from Earth, with a return to steady state operation after system expansion is complete.

TRANSPORTATION MODEL RESULTS

The transportation model has been used both in refining the reference transportation infrastructure and in assessing modifications of this infrastructure with more advanced technology. Results of calculations using the transportation model are portrayed in the following charts, with mass payback ratio indicated on the vertical axis. While the scale changes somewhat to accommodate the range of results, the reference transportation system's MPR of 1.31 is indicated on all of the charts by a dashed line, and a solid line indicates an MPR of one (the limit for economic feasibility of transport of material down to LEO from the Moon, rather than up from Earth).

The significance of both the number of OTV tanksets and the high mixture ratio for the lunar lander is illustrated in Figure 9. As the number of OTV tanksets increases, the system yields greater mass payback ratios. A large improvement is realized by increasing from one to three tanksets, with far less benefit thereafter. The three tanset OTV configuration is considered to be most desirable, as it achieves relatively high MPRs, yet keeps the total LOX load (which the lunar OTSF must store prior to transfer into the OTV) at a reasonable level. When the three tankset OTV is combined with a 6:1 mixture ratio lunar lander, it obtains an MPR slightly greater than one (1.07); however, the use of the 12:1 lander results in a much greater MPR (1.32). The difference between these MPRs becomes significant when one considers that the net gain per unit mass invested in the 6.25:1 lander case is only 7%, as compared to a 32% gain in the case of the 12:1 lander. The lower mixture ratio lander is, in fact, marginal for use with the three tankset OTV, as unforeseen difficulties could easily turn this small mass profit into a net mass loss. MPRs for the lower mixture ratio lander configuration improve somewhat as the number of OTV tanksets increases. However, the MPRs for the 12:1 mixture ratio lander also increase by similar increments. The selected reference system, with three tanksets on the OTV and a 12:1 mixture ratio for the lander, is clearly indicated on Figure 9 by the bold bar.

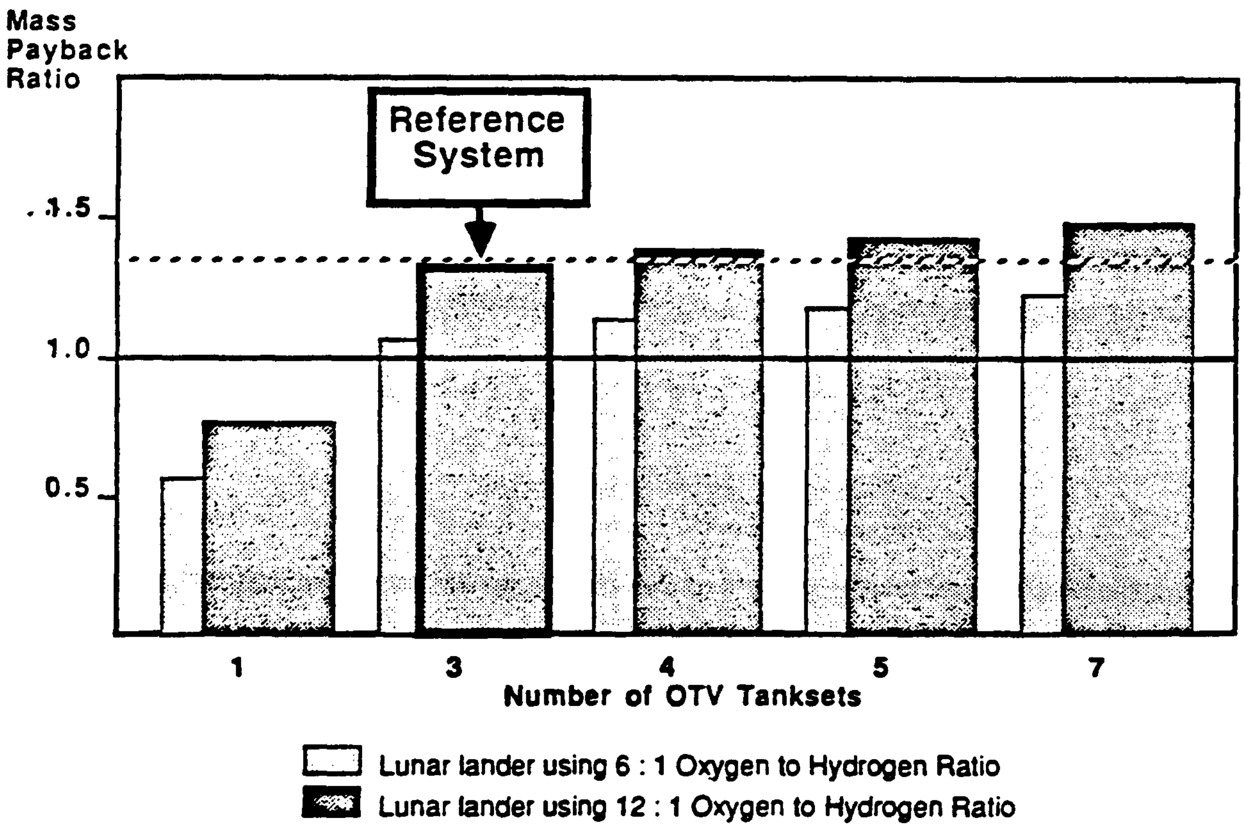

Figure 9. Reference Infrastructure: Sensitivity to Number of OTV Tanksets and Lander O:F Ratio.

Aerobraking is essential to the success of the reference system, and the mass of the aerobrake is a dominant factor in its mass payback ratio. If aerobrakes can be produced on the Moon, substantially larger mass payback ratios may result; the OTV would not have to carry the aerobrake mass from LEO to LLO, but the lander would instead carry the aerobrake for the much lower ΔV from the lunar surface to LLO (Duke et al, 1985). Figure 10 illustrates the sensitivity of mass payback ratio to aerobrake mass for the reference OTV, and for alternative configurations that use aerobrakes produced on the Moon. Aerobrake mass is varied here as a percent of mass entering the Earth's atmosphere. Nominally, 13% of entry weight is used for the reference system's aerobrake, resulting in large aerobrake masses, as the returning OTV's mass (with nearly full LOX tanks) is relatively large. Multiple aeropass trajectories, with each pass successively lowering perigee, might reduce the aerobrake mass required. If lunar aerobrake manufacture proves to be feasible (for example, using the titanium dioxide by-product of ilmenite reduction as a refractory heat shield material, the aerobrake mass could be significantly higher than that of an aerobrake manufactured on Earth and still be competitive. An expendable lunar aerobrake (discarded at LEO) weighing 25% of the entry mass would be preferable to the reference system. If the used lunar aerobrake had intrinsic value in LEO (if the mass of the brake discarded at LEO is considered to be part of the payload to LEO), the mass payback ratio would continue to increase with increasing aerobrake weight. While the possibility of manufacturing aerobrakes from lunar materials is clearly attractive, the terrestrial aerobrake is retained as a baseline for the reference system.

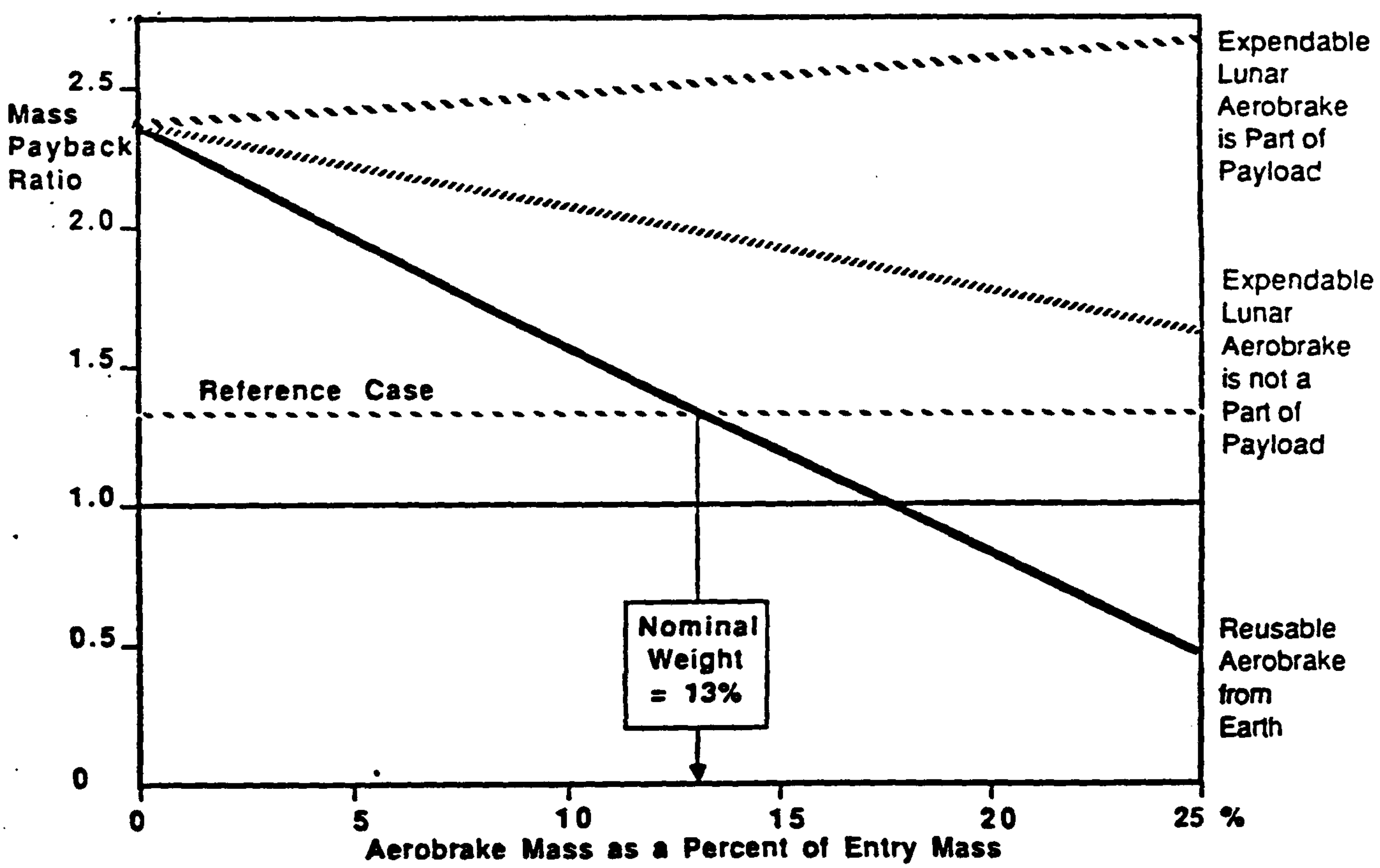

Figure 10. Reference Infrastructure: Aerobrake Weight Sensitivity.

Tether - Assisted Transportation

Alternative systems which use tether-assisted OTV transportation have been emphasized in this Advanced Propulsion for LEO-Moon Transportation study (Arnold and Thompson, 1988; Babb 1988; Stern, 1988). These systems are considered in the model as modifications of a reference transportation facility in LEO or LLO, or as an additional facility in an elliptical Earth orbit (EEO). Tether-assisted transportation alternatives are assumed to compensate for any net imbalance in momentum exchanged toward and away from the Moon through high-Isp propulsion (e.g., ion engines) using propellant from the Moon.

Tether-assisted transportation systems can reduce the ΔV requirements of the vehicles in the reference transportation infrastructure, and thereby increase payload (multiple references). The ΔV supplied by throwing or catching the OTV or lander with a tether is subtracted from the velocity increment needed for a given mission phase. Velocity increments of 500 m/s (1640 ft/s) and 1 km/s (3280 ft/s) are considered for each tether system alternative. The tether that can throw (release) a vehicle with an initial 500 m/s velocity, but not catch a similar incoming vehicle, is the least ambitious of the alternatives selected for study, and would be the most reasonable for consideration in "near-term" (early 21st century) transportation between LEO and the moon. Tether-supplied velocity is limited to the maximum velocity increment needed, thus the "1 km/s" system in LLO would throw an OTV toward Earth at 820 m/s (2690 ft/s), the velocity used to escape from LLO. Similarly, 95 m/s (310 ft/s) is the maximum increment achieved in catching an aerobraked OTV to achieve circularization at LEO.

Tether platforms can also provide a means of energy storage (Arnold and Thompson, 1988; Babb, 1988). Consider a platform in eccentric Earth orbit (EEO) with the capability to throw the OTV outward toward the Moon. The OTV uses chemical propulsion to transfer from LEO to EEO, docks with the tether facility, and is thrown by the tether. The momentum given to the mass of the OTV by throwing it at some initial velocity must be compensated by an equal and opposite change in the momentum of the platform in EEO (its mass multiplied by its ΔV). If the platform is heavy relative to the OTV, its resulting velocity change will be small, with little change in its orbital trajectory (a somewhat lower apogee if the OTV is thrown at perigee). Upon returning from the LLO, the OTV aerobrakes into EEO, docks with the platform, and is then thrown downward into LEO, at the required remaining ΔV. The momentum of the EEO platform is now changed in the opposite direction (returning to a higher apogee if the OTV is thrown at perigee). Energy transfered to the platform by the action of throwing the OTV toward LLO is thereby returned to the OTV as it is thrown down into LEO.

Similar momentum transfer could be achieved at a tether platform in LEO, which de-orbits mass returning to Earth in exchange for upward boosting of OTVs toward the Moon, or at a platform in LLO, which exchanges momentum gained in the downward boost of lunar landers for the outward boost of OTVs returning to LEO. If platforms can be made to catch vehicles as well as throwing them, further improvements in energy storage can be obtained, with additional increases in MPR. While such transfers of momentum do not fully cancel in practice, the net momentum deficit or surplus is substantially reduced.

In a system with an MPR greater than one, the net momentum imbalance will tend to make the tether platform move toward the Moon as the net lunar mass transported by vehicles moves toward Earth. Momentum could be balanced by several methods, including:

1) Throwing the vehicles at a lower velocity toward the Earth than the velocity with which they are thrown toward the Moon,
2) Sending additional mass from Earth toward the Moon,
3) Conversion of orbital energy into other forms (e. g., into electrical energy via an electrodynamic, conducting tether cutting through geomagnetic field lines, or
4) Consumption of propellants at the affected platform.

Platforms equipped for tether-assisted transportation are presumed to use low thrust, high specific impulse propulsion to cancel any net momentum imbalance. The propellant for such momentum makeup is considered to be a lunar product, and is for the purposes of the transportation model, included as a part of the lunar oxygen produced and transported. Argon, known to be present in substantial concentrations (~0.3%) in lunar regolith, is easily released by heating (Kirsten and Horn, 1974), and could be a reasonable propellant choice in place of oxygen. A specific impulse of 5000 s is presumed for momentum makeup, consistant with the value used for ion engine OTV propulsion discussed later. As the net momentum deficit or surplus is generally small, mass payback ratios are not very sensitive to this selection of advanced propulsion for the facilities equipped for tether-assisted transportation.

Figure 11 contrasts the MPR achieved through tether-assisted transportation from a facility in LEO, EEO, or LLO. Each case considers two velocity increments supplied in a system that 1) only throws vehicles, and 2) both throws and catches vehicles. While any of these alternatives is clearly better than the reference case, several interesting observations can be made through comparison of the alternatives with each other. The LEO tether facility gains little by adding the ability to catch, due to the small velocity needed for circularization of the OTV from in its low perigee orbit after aerobraking. (Tether-assisted transportation of mass between Earth and LEO has not been considered for the LEO OTSF, due to the groundrules of the present study, but would tend to increase its effective MPRs). The EEO tether facility, in contrast, would benefit considerably from the ability to catch vehicles in addition to throwing them. The increased MPRs for the EEO facility, however, must be traded against the increased operation complexities of such a system (Babb, 1988). Tether-assisted transportation from the LLO OTSF results in the largest MPRs for any single facility location, as the facility is used to reduce propulsive velocity requirements for the lunar lander as well as the OTV. Here the MPRs achieved by throwing alone equal or exceed those that would be obtained by combined throwing and catching from LEO or EEO facilities. The improvement in MPR that would result from a LLO facility that could catch as well as throw is also far more significant than that for the previous LEO and EEO cases.

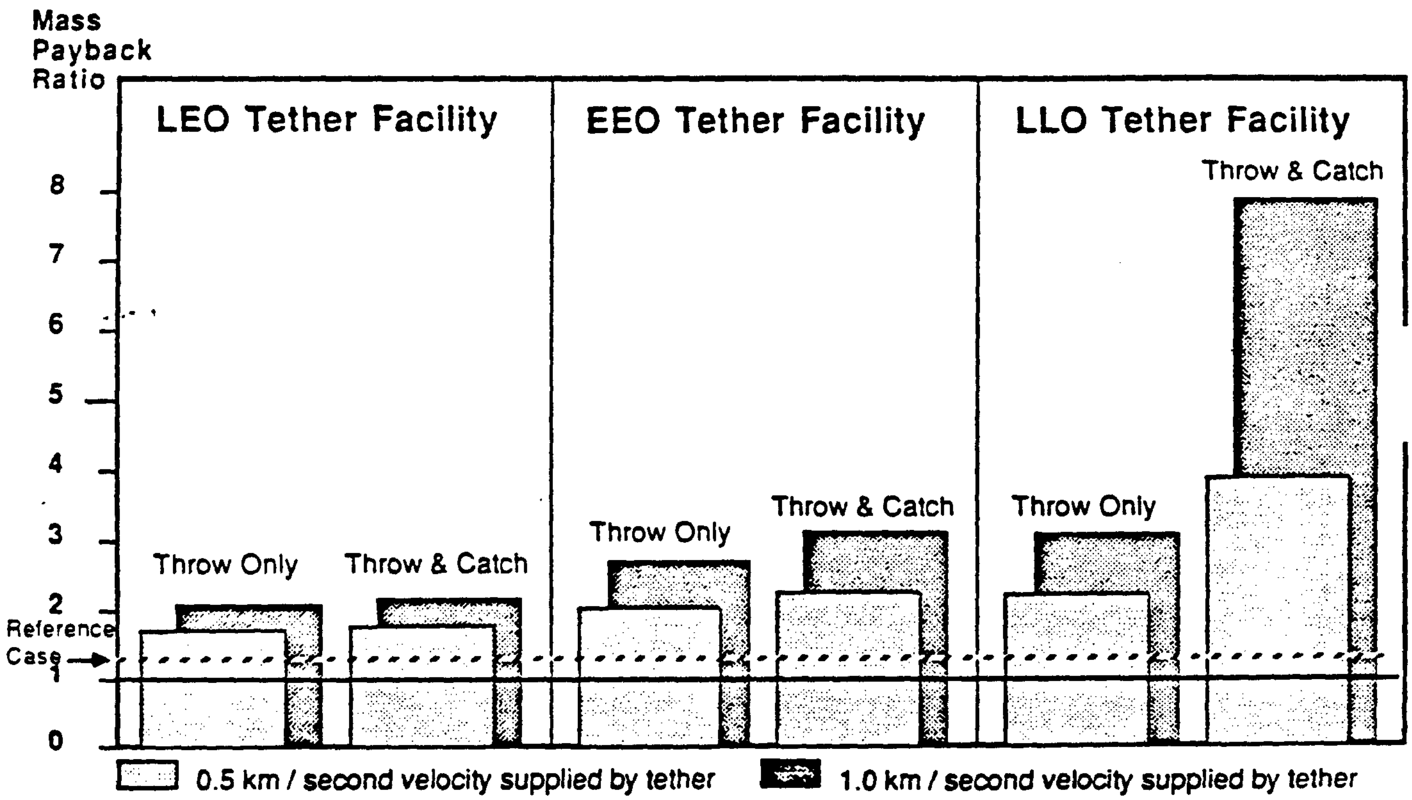

Figure 11. Tether-Assisted Transportation Infrastructure Comparison.

At a high enough velocity, catching and throwing the OTV with a tether may be preferable to aerobraking (Eder, 1987). Figure 12 plots the MPR achieved with and without the use of an aerobrake versus velocity supplied by tether for the case of a tether facility in EEO that can both throw OTVs and catch them. As calculated using the transportation model, the aerobrake becomes a detriment, rather than an asset, if the tether facility can impart a velocity of approximately 1.4 km/s both in throwing and catching. At low tether-supplied velocities (below 0.7 km/s), this type of system would be less effective than the reference infrastructure.

Combined Systems

Two combined systems have been selected to date for investigation by the working groups involved in the Advanced Propulsion for LEO - Moon Transportation study. These systems, hanging tethers in LEO and LLO, and spinning tethers in LEO and LLO, are identical as evaluated in the transportation model. Results from the transportation model would apply equally well to the use of swinging tethers, which may be another reasonable alternative.

Figure 14 illustrates the LEO and LLO systems alone (as they were shown in Figure 11) and the combined system of tether-assisted transportation from both LEO and LLO. We can again improve the MPR substantially through the combination of two similar or identical systems in LEO and LLO. The development cost of two such facilities should be a relatively small increase over that for a single facility.

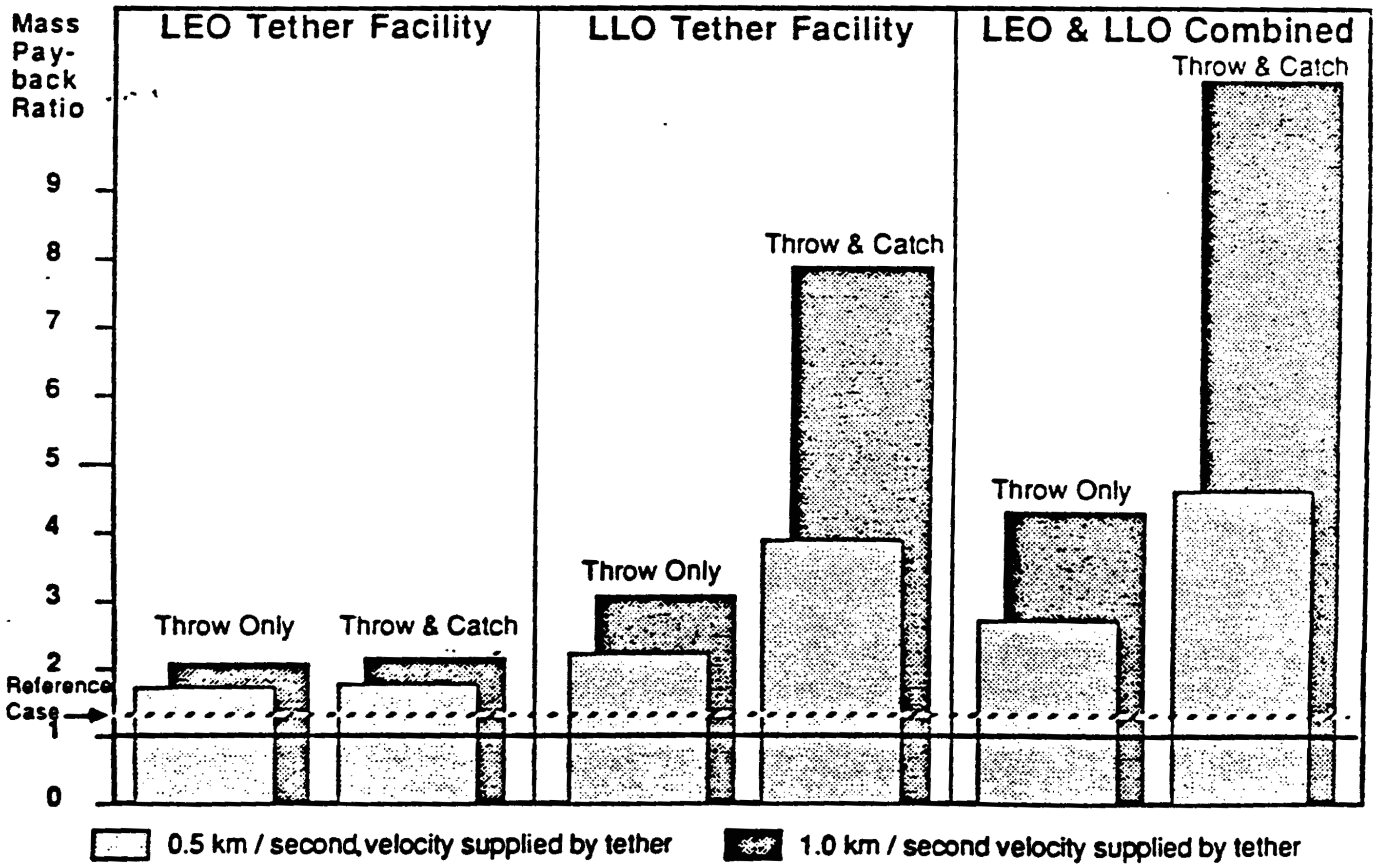

Figure 14. Combined Tether-Assisted Transportation in Low Earth Orbit and Low Lunar Orbit.

CONCLUSIONS AND RECOMMENDATIONS

The results produced by this LEO-Moon transportation model suggest that advanced technology, applied in a reference LEO-Moon transportation infrastructure, can significantly improve the potential for lunar resource utilization in LEO. High mixture ratio lunar lander engines are important for efficient use of lunar oxygen in this reference system. Aerobrake production on the moon could also have a dramatic effect in increasing the MPR achieved by the reference system. Tether-assisted transportation, as applied at individual facilities in a LEO, EEO, and LLO, provides a significant improvement in MPR over the reference infrastructure, with a lunar facility being the most attractive single choice. Combined facilities for tether-assisted transportation in LEO and LLO provide a further improvement in MPR. Other new technologies of laser OTV propulsion, ion engine OTV propulsion, and mass driver use for transport of lunar material to LLO are also attractive, when implemented in the modified reference infrastructure.

In order to reap the benefits of such advanced technology options early in the development of permanent lunar operations, exploratory research and development is required in the near term. It is hoped that the analysis reported here will give impetus to the planning for lunar transportation infrastructure evolution, and to the timely development and implementation of such new technology alternatives. More detailed analytical studies of reference and alternative systems are needed, as well as further conceptual definition of reference systems which are designed for modification over time as new technologies mature. Technology development for reference and alternative systems is necessary now for assessment of appropriate design considerations for initial LEO-Moon transportation systems.

ACKNOWLEDGEMENTS

Credit is due in several respects for assistance in development of this transportation model. The participants in the Advanced Propulsion for LEO-Moon Transportation study have made useful contributions in defining the advanced and reference configurations to be studied. Bob Ford's performance calculations for high O:F ratio propulsion were an important input to the reference lunar lander definition. Howard Bonesteel, Mike Felix, Alan Schneider, and Martin Stemare thanked for their comments on earlier drafts of this paper. The research reported herein fulfills part of the requirements for the master of science degree in mechanical engineering (systems science) through the University of California, San Diego. General Dynamics Space Systems Division has assisted in tuition reimbursement, computer resources, and arrangements for publication and presentation of this paper.

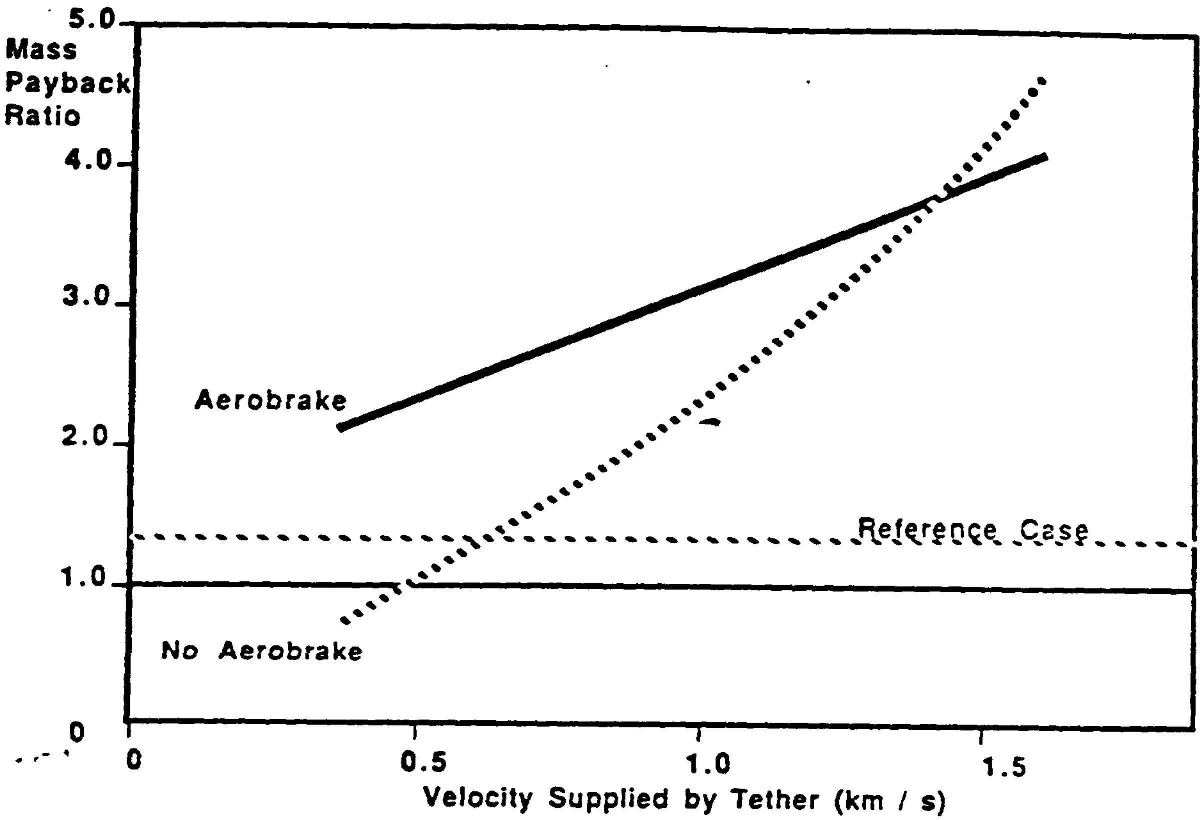

Figure 12 Eccentric Earth Orbit Tether-Assisted Infrastructure: Aerobrake versus No Aerobrake.

Other New Technology Applications

Other modifications of the reference infrastructure with new technology could also increase MPR substantially. Figure 13 compares laser OTV propulsion, ion engine OTV propulsion, and a lunar mass driver as modifications to the reference system.

The laser propulsion case, as defined by Ron Glumb of Lawrence Livermore Laboratory, uses a laser to heat hydrogen propellant for departure of the OTV from LEO. The propulsion system from the reference OTV is retained for use in the vicinity of LLO. This alternative results in a relatively high MPR if the aerobrake is retained, but a somewhat lower MPR if the aerobrake is relinquished in favor of carrying additional hydrogen for laser propulsion in return to LEO.

An OTV equipped with an ion engine, as defined by Ralph Lovberg of UCSD's Physics Department, also achieves a very high MPR, provided that it's propellant is supplied from the Moon. This vehicle has a large mass, no aerobrake, and low-thrust ion engines. The low thrust of the vehicle substantially increases the effective mission ΔV, as well as the mission duration. Use of an aerobrake in conjunction with ion engine propulsion was not considered due to the presumption that a large power supply would be needed. Nuclear power safety implications or large, fragile solar cells could prohibit aerobraking. (For the purposes of the transportation model, OTV transportation reached LEO rather than being limited to a higher, "nuclear safe" altitude, which would have required a separate vehicle for intermediate transportation to LEO). If aerobraking were feasible, the mission duration and ΔV requirements for ion engine propulsion could be reduced substantially, with a corresponding increase in MPR.

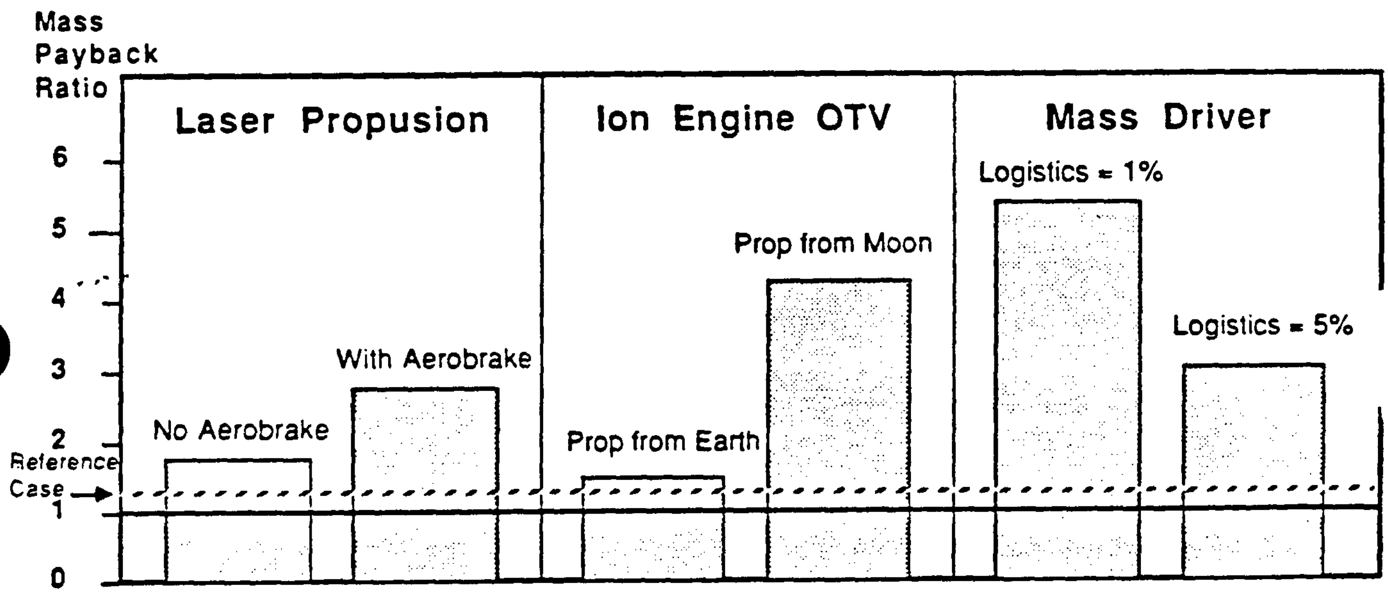

Figure 13. Laser Propulsion, Ion Engine, and Mass Driver Influence on Reference Infrastructure.

A mass driver situated on the Moon would also result in a high MPR. Two cases are considered here through the transportation model, with logistics mass taken down to the Moon by the lander equaling nominal (1%) and increased (5%) fractions of lunar oxygen produced. An increase in logistics mass may be warranted, as the mass driver (as defined by Hugh Davis of Davis Aerospace) launches LOX payloads with apogee kick motors attached for self-circularization in LLO, and these kickmotors are presumed to be imported from Earth. Propellant required for the collection of LOX payloads in LLO would also result in an effective increase in logistic mass requirements.

LEO - Moon Transportation Model

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OFFICE OF EXPLORATION
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VOLUME III: INDEPTH TECHNICAL ANALYSES AND SPECIAL REPORTS
FOR
ADVANCED LIFE SUPPORT SYSTEMS

1.0 LIFE SUPPORT ARCHITECTURE AND TECHNOLOGY REQUIREMENTS IDENTIFICATION

2.0 EXTRAVEHICULAR ACTIVITY SYSTEM REQUIREMENTS DEFINITION

JOHNSON SPACE CENTER

CONTENTS

1.0 LIFE SUPPORT ARCHITECTURE AND TECHNOLOGY REQUIREMENTS IDENTIFICATION

1.1 OBJECTIVE

The overall objective of the life support special assessment study is to conceptualize a Life Support System (LSS) design which is compatible with the Office of Exploration (OEXP) advanced mission case studies. This will allow identification of specific LSS and technology development requirements.

The OEXP advanced mission case studies will be examined in detail to determine how life support is influenced by mission characteristics and interfacing system considerations. This evaluation will be a continuing process as missions become better defined. Top level requirements and a LSS conceptual approach will be determined for various mission options to meet mission goals and physiological needs. From this top level approach various LSS options can be evaluated (i.e., Closed Ecological Life Support System (CELSS) payoff point, alternate technology potential, utilization of insitu materials, etc.).

1.2 INTRODUCTION

The advanced mission case studies being examined by the OEXP may require unique life support technology differing from Shuttle or planned for the Space Station program. The life support requirements for Shuttle are satisfied by using open-loop approaches (i.e., stored oxygen, water, and supplies with wastes returned to Earth). The existing technology for

Shuttle is only applicable for small crew sizes and short mission durations because of large consumable requirements. Present Space Station planning includes technology for partial recovery of consumables by physicochemical means, enabling a partially closed life support system. However, this technology is immature and has not been evaluated in a long duration integrated test program.

Future manned planetary missions generate new life support requirements that must be considered in addition to those which drove the design of the Shuttle system and are shaping the technology planned for the Space Station. Most significant of these include increased mission complexity, high system reliability associated with long missions and abort times, and the inability to resupply consumables or expendables rapidly. This combination of requirements will dictate a life support system that is characterized by a higher degree of closure, high reliability, increased automation, and independence from terrestrial resources.

1.3 FY88 STUDY RESULTS

The life support special assessment activity was started in June 1988 by conducting a preliminary Shuttle and Space Station technology assessment. This study was performed with the assistance of the Langley Research Center using their Environmental Control and Life Support System (ECLSS) data base and computer aided life support analysis program. The study examined LSS functions only and was not intended to optimize a life support approach for mission case studies or estimate the total LSS weight or volume for these missions.

1.3.1 Objective

The objective of this SAA preliminary study was to estimate the impact of using Shuttle open-loop and Space Station regenerable life support technology (as presently baselined by MSFC) to accomplish the proposed four OEXP mission case studies. This study was conducted in response to a Code Z request. The results reflect the estimates necessary to maintain the basic life support functions of air revitalization, water reclamation, waste management, and provision for food and clothing. A full-up vehicle or habitat LSS would need to include safe haven considerations, stored gases for leakage and repressurization, interconnecting ducts and plumbing, tankage, etc.

1.3.2 Assumptions

a. Only the life support functions of carbon dioxide control, metabolic oxygen generation, potable and hygiene water supply, and waste management are considered.

b. Food and clothing are included for consumable estimates and influence upon the overall water balance and requirements.

c. The Shuttle and Space Station technology options considered will meet the environmental requirements for human expeditions to Mars and the Moon.

d. The Space Station regenerable technology performance capability remains consistent with present estimates.

e. The Space Station regenerable technology will meet the long term operational requirements for reliability and minimal maintenance.

f. Biological life support is not included because of small crew sizes and immaturity of the existing data base to adequately represent this technology.

g. The study results reflect LSS function only and not a complete integrated LSS system. For example, water tankage, ducts, plumbing, fire suppression, spares, and safe haven considerations are not included.

h. Space Station regenerable technology (as presently baselined by MSFC) was assumed for all case studies except humans to Mars (surface) and lunar observations.

i. Shuttle open-loop technology was assumed for humans to Mars (surface) and lunar observations due to small crew sizes and short surface stay times.

1.3.3 Mission Case Studies

Shuttle and Space Station LSS technology was applied to the following basic OEXP mission cases:

a. Human expedition to Phobos.

b. Human expedition to Mars.

c. Human tended lunar observatories.

d. Lunar outpost to early Mars outpost.

1.3.4 Discussion

Table 1 shows the mission case study top level requirements pertaining to LSS (crew size, duration, number of flights) used as a basis for this study. Table 2 is an overview of the technology to accomplish each mission or mission phase. As shown, the relatively short missions (humans to Mars surface and lunar observatories) make use of open-loop Shuttle technology. Also disposable clothing is used for short missions to eliminate the extra water and hardware necessary for washing. Table 3 summarizes the study results to show LSS impacts on the OEXP mission case studies. Tables 4 through 9 provide more detail for each case study including subsystem weights and performance efficiencies. The lower process efficiencies in the water reclamation system directly relates to required makeup water which is a significant portion of the total mission weight. The subsystem spares reflect estimates based upon the technology and associated lower replaceable units. The consumables represent an estimate of food, clothing, and expendables necessary to maintain system operation.

1.3.5 Conclusion

The study results show only one approach to accomplish life support for the

OEXP mission case studies. Shuttle and Space Station technology is theoretically possible if associated penalties can be justified. However, an integrated LSS using Space Station technology has not been proven to date. Once integration occurs, estimates for regenerable Station LSS technology performance could change, impacting reliability or performance predictions.

The major weight penalties are associated with consumables (including food) and make up water. Improvements in water reclamation technology and system efficiencies to reduce consumables ofter high opportunity for LSS optimization.

1.4 FY89 PLANS

A two-phase program (figure 1) was started in September 1988 with the Boeing Aerospace Company. Phase I, which will be completed in FY89, will generate life support requirements based on detailed mission case study characteristics and potential interfacing system influences. The preliminary Shuttle and Space Station technology study completed in FY88 will be improved to include integration and total habitat or vehicle LSS impacts. This technology will be used as a starting point for examination of LSS requirements for future missions. Phase II, which will be conducted in FY90 will conceptualize a basic LSS approach to satisfy requirements identified in Phase I and identify technology requirements.

The Phase I activity involves indepth study of four Office of Exploration (OEXP) mission case studies with emphasis on possible variations within each mission which will influence the LSS. A comprehensive literature review will be conducted of prior and current studies on the four advanced planetary manned missions. The survey will review all related NASA, industry, and other available publications on the four missions. An informational data base for the literature search will be compiled to characterize the four primary mission types including subgroups within each mission type by the key parameters tht fundamentally drive and define an individual mission. Examples of these parameters include, but are not limited to, overall mission purpose, environmental conditions (gravity, atmosphere, and radiation); mission timelines including launch opportunities, length, significant events, EVA events, and resupply periods; crew mix; and abort options.

The various planetary base/spacecraft systems and subsystems and their options that are being considered for long-duration planetary missions will be examined. This planetary base/spacecraft systems data base will be used to describe what the various system options are, and what factors govern the choice of one option over another. These options will be identified for all mission subsystems except ECLSS requirements. An understanding of base/spacecraft subsystems will significantly help in the Phase II design of an LSS that is completely compatible with the rest of the vehicle. Examples of the various planetary base/spacecraft systems and their options will include, but are not limited to the following: (1) electrical power systems including nuclear, photovoltaic, solar thermal, isotope power, fuel cells, and open-loop conversion of planetary resources; (2) propulsion systems including concepts in the areas of chemical, nuclear fission (thermal or electric conversion), chemical boost with upper stage concepts, and pulsed nuclear fusion or antiproton concepts; and (3) thermal control systems including cryogenic systems and heat rejection radiators.

Information from the previous two subtasks will be integrated to identify the most reasonable planetary base/spacecraft systems, excluding life support, to accomplish each of the identified case studies for each primary mission. Rationale will be provided to support system selection to accomplish each unique mission case study. As part of the rationale, mission segment performance analysis will be performed and mission/system parametric models will be constructed. The output of this subtask provides mission variable data for task 2.0 life support requirements and task 3.0 mission case study recommendations.

The intent of the life support requirements identification task (Phase I, task 2.0) is to establish vehicle and base LSS requirements for each of the four mission case studies. Task inputs are from mission case study refinement and from standards such as NASA-STD-3000. Inherent human life support requirements will be examined and compared to mission variables and mission system variables to determine if there are any variations in what are normally assumed to be constant minimum and maximum values. It is anticipated that there are a few such requirements to be discovered. For example, the transit mission requires more crew extra exercises than planetary missions and therefore more crew showers following the exercise. The results of this first evaluation will then feed into the examination of life support system requirements. Life support system requirements and constraints will be treated in two general categories: (1) those that are generic to basic life support functions (i.e., air revitalization, water reclamation, and waste management) and (2) those that are influenced by the mission and related mission systems, either directly or indirectly. These categories will be compared to determine any new mission-specific LSS requirements.

Mission case study analysis will take place in Phase I task 3.0. The LSS requirements resulting from task 2.0 will be evaluated and used to generate the most applicable top-level LSS approach for the four mission case studies. The "top-level" approach will consider criteria such as degree of closure, use of terrestrial resources, and physical-chemical versus biological approaches. Requirements trades and sensitivity analysis will be conducted to fine tune or focus life support requirements. The LSS technology challenges offered by each case study will be considered. These four case studies will form a basis for the Phase II, Life Support System Configuration Concept. LSS requirements for each mission will be developed and an Advanced Mission Life Support Requirements Definition Summary Report will be provided upon the completion of this task, concluding Phase I of this program.

In Phase II the life support technologies required for each mission LSS will be examined in task 1 and the LSS conceptual configuration will be developed in task 2. An understanding of the base/spacecraft LSS requirements developed during Phase I will assist with identification of technologies that are compatible with the rest of the vehicle. Viable LSS technology alternatives will be identified using the Phase I mission objectives anbd the LSS requirements as search criteria. For example, technologies for reducing carbon dioxide to carbon and water will be investigated. Trade studies on those alternatives will then be performed. These trades will consider how well the alternative technologies contribute to subsystem/process level requirements. For example, Sabatier, Bosch, and plants can be compared as carbon dioxide control technologies, as to their relative effectiveness at reducing the required amount of CO_2, and an assessment made based upon their impact on the entire LSS mass and energy balances. Finally, identification of the LSS requirements that need further technology development will take place. These will include the most promising candidates from both the technology trades and the system level impact assessments. A systematic approach will be used in performing this task to ensure that mission and life support requirements are consistent, correlatable, and traceable throughout the documentation. This data base will result in system level configurations for each mission case study. LSS conceptual configuration data for near term development and LSS technology needing additional development will be defined and documented.

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Table 1

MISSION CASE STUDY REQUIREMENTS

Parameter	Units	Human Expedition to Phobos	Human Expedition to Mars	Lunar Observa-tories	Lunar Outpost-Early Mars Outpost	
					Lunar Portion	Mars Portion
Crew Size	Qty	4	8 (4 to Mars Surface)	4	4	8
Crew Total Sortie Time	Days	420	420	20	365	1350
Surface Stay Time	Days	30	20	14	365	730
EVA	Qty Per Mission	4	4 at Moon 10 at Mars 5 Unpress Rover	Twelve-6 hrs Each (10km Unpress Rover Traverse)	10 km Unpress. Rover Traverse	10 km, 100 km Rover Traverse
Resupply Interval	Days	420	420	365	365	1350
Flights	Qty	1 Cargo 1 Crew	3 Cargo 3 Crew	2 Cargo 2 Crew	NA	3 Cargo 3 Crew

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Table 2

MISSION CASE STUDY

ECLSS Concepts

Parameter	Human to Phobos	Human to Mars		Lunar Observatories	Lunar Outpost-to-Mars Outpost	
		Transport	Surface		Lunar Portion	Mars Portion
Crew Size, Qty	4	8	4	4	4	8
Mission Duration, Days	420	420	20	20	365	1350
Surface Stay Time	30	-	20	14	365	730
Clothes	Reusable	Reusable	Disposable	Disposable	Reusable	Reusable
Metabolic Oxygen Supply	4-Bed MS, Bosch, SFWE	4-Bed MS,Bosch, SFWE	LIOH, CRYO O_2	LIOH, CRYO O_2	4-Bed MS, Bosch, SFWE	4-Bed MS, Bosch, SFWE
Water Supply	Times, RO, MF	Times, RO, MF	Stored	Stored	Times, RO, MF	Times, RO, MF
Food	Stored	Stored	Stored	Stored	Stored	Stored
Trash/Waste Management	Compactor, Vacuum Dryer	Compactor, Vacuum Dryer	Compactor, Vacuum Dryer	Compactor, Vacuum Dryer	Compactor, Vacuum Dryer	Compactor, Vacuum Dryer

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Table 3

FY88 LIFE SUPPORT SYSTEM STUDY RESULTS

Utilization of Space Station and Shuttle Technologies to Accomplish Mission Case Studies

MISSION (Technology)	Weight (kg/lbs) per flt					Launch Volume (ft^3/flt)	Power (kw)
	Systems[1]	System Spares[2]	Consum-ables[3]	Water Make-up[4]	Total Launch		
Human to Phobos (Station)	964/ 2125	277/ 610	3097/ 6829	4203/ 9268	8541/ 18833	1034	4
Human to Mars							
Transport (Station)	1927/ 4251	553/ 1220	6195/ 13659	8406/ 18535	17081/ 37665	2069	8
Surface (Shuttle)	574/ 1265	3/6	1955/ 4312		2532/ 5583	250	1.3
Lunar Observatories (Shuttle)	574/ 1265	3/6	1955/ 4312		2532/ 5583	250	1.3
Lunar Outpost-to-Mars Outpost							
Lunar Portion (Station)	964/ 2125	240/ 530	2692/ 5935	3652/ 8054	7548/ 16644	922	4
Mars Portion (Station)	1928/ 4251	1778/ 3920	19911/ 43904	27019/ 59578	50636/ 111653	5869	8

NOTES:

1 Single subsystem only: (no redundancy, tankage, atmosphere makeup, plumbing, etc.) to accomplish functions of air revitalization, water and waste management - not a complete LSS ship set.

2 Spares estimate to maintain subsystem operation for mission duration.

3 Includes food and consumables necessary to maintain system operation.

4 Includes water and tankage to make up for subsystem inefficiency.

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Table 4

HUMAN EXPEDITION TO PHOBOS

Life Support System Spares and Consumables

Function/ Technology	Efficiency %	Subsystem			Water Makeup lbs	Total lbs
		Weight lbs	Spares lbs	Consum-ables lbs		
Air Revitalization						
Molecular Sieve (4-bed)	-	400.6	107.3	6.1	N/A	514.0
Bosch CO_2 Reduction	-	365.6	46.7	275.3	N/A	687.6
Static Feed Water Electrolysis	-	102.0	42.0	9.3	N/A	153.3
Water Reclamation						
Times (urine-flush)	93	192.3	98.0	40.6	669.6	1000.5
Multifiltration (condensate)	99	12.2	7.1	71.3	152.4	243.0
Reverse Osmosis (wash)	90	79.0	197.7	593.0	8445.8	9315.5
Food						
Conventional Food	-	290.6	42.0	5814.7	N/A	6147.3
Waste Management						
Vacuum Drying	-	260.0	46.7	0.0	-	306.7
Compaction	-	144.0	9.3	18.7		172.0
Clothing						
Reusable	-	279.1	13.1	0.5	-	292.7
Total		2125	610	6829	9268	18833

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Table 5

HUMAN EXPEDITION TO MARS - TRANSPORT

Life Support System Spares and Consumables

Function/ Technology	Efficiency %	Subsystem			Water Makeup lbs	Total lbs
		Weight lbs	Spares lbs	Consum-ables lbs		
Air Revitalization						
Molecular Sieve (4-bed)	-	801.2	214.7	12.1	N/A	1028.0
Bosch CO_2 Reduction	-	731.2	93.3	550.7	N/A	1375.2
Static Feed Water Electrolysis	-					
Water Reclamation						
Times (urine-flush)	93	384.6	196.0	81.2	1339.1	2000.9
Multifiltration (condensate)	99	24.4	14.2	142.7	304.8	486.1
Reverse Osmosis (wash)	90	158.1	395.3	1186.0	16891.6	18631.0
Food						
Conventional Food	-	581.2	84.0	11629.3	N/A	12294.5
Waste Management						
Vacuum Drying	-	520.0	93.3	0.0	N/A	613.3
Compaction	-	288.0	18.7	37.3	N/A	344.0
Clothing						
Reusable	-	558.2	26.1	0.9	N/A	585.2
Total		4251	1220	13659	18535	37665

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Table 6

HUMAN EXPEDITION TO MARS - SURFACE

Life Support System Spares and Consumables

Function/ Technology	Efficiency %	Subsystem			Water Makeup lbs	Total lbs
		Weight lbs	Spares lbs	Consum-ables lbs		
Air Revitalization						
Lithium Hydroxide	-	37.4	0.1	392.7		430.2
Cryogenic Oxygen	-	445.6	1.1	174.0	-	620.7
Water						
Supply - Stored	-	37.4	0.3	2742.0	-	2779.7
Waste - Stored	-	-1.4	0.3	680.6	-	679.5
Food						
Conventional Food	-	290.6	2.0	276.9	-	569.5
Waste Management						
Vacuum Drying	-	260.0	2.2	0.0	-	262.2
Compaction	-	144.0	0.4	0.9	-	145.3
Clothing						
Disposable	-	50.0	0.0	45.9	-	95.9
Total		1265	6	4312	-	5583

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Table 7

HUMAN TENDED LUNAR OBSERVATORIES

Life Support System Spares and Consumables

Function/ Technology	Efficiency %	Subsystem			Water Makeup lbs	Total lbs
		Weight lbs	Spares lbs	Consum-ables lbs		
Air Revitalization						
Lithium Hydroxide	-	37.4	0.1	392.7	-	430.2
Cryogenic Oxygen	-	445.6	1.1	174.0		620.7
Water						
Supply - Stored	-	37.4	0.3	2742.0	-	2779.7
Waste - Stored	-	-1.4	0.3	680.6	-	679.5
Food						
Conventional Food	-	290.6	2.0	276.9	-	569.5
Waste Management						
Vacuum Drying	-	260.0	2.2	0.0	-	262.2
Compaction	-	144.0	0.4	0.9	-	145.3
Clothing						
Disposable	-	50.0	0.0	45.9	-	95.9
Total		1265	6	4312	-	5583

Table 8

LUNAR OUTPOST TO EARLY MARS OUTPOST - LUNAR PORTION
Life Support System Spares and Consumables

Function/ Technology	Efficiency %	Subsystem			Water Makeup lbs	Total lbs
		Weight lbs	Spares lbs	Consum- ables lbs		
Air Revitalization						
Molecular Sieve (4-bed)	-	400.6	93.3	5.3	-	499.2
Bosch CO2 Reduction	-	365.6	40.6	239.3	-	645.5
Static Feed Water Electrolysis	-	102.0	36.5	8.1	-	146.6
Water Reclamation						
Times (urine-flush)	93	192.3	85.2	35.3	581.9	894.7
Multifiltration (condensate)	99	12.2	6.2	62.0	132.4	212.8
Reverse Osmosis (wash)	90	79.0	171.8	515.0	7339.8	8106.0
Food						
Conventional Food	-	290.6	36.5	5053.2	-	5380.3
Waste Management						
Vacuum Drying	-	260.0	40.6	0.0	-	300.6
Compaction	-	144.0	8.1	16.2	-	168.3
Clothing						
Reusable	-	279.1	11.4	0.4	-	290.9
Total		2125	530	5935	8054	16644

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Table 9

LUNAR OUTPOST TO EARLY MARS OUTPOST - MARS PORTION
Life Support System Spares and Consumables

Function/ Technology	Efficiency %	Subsystem			Water Makeup lbs	Total lbs
		Weight lbs	Spares lbs	Consum-ables lbs		
Air Revitalization						
Molecular Sieve (4-bed)	-	801.2	690.0	39.0	-	1530.2
Bosch CO2 Reduction	-	731.2	300.0	1770.0	-	2801.2
Static Feed Water Electrolysis	-	204.0	270.0	60.0	-	534.0
Water Reclamation						
Times (urine-flush)	93	384.6	630.0	261.0	4304.3	5579.9
Multifiltration (condensate)	99	24.5	45.9	458.5	979.6	1508.5
Reverse Osmosis (wash)	90	158.1	1270.7	3812.2	54294.3	59535.3
Food						
Conventional Food	-	581.2	270.0	37380.0	-	38231.2
Waste Management						
Vacuum Drying	-	520.0	300.0	0.0	-	820.0
Compaction	-	288.0	60.0	120.0	-	468.0
Clothing						
Reusable	-	558.2	84.0	3.0	-	645.2
Total		4251	3920	43904	59578	111,653

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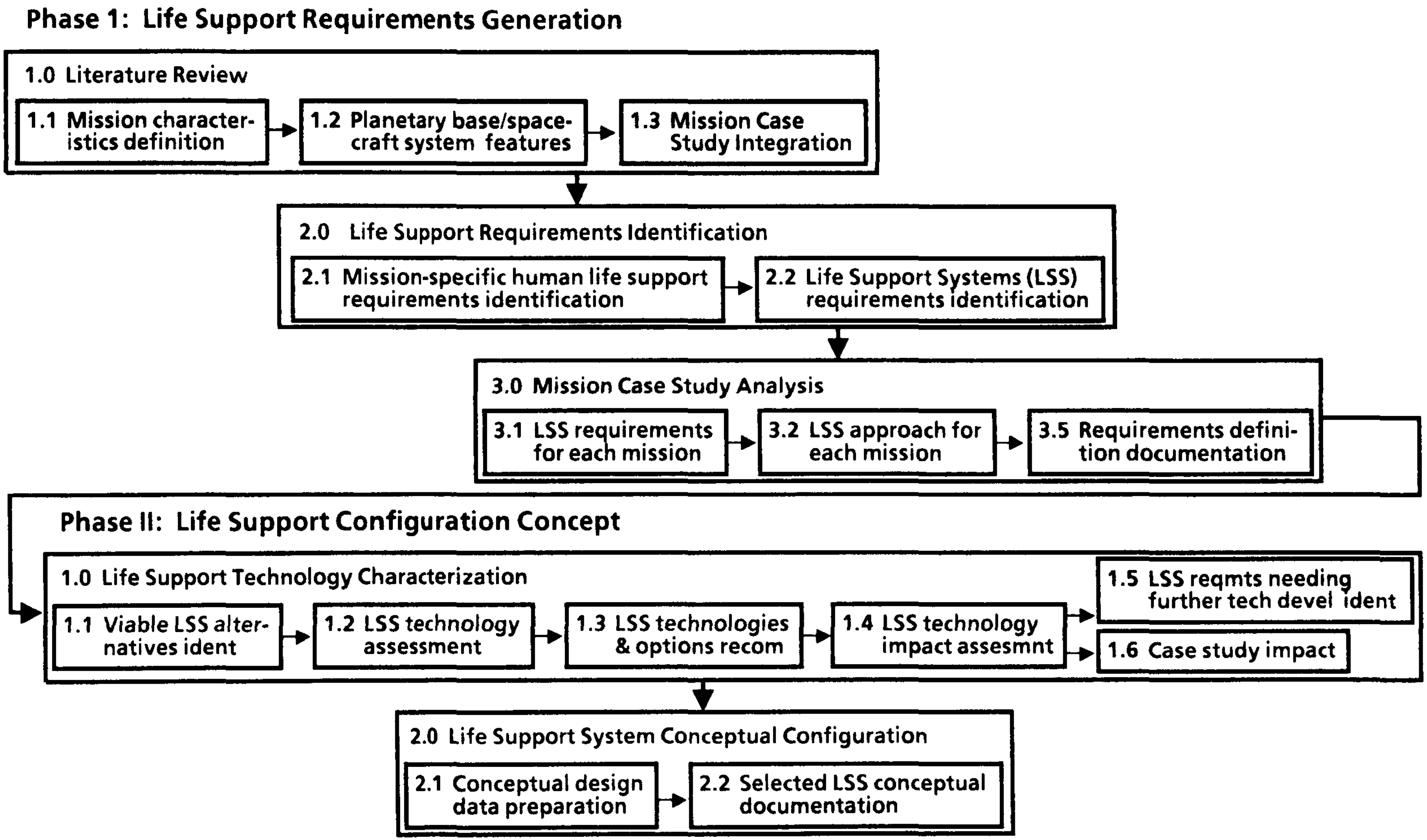

Figure 1

PROGRAM FLOW

Mission Case Study Life Support Conceptual Approach

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2.0 EXTRAVEHICULAR ACTIVITY SYSTEM REQUIREMENTS DEFINITION

2.1 OBJECTIVE

The objective of this study is to identify technology requirements for the Extravehicular Activity System (EVAS) necessary to compliment future Lunar and Mars missions.

2.2 METHODOLOGY

2.2.1 Background

Extravehicular activity capability will be necessary to accomplish the goals of future manned missions to the Moon and Mars. The extent of this capability could vary widely depending upon specific mission objectives. In synergism with the previous Space Station and geosynchronous orbit EVAS definition studies, this work will examine the Office of Exploration (OEXP) case study missions in detail to determine EVAS requirements. Results will provide planners with data to define the extravehicular portion of future missions and to identify necessary EVAS technology requirements to support those activities.

2.2.2 Key Assumptions/Groundrules

a. The study will examine and consider the entire extravehicular activity operation including suits, vehicles, special equipment, and airlocks. This broad assessment is necessary to insure compatibility between equipment in both physical interfaces and operational requirements. Not only must the mechanical, electrical, and software interfaces match, but the operational concepts upon which the design of each piece of equipment is based must be compatible. It does little good, for instance, to have an extended range crew transfer vehicle if sufficient life support is not available to keep the crew alive during the traverse. The broad assessment is also necessary to insure that the proper equipment and, especially, the proper mixture of equipment is provided to accomplish the desired tasks. Each piece of equipment is part of the complete picture required for accomplishing a given mission.

b. Initial results will be top level due to the speculative and conjectural nature of Lunar and Martian mission planning. General types of missions have been defined and overall characteristics of such missions identified, but it is not expected that any specific missions will be determined in enough detail in the near term to allow EVAS requirements definition based on exact task requirements. Instead, generic task requirements will be determined and these will guide the study with EVAS alternatives being specified for each class of capabilities. For instance, if one possible task is to move 3000 kg mass modules from point to point within a 5 meter radius, then a corresponding EVAS alternative would be to provide a crane capable of such performance. Usually, tasks can be grouped to yield classes of capabilities required and, therefore, classes of equipment required. Trade studies will then be implemented to determine the desired alternative from the overall class. Refinements will be made as missions become more defined.

c. The EVAS is considered a support function to the mission. Therefore, the EVAS should impose as few limits or constraints upon crewmembers and mission objectives as practical. As noted above, it is currently not possible to define EVAS requirements based on precise mission specification. It is possible, however, to specify that the greatest capability for work shall be striven for. In other words, the EVAS, as much as possible, shall allow the crew to perform to the maximum extent of their abilities and shall enhance rather than limit those abilities. Following this approach allows specifics of the EVAS to be determined even though exact missions are not known. And where an unusually high cost is associated with a certain capability, alternate capability/technology can be determined and the associated costs estimated to provide planners with data for decision making.

d. The Apollo Lunar surface experience provides a data base from which to define generic Lunar EVAS operational requirements. Thus, Apollo experience is directly applicable to the current study and direct extrapolation may be made from it. This is most important since all recent experience has been in EVA in a much different environment, the relatively benign 0-g environment of low Earth orbit. The direct applicability of Apollo data both simplifies the task of advanced Lunar EVAS specification and allows simultaneously more accurate and more ambitious EVAS requirements definition.

e. Commonality of hardware and technology between Space Station, Lunar, and Martian systems will be considered during this study. Differences in gravity and in radiation, thermal, dust, and atmospheric environments will tend to drive each system to a different design, but individual components may be usable in more than one system. Furthermore, if commonality is considered during this study, and beyond this study in the design phases of each system, then savings in design, development, and procurement may result.

2.2.3 Approach

In FY88, prior to definition by OEXP of the advanced mission scenarios, two independent studies, one by A. D. Little and one by the Essex Corporation, were initiated to examine top level EVAS requirements for typical Lunar and Mars missions. Results from these efforts have and will be used as starting points for further detailed examination to determine EVAS requirements for the OEXP mission case studies. A description of the approach is given below.

a. Review and critique Lunar/Mars mission studies. The final A. D. Little Lunar EVAS study and the interim Essex Corporation Lunar EVAS study have been reviewed and findings to date are based on them. The final Essex study will be delivered shortly. Each study was read and the data, results, and conclusions were examined in the light of the team's expertise in NSTS EVA systems and operations, knowledge of Apollo experience, and direct experience with advanced EVA system studies for the Space Station. The Lunar EVAS studies were found to be sound though some areas necessarily suffered from lack of definition due to paucity of data or immaturity of mission plans. In these cases, trade studies have been identified to allow generation of more detail.

Studies of Mars-system missions are just beginning. As results of these studies begin to become available near the end of the first quarter of FY89, they will be reviewed in a fashion similar to that employed with the Lunar studies.

b. Derive generic mission requirements for the respective EVAS based on the review of mission studies. Each mission can theoretically impose different mission requirements, but a set of generic requirements covering all missions is in most cases derivable. Such a set of requirements was derived for the Lunar EVAS, based on the A. D. Little and Essex studies, and is presented below. The technique employed was to review each mission and define the parameters of an EVAS to support that mission. For each parameter, a boundary value representing the limiting capability for a generic (non-mission-specific) EVAS was selected and this became the generic requirement.

c. For each case study, identify any unique EVAS drivers based on mission, environment, or operational factors. Due to the radically different nature of the missions and especially of the environments, each case study is expected to have unique drivers which will force the designs of the supporting EVAS. Lunar dust and the 0.165-g gravity field are two examples of case-unique drivers which are especially apparent when comparing the Lunar EVAS to that used at the Space Station. Unique drivers for the Lunar mission case have been derived and are presented under "FY88 Results."

d. 4. For each case study, define strawman EVAS performance and design requirements to meet the generic mission requirements and unique drivers. A knowledge of mission requirements and EVAS drivers allows performance and design requirements to be generated as the hardware embodiments of the mission requirements and case-unique drivers. Most top-level performance and design requirements can be generated by inspection while other, more detailed, requirements require further analysis to specify how to implement the requirement in hardware. Carrying the process to its extreme, of course, leads to an exact specification and ultimately to a precise design.

e. Assess currently available EVAS technology and identify areas where technology must be developed in order to meet derived EVAS performance and design requirements. Starting from a familiarity with current Shuttle equipment and the results of Space Station related EVAS studies, status of currently available technology will be assessed and a comparison will be made to requirements for Lunar EVAS. Where deficiencies exist, required development will be identified and possible alternative technologies suggested. Further, high-leverage technologies, that is, those where further development will yield the most proportionate gain, will be identified. These technologies may include areas which are adequately served by current technology but in which valuable gains can be made.

2.3 FY88 RESULTS

Since the Lunar case studies were only recently completed and studies involving the Martian system have just begun, findings are limited to the Lunar EVAS.

The Lunar Extravehicular Activity System can be divided into five categories: pressure envelope, life support system, support vehicles, EVA support equipment, and airlock. Findings, basically generic performance requirements, for each of these areas are discussed below.

2.3.1 Pressure Envelope

The pressure envelope required for the Lunar EVAS will be an anthropomorphic space suit assembly which will allow the suited crewmember, with life support system, to stand, walk, and kneel on the Lunar surface (.165-g). This surface will include flat, dusty regions, rubble strewn regions, inclined crater surfaces with dust or rubble regolith, and broken terrain with dust or rubble regolith. Included among the pressure suit requirements will be the capability to rise after a fall. These capabilities will be arrived at by a combination of flexibility in the suit itself, small size, and low mass in the suit and life support system with proper center of gravity positioning overall. A centered c.g. on a low-mass unit possessing great joint flexibility will allow the crewmember to maintain balance easily and reestablish balance if it is lost.

Maximal dexterity practicable should be provided in the suit gloves through use of special glove designs and minimal suit pressure. Both Apollo and Shuttle experience indicate that the hands are the points of maximal fatigue during an EVA. For multiple EVA missions, much improved gloves over those currently available must be provided if more than three 6 to 8 hour EVA's are required. Otherwise, after 3 or 4 days work, current technology would yield a crewmember unable to perform tasks requiring manual dexterity or strength due to fatigued and injured hands. While current designs for Space Station EMU gloves are much improved over Apollo Lunar surface gloves, the design operating pressure is higher with the result that performance is approximately the same. Since some OEXP missions call for 12 back-to-back 6 hour EVA's on the Lunar surface, current technology is clearly unacceptable. After the first few days, unless tasks and hardware are designed to require minimal hand strength and dexterity, the crewmember would not be able to meet performance goals. These are the clear lessons of Apollo and Shuttle.

The suit should provide protection from the Lunar environment to the crewmember. Its primary purpose, of course, is to protect the crewmember from the Lunar vacuum by providing pressure retention, but it must also provide radiation, thermal, and impact (both micrometeoroid and mechanical) protection. These are all nominal functions of Shuttle and Space Station hardware. The degree of protection from mechanical impact damage may have to be increased, however, due to the additional dangers of falls in a 0.165-g gravity field at unprepared rubble-strewn sites. Even without a loss of balance leading to a fall, operations in Lunar boulder fields will be more dangerous than similar operations on the Shuttle or Station because of the presence of relatively sharp edged protruberances on the boulders. The usual Thermal/Micrometeoroid Garment (TMG) may require reinforcement to provide the additional required protection.

Special attention must be paid to the problem of abrasion and contamination caused by the Lunar dust. The suit must be highly abrasion resistant, easy to clean, rugged, and possessing a long service life. Several measures can be taken to ensure that these requirements are met. First, a coverall can be worn over the suit to isolate it from the Lunar dust both to provide abrasion resistance and to keep seals and bearings clean. This coverall may also double as the TMG. It would preferably be constructed of materials which would not easily retain Lunar dust (Teflon impregnated fabric?) and which could be easily cleaned of whatever dust did adhere. The outer portions of the pressure garment itself can be constructed of such materials to provide a "layered defense" against dust retention. In operation, most dust would be stopped by the coverall which would be cleaned and then removed at the end of the EVA and then the pressure garment would be cleaned before airlock ingress. Proper design of the pressure garment is essential to ensure ease of cleaning. Crevices and seams where dust can accumulate and resist removal should be avoided.

One of the most important areas for consideration of abrasion resistance is that of the helmet visor. Unimpaired vision is a necessity. Resistant materials for visors, coatings, and discardable visor shields should all be used along with provisions for visor cleaning during the course of an EVA to ensure continued good vision for the crewmember.

Seals and bearings on the pressure garment must be designed to resist the invasion of Lunar dust. Labyrinth seals and the use of wipers are two, not exhaustive, methods. The experience of the military in operating high-technology equipment including turbine machinery, helicopters, conventional aircraft and other motor vehicles in desert environments should provide a ready data base on antiabrasion measures.

The pressure suit must allow for sufficient cooling and ventilation. Whether it does this through built-in coolant and ventilation ducts or through the use of a separate garment carrying such ducts (such as the current Liquid Cooling and Ventilation Garment or LCVG) is a design consideration, though built-in ducts provide for easier donning and doffing of the suit. Food and water for an 8 hour nominal EVA must be provided within the suit. While energy bars and a drink bag may be accessed from within the helmet, this requirement probably means that an ability to pull one or both hands into the suit upper torso (hand-in capability) must be allowed by the suit design. This is because simple energy bars will not be sufficient as food over an extended series of EVAs and food and drink handling within the suit will become a necessity. The capability to handle liquid and solid waste products during a nominal 8 hour EVA must also be provided. Liquid waste will consist mostly of urine, though perspiration must be handled by the ventilation apparatus and must not be allowed to build up sufficiently to hazard the crewmember's vision or cause work-limiting discomfort. Vomitus might also need to be handled though this is still problematic and is the subject of a recommended trade study. The requirement to handle solid waste, basically feces but also including such items as food bags and tissue paper, is also part of this trade.

The pressure suit must be sizable at the Lunar base to fit from a 50th percentile female to a 95th percentile male. It is not a necessity that one size suit would fit all EVA crewmembers, but that those size ranges could be accommodated by the hardware at the Lunar base.

The pressure suit must also be maintainable at the Lunar base with no Earth depot maintainence required during its service life. Transportation and logistics costs for transport of suits from the Moon to Earth and back again demand that such trips occur with the least frequency possible. While depot maintenance will no doubt be a necessity from time-to-time, all nominal manitenance and servicing and all other possible maintenance and servicing should be carried out on the Lunar surface by non-specialist crewmembers with appropriate equipment and minimum support from the Earth. This requirement will have a decided impact on the design of the garment since it must be easily maintained and serviced. At the same time, designing the suit to be rugged should drive it to require minimum maintenance since one definition of "rugged" is "requiring little maintenance."

2.3.2 Life Support System

The life support system must provide atmospheric and temperature control for a nominal 8-hour EVA. It must be compact and of low mass, sufficient to allow ease of crewmember mobility without requiring excessive effort to maintain or reestablish center of gravity or balance and to prevent crewmember fatigue. The life support system's functions will include provision of breathing oxygen, control of suit pressurization, removal of exhaled carbon dioxide, and removal of moisture, odors and other contaminants (hair, skin cells, etc.) from the suit environment. It will also maintain pressure suit interior temperatures at acceptable values during a Lunar EVA, whether during the day or night cycle. Each of the above functions will be examined below in some detail.

Breathing oxygen should be stored at a moderately high pressure to minimize bulk of storage tanks. However, increasing storage pressure causes other problems, especially with pure oxygen systems, so that decreasing storage tank mass and volume is not the only consideration. A design trade study should be performed just prior to actual design start to determine the optimum storage pressure.

Control of suit pressurization is fairly straightforward in and of itself, but it has not yet been determined what the suit (or Lunar habitat pressure) will be. Additionally, it has been proposed that a dual or multiple pressure suit be used so that suit pressure could be lowered for tasks requiring large amounts of manual dexterity while preserving higher nominal suit pressure for dysbarism protection. It has also been proposed that work in the suit be viewed as a type of prebreathing, allowing the suit pressure to be lowered gradually during the course of an EVA as nitrogen is flushed from the body. Trade studies examining these questions are currently underway.

Removal of contaminants, particularly exhaled CO_2, from the pressure suit atmosphere is an area requiring much work to drive mass and volume of associated hardware to lower values, agreeable to Lunar EVA. While the straightforward system of particulate filters combined with activated charcoal and Lithium Hydroxide (LiOH) odor and CO_2 scrubbers used in the Shuttle system is reasonably low in weight and mass, it has the distinct disadvantage of requiring heavy logistics support to resupply expended charcoal and LIOH. Regenerable systems are available for performing the atmosphere scrubbing functions but are currently too massive and bulky to be used for the Lunar surface EVAS. Much development work remains in this area. Removal of exhaled moisture and perspiration vapor from the suit atmosphere is also a requirement, but this problem should be handled fairly well by current technology.

The life support system must also provide active thermal control for the suit environment in concert with the passive thermal protection provided for the crewmember by the suit itself. In the Lunar environment of extended day/night cycles and shading terrain features, this thermal control may include heating as well as the cooling found in Shuttle/Station EVA life support systems. The entire area of active thermal control requires extensive further study to force the hardware mass and volume down. Bulky, massive systems such as are currently being proposed for the Space Station LSS are not suitable in the 0.165-g Lunar gravity field. Like the contaminant control system, the thermal control system should use regenerable, rather than expendable technology to avoid unacceptable logistics support requirements, but further research is required to reduce the size of these systems to reasonable values. Crewmember interface will be provided either through the use of a separate garment to carry the heat transfer elements like the current LCVG or through building the elements into the interior of the suit.

Emergency oxygen for breathing and pressurization must be provided to allow a viable atmosphere to be maintained within the pressure suit with a maximum leak rate (to be determined) for a minimum time period (also TBD) to allow for corrective action. The simplest, most reliable system for this purpose is an open circuit blowdown system with a mechanical regulator, similar to the system currently in use in the Shuttle EVAS. This is the recommended technology for the Lunar system. The two drawbacks of this system are, first, it requires oxygen to be stored at very high pressures for conveniently sized tanks and, second, the high pressure oxygen is very sensitive to contamination (such as Lunar dust) so that all internal parts of the system must be kept clean in order to avoid a catastrophic fire. The first drawback requires a high pressure oxygen charging system at the Lunar habitat to service the emergency O_2 system. The second requires the emergency oxygen system to be serviced in a cleanroom or, more likely, a cleanbox environment. Both of these requirements can be met, but further research and innovative design of the Lunar EVAS might relax the need for high pressure oxygen in the emergency system by providing more expedient means of storing the gas.

The life support system must provide power for all suit and life support functions for a nominal 8-hour EVA. Rechargeable batteries have been used in the past and are still the prime candidate, especially considering their reliability and ruggedness. However, fuel cells are also strong candidates because of their power to weight and power to volume advantages.

The life support system must be conveniently serviceable between uses, rugged, reliable, and possess an extended service life. It must be maintainable at the Lunar base with no Earth depot servicing required during its normal operating life. This produces a dilemma. We desire low mass and volume systems but want them to be reliable, rugged, and serviceable, which qualities usually add mass and volume. Innovative design will be called for with the key being to keep the service and maintenance requirements firmly in mind from the start. Further research may yield performance advantages which will allow designers to add the desired reliabilty and serviceability at acceptable mass and volume penalties. Or, research may allow designers to utilize alternative approaches which intrinsicly increase reliability, ruggedness, and serviceability.

2.3.3 Support Vehicles

Ground transport vehicles are required for crew and equipment transfer to sites remote (>2 km) from the Lunar habitat and for exploratory traverses. Such vehicles will increase EVA productivity by decreasing travel time to and from remote sites and allowing longer range traverses than would be possible on foot. They will also allow access to worksites too remote from the habitat for access on foot. These vehicles must be highly reliable to avoid stranding crewmembers away from the Lunar base. This can be obtained through the use of redundancy in the power train and of a simple and reliable suspension design. Each vehicle capable of operating over the horizon from the main base must be equipped with appropriate navigation aids to ensure a safe return and to provide data for pathfinding during exploration. All vehicles will be designed to endure the Lunar day/night cycle and the sudden shifts in temperature (thermal shock) encountered by going from sunlight to shade either in the shadow of a Lunar formation or at dawn and dusk. Unless a protective shelter is always within reach, the vehicles will also need to be hardened against the radiation encountered during solar flares.

Each vehicle should be designed to operate in the Lunar dust environment. First and foremost this means that measures must be taken to supress dust plumes from tires, treads, or feet of all vehicles. Second, all vehicles should be designed to allow for easy cleaning of all surfaces where dust might accumulate and protection and sealing off of all surfaces and equipment where dust cannot be allowed. All bearings should be designed with appropriate seals, should be resistant to damage from accummulated dust and should allow for easy cleaning when these measures are not sufficient. Brakes must be designed to avoid heavy erosion from accummulated dust and new, sealed, brake designs may be necessary. Special attention must also be payed to designing dust-tolerant steering mechanisms and components.

All vehicles must be maintainable on the Moon. None will ever be returned to earth for servicing. This means that top-level components of each vehicle must be designed to interface with EVA crewmembers and their equipment. Second level components must be replaceable in a modular fashion via EVA and must be able to pass through airlock hatches for IVA servicing.

Support vehicles are also required for material handling at the habitat and at remote worksites. Forklift type vehicles, backhoe and/or bulldozer type vehicles and other vehicles specialized to handle task-specific equipment (such as core sample drills, post hole diggers, etc.) are examples of these materials handling vehicles. While materials handling equipment need not be mobile in general, for reasonable utility at a Lunar base such equipment must be self transporting or capable of being easily towed by another vehicle. Several functions (e.g. cargo hauling, digging, and drilling) may be combined on a single chassis.

Currently, several specific types of required support vehicles can be identified. These are the rover, the crane, the truck, the regolith processor, and the bulldozer. Each is discussed in turn below.

The basic rover would be a derivative of the vehicle used on Apollos 16 and 17. It would carry four crewmembers and 200 kg of additional equipment up to 10 km from the main base. Because of terrain limitations and assuming no roads during initial operations, top speed would be limited to approximately 20 kph. Towable trailers could be provided to increase the cargo handling capability of the rover. For extended range, a consumables cart could be towed behind the rover to allow it to make traverses up to 50 km from base. If longer ranges than this are desired, a vehicle with pressurizable cabin, airlock, life support system, food and water for the crew and other habitability provisions such as bunks, waste disposal facilities and the like, as well as extended vehicle consumables, should probably be provided. This is because the extended range combined with a 20 kph top speed would require a long travel time, expending the majority of self-contained life support consumables during the traverse, allowing little time onsite for work. As a minimum, then, supplementary life support would be required for an extended range rover. Preferably, facilities would be provided for the crew to rest after an out traverse or before a return traverse to allow spending a full workday onsite at the work area. Better defined mission requirements will allow the decision to be made on an extended capability rover.

A materials handling crane or a forklift type vehicle is a necessity for removing habitat modules from their lander vehicles and transferring them to transporters for transfer to the base area. Such transport is necessary because of the debris kicked up by landers at the landing area which would damage habitat or other modules if sited in the vicinity. It is desirable to avoid arriving at design solutions too early in the study cycle. However, a crane appears to be the most flexible machine for the job of general materials handling and positioning. With an extendable boom, it can possess a much greater vertical and horizontal reach than a forklift and, in combination with a transporter, it can still transfer objects around the Lunar surface as well as a forklift. Purpose-built transporters can, of course, be built to extract a module from a lander, remove it to the base site, and position it on the Lunar surface at the correct location. Such a purpose-built transporter, however, would probably not be useful for much of anything else and would certainly not have the reach of a crane. While further study of all support vehicles is required, it seems likely, then, that a crane will be included in the equipment of a Lunar base. This crane should be capable of handling a fully equipped habitat module and, as a minimum, should be able to reach to the top of a module covered with its layer of protective Lunar regolith. It must possess all of the Lunar environment tolerance of the rest of the support vehicles, as well as the maintainability and general design for Lunar surface operations.

A truck-like vehicle is required to transport modules from landers to the base-site and larger equipment to the site (perhaps a remote science station) where it will be employed. It is also required for transport of regolith from an excavation to either a dump or use site. Because of this last requirement, the Lunar truck must be especially dust resistant and must provide special protection for its crew from the clouds of Lunar dust likely to be associated with its use. This probably means that a sealed, though not necessarily pressurized cabin will be a must. The truck should be equipped with a standard bed on which adapter modules can be emplaced to tailor the truck to its required function. For instance, support ribs could be fitted to turn the truck into a module transporter while a high-sided hydraulically actuated box structure could be used to form a dump truck. Careful design with actual, not conceptual, missions in mind will be required to minimize the number and type of adapter modules and so relieve the logistics burden of transporting and maintaining them.

Some type of regolith processor is required for producing the protective covering of the habitat necessary to shield against solar flare radiation. This processor could take many forms. It could be a self contained unit which digs the regolith, bags it, and uses a conveyor belt system to dump the bagged regolith on top of the habitat pressure shell to form the shield. It could merely be a bagger which takes basic material dug by the bulldozer and delivered by the truck in its dump truck mode, bags that material, then piles it conveniently for loading into the truck for hauling to the habitat site. Many factors will influence the final design of the processor. For instance, the dust cloud raised by the digging operation may require that it take place some distance from the habitation and any science modules with dust sensitive equipment. This would probably prohibit the use of a self-contained processor, directly transferring its product to the roof of the module to be protected, and instead require some sort of transport system, including a conveyor or similar regolith bag emplacing machinery at the module site. Further study of regolith processors in general is required.

A bulldozer is required for shaping of the Lunar surface prior to siting of the base modules. Whether this is merely leveling or includes excavating for partial burial is still to be determined. A bulldozer blade might be attached to the truck core vehicle and suffice for this function. Extensive excavation will require purpose-built machinery optimized for digging.

2.3.4 EVA Support Equipment

EVA support equipment comprises the generic and specialized tools used by the EVA crew as well ancillary equipment (such as lights and television cameras) and the solar flare shelters used at remote sites.

Tools used by the EVA crew should, as much as possible, be selected from a generic tool kit containing a wide range of tools designed for Lunar EVA use. The Space Station EVA tool kit will provide a starting point for the Lunar kit. Tool design will be modified where necessary or a different tool substituted to ensure that each tool is tolerant of the Lunar dust. Should a unique tool be intolerant of the dust, it will have to be redesigned to allow for easy repeated cleaning. Tool specification will depend on overall Lunar base equipment design. Equipment should be designed such that removal of a module might occur during an EVA but that most work on the module and particularly on dust-sensitive portions of the module would occur IVA after the exterior of the module has been cleaned. Specialized tools for individual tasks should be avoided but may be used when a generic tool capable of performing the task does not exist. Appropriate standards for equipment design, specifying tools available for servicing (among other things), should be published as much prior to that design as possible to promote the use of generic rather than specialized tools. Lunar-specific tools will be added to the genric tool list for tasks unique to the Lunar base. Rock hammers and chisels are two examples. It is also assumed that dust cleaning equipment will be an important part of the generic tool kit.

A generic set of ancillary equipment should be provided to compliment the generic tool kit. This equipment, like all Lunar EVA equipment must be designed to be dust tolerant, have a long service life and be easy to repair, and must tolerate the thermal stresses due to the Lunar day/night cycle. Lights, for illumination in shaded areas and/or to allow continued work during the Lunar night, along with power supplies and supporting/ mounting structures must be provided. The lights must be ruggedized for use by EVA crewmembers. Auxiliary power supplies will be used to power tools and other ancillary equipment. Depending on actual design trades, these supplies may be batteries, fuel cells, or (if claims as to their performance are accurate) the nuclear batteries of Peripheral Systems Inc., in Portland Oregan. Batteries and fuel cells must be capable of being recharged at base camp. Still and motion picture cameras for documentation of work are required along with video cameras and the accompanying recorders or transmitters. These will be especially sensitive to Lunar dust and particular care will have to be taken in their design and in the design of their environment covers to exclude dust. They will also be sensitive to the sudden radical changes in temperature possible on the Lunar surface and must be protected accordingly. Film or tape changing while EVA could be an especially hazardous (to the equipment) operation and will require appropriate care during design. Broad-use sensors and test equipment such as thermography apparatus or electrical test sets will also be included in the generic ancillary equipment. This equipment will have to meet Lunar surface standards.

Solar flare shelters must be provided at worksites far enough from the main Lunar habitat such that assured return to the habitat within the assumed warning time is not guaranteed. These should be permanent shelters since expedient shelters will not provide a sufficient degree of comfort and habitability to guarantee crew health for the worst-case 96-hour shelter requirement. Preferably all shelters would be pressurizable, allowing the EVA crew to doff their suits during the wait for the flare to abate. Barring this, enough room should be provided in the shelter for one crewmember at a time to stand and for all crewmembers to lie in their suits. If it seems at all likely that a 96-hour-plus stay in the pressure suits could seriously become a requirement, then this requirement must be factored into suit design. Provisions for passing supplementary food and water into the pressure envelope interior must be made. Further provisions for handling the wastes of the crewmember must also be made since the nominal waste handling capability will be for an 8-hour EVA. The suit and backpack should be designed so as to allow the crewmember to lie (in Lunar gravity) for extended periods in some relative degree of comfort. These are all necessities for the unpressurized shelter scenario and should not be viewed as mere frills. An enormous physical and mental stress will be placed on the crewmembers otherwise, with a consequently much increased chance of serious bacterial or viral infection arising, a long way from proper hospital care. In the worst case, if these requirements are ignored, the crewmember would be weakened by lack of proper food and water and would be further weakened by lack of proper sleep due to the painful sores rubbed on his back and hips by his attempts at lying down to sleep. He would entrain accumulated liquid and solid waste in these sores, develop a massive infection, and require rescue by other personnel at the end of the solar flare. His survival would be very much in question.

2.3.5 Airlock

The airlock, of course, provides the interface between the Lunar habitat interior and the Lunar surface environment.

The most obvious interface is that between the atmosphere in the habitat and the vacuum on the surface. The airlock should provide passage for the EVA crewmembers to or from the Lunar surface with minimal disturbance to the interior pressure of the habitat. At the same time, minimal air volume must be lost for each cycling of the airlock to reduce logistics requirements for pressurant gases. This can be obtained via two primary methods.

First, a reasonably large percentage of the atmosphere in the airlock can be extracted and stored before equalization valves dump the remaining air to the Lunar vaccuum. The precise percentage to be recovered would depend on logistics to resupply lost air, size of the recovery pump and tank and the logistics surrounding their transport, operation, and maintenance, and time allowed for total depressurization of the airlock.

The second method for obtaining minimal air volume loss is to use a multichamber airlock or two separate airlocks. In this method, crewmembers and smaller equipment would pass to the Lunar surface and back via a small depressurized volume and larger equipment would pass via a larger depressurized volume. In the multichamber lock, only the "manlock" chamber of the device would be depressurized for crew transfer while the entire lock would be depressurized for large equipment passage. In the two airlock method, one lock would be a manlock, the other a large equipment lock. Some discussion has ensued over the possibility of providing an equipment passthrough in order to be able to pass equipment for repair or replacement from the EVA crew to the IVA crew and vice-versa. Since the size of such passed-through equipment has not yet been determined, no recommendation can be made at this moment. Most emphatically, the airlock which the EVA crew is depending on for ingress to the habitation module cannot be used for this purpose since such use would effectively strand the EVA crew on the surface in case of some failure in the airlock mechanism during passage of the equipment. In other words, the crew's airlock should be reserved for the crew and its use to pass equipment should be incidental to the passage of the crew.

The airlock also must provide isolation for the habitat interior from Lunar surface dust. Perhaps in concert with pressure suit coveralls and a possible dust-off porch on its exterior, the airlock will provide apparatus for cleaning the surfaces of the EVA mobility units and for trapping the maximum amount of Lunar dust possible after an EVA. In addition, the airlock must tolerate contamination by the dust and continue to function in spite of it. Seals and mechanisms must tolerate such contamination and filters and dust traps must be incorporated to protect sensitive components. Like the rest of the Lunar base, the airlock must be designed for long life and easy maintenance, with particular attention being payed to the hatch seals which, due to the abrasive nature of Lunar dust, can be expected to be subjected to unusual wear.

2.4 ISSUES/OPEN ITEMS

Each of the EVA system areas, as delineated above, have issues and/or items requiring further study. A list by category is presented in Table 1 and discussed below.

2.4.1 Pressure Envelope

a. The need/utility for a hand-in capability versus technological, design, and operational difficulties of providing such a capability. The ability to pull one or both hands within the upper torso of the suit has obvious utility, especially for scenarios (such as solar flare sheltering) involving extended stay times in the suit, for contingencies (such as vomitimg episodes), and for general utility (such as eating and drinking within the suit). However, it may so unfavorably impact suit design and hinder mobility and dexterity to such a degree as to be counter productive.

b. Mobility/dexterity gain realized with lower suit pressures versus difficulties induced in habitat design because of lower habitat pressure forced by zero prebreathe requirement. While gains in mobility and dexterity universally result from lowering suit pressure, unfavorable impacts are visited on the habitat as a result. Due to oxygen toxicity problems and difficulties and hazards induced when operating in an enriched oxygen environment, a mixed gas (nitrogen-oxygen) environment is a requirement for the habitat. The lower bound of mixed gas habitat pressure is set at 10.2 psi (with a roughly 70/30 nitrogen/oxygen mix), which is at the current flammability limit for oxygen content. Higher oxygen content, and lower total pressure, than this could be arrived at by careful selection of low-flammability materials for the habitat. The current lower limit 10.2 psi habitat allows a zero prebreathe suit pressure of 5.1 psi. A 14.7 psi habitat allows a suit pressure of 8.3 psi. The 60 percent greater pressure causes a large decrease in dexterity in all suit designs and may have proportionate impacts on mobility. However, the higher pressure increases safety both by reducing flammability and by staying well within the bounds of known physiological performance. It also allows use of generally lower cost equipment for the habitat design by making possible the use of off-the-shelf items intended for terrestrial use. For example, cooling fans designed for 14.7 psi service on Earth should function as well at 14.7 psi on the Lunar surface, but might be undersized at 10.2 psi and might require redesign to be safe in the enriched oxygen atmosphere. These and other questions must be studied in order to make a recommendation.

One other possibility which should also be investigated is the use of multiple suit pressures during an EVA. In this scenario, work in the oxygen enriched suit would count as prebreathe time so that after some minimal period suit pressure could be reduced even further. This process could be used on an "as needed" basis for tasks requiring a high order of manual dexterity, or it could be implemented as a nominal and usual reduction in suit pressure over the course of every EVA.

c. Provision of removable coverall for suit/life support system for control of Lunar dust contamination versus design and operational difficulties induced by such a garment and compared to results obtainable with cleaning apparatus at the airlock. Suit mobility and dexterity are already a problem. The addition of another layer of material for dust protection may decrease mobility beyond acceptable limits. Dexterity should not be involved since the coverall would not cover the gloves. Utiltiy of such a coverall needs to be considered since removal of the garment may redistribute dust to the underlying suit and since the cleaning apparatus alone might suffice without a coverall. Some cleaning apparatus is probably a requirement, to clean the coveralls if nothing else, and may manage the entire task of dust control with proper engineering of the surface of the suit's Thermal-Micrometeoroid Garment (TMG). Even so, wear and abrasion of the surface in contact with the dust during the course of an EVA will occur and coveralls can be replaced much more easily than TMG's. Further study is required.

d. Actual requirements for solid waste handling and/or vomitus handling in-suit versus design and operational difficulties induced by such capability and compared to alternate methods of dealing with each problem. Crewmembers on the Lunar surface should be entirely adapted to the reduced-gravity environment and so space adaptation syndrome induced vomiting should not be a problem. Over the long duration of a Lunar mission, however, illness might be a problem and provision for vomitus control could prove useful. The form of such control is problematic. The simplest method would be to employ hand-in capability to allow manipulation and positioning of a containment bag. Hand-in capability, though, may not be possible for the above mentioned reasons. If it is not, the complexity of vomitus collection systems grows enormously and may not be worth providing, considering the likelihood of use.

Solid waste disposal in-suit might have greater utility. Currently on Shuttle, low-residue diets can be employed before the two planned EVA's to ensure that solid waste does not have to be dealt with. Such diets cannot be employed over the extended course of a Lunar mission. However, provision of a complete self-contained waste collection system within the confines of the EVA garment may be impossible. The hand-in capability with a containment bag is probably the best system but, as stated before, may also not be possible. The answer may lay in diet regulation to force timing of evacuation, but this is by no means certain or reliable. This is an extremely important area requiring careful further study.

2.4.2 Life Support System

a. Use of regenerable versus replenishable consumables and impacts on Life Support System mass/volume and Lunar base logistics. Two basic approaches are available in supplying life support consumables. One is to make the system regenerable, the other to make the consumables resuppliable. The sytems of the PLSS where these trades are performed are the thermal system and the CO_2 removal system.

The thermal system study is basically a trade between the increased mass/volume of the selected regenerable system, along with its increased cost, and the cost of consumable replacement for a replenishable system. As an example the STS EMU uses a sublimator, an expendable thermal control system. It is designed to use a maximum of 1.5 lbs/hr of water for each EVA crewmember. This would be, with reasonable EVA overhead, 21 lbs of water per EVA. It should be noted that Shuttle experience has shown the coolant requirements to be somewhat less than this. Lunar requirements for cooling will likely be greater than Shuttle, due to close proximity to the Lunar surface which will act as a heat sink and reflector for solar radiation. Addtionally, heating may be required of the Lunar EVAS for night or shadowed EVA operations. This requirement will have an added mass/volume impact on the PLSS hardware.

The trade involving the CO_2 removal system is similar to that for the thermal system. It involves the increased mass/volume of a potential regenerable system as opposed to the logistics impacts of a replenishable system. As an example the STS EMU uses LiOH cartridges for a CO_2 removal system. These cartridges weigh 7 lbs each and are replaced after each EVA. This LiOH cartridge replacement would mean a logistics impact of 14 lbs per two crewmember EVA. This logistics increase must be weighed against the increase in size and weight of the PLSS and its effect on crewmember mobility.

b. Use of 8 hour versus partial day recharge/replenishment of consumables in Life Support System and impacts on LSS mass/volume, design, operational safety, and Lunar base logistics.

Requirements for an 8 hour PLSS can drive the size of the PLSS to be larger than desirable, yielding decreased mobility for a suited crewmember. A possible alternative is to design a life support system with a less than 8 hour capability that can be replenished during a "work day." This could involve either EVA replenishment while EVA or a "mid-day break" during which the system would be replenished IVA. EVA replenishment could compromise the integrity of the EMU system while the crewmember is operating in the Lunar vacuum. IVA replenishment would require use of extra consumables for airlock repress/depress, and more crew overhead time for the airlock ingress/egress operations. The trade study must weigh these advantages/disadvantages.

c. Requirement/utility of "buddy system" connection between Life Support Subsytems for use in emergencies versus design difficulties and operational safety concerns induced by such a connection.

In the event of a problem with one of the EVA crewmembers EMU (a PLSS fan failure or pressure envelope leak for example), a possible safety feature in the EVA system would be to allow the capability for one EMU to "tie-in" to another until the anomaly can be corrected or a safe location (airlock, habitation module) reached. A drawback to this concept is the necessity of breaking the oxygen pressure enclosure of both EMU's, including the EMU with no intial problem. Design saftey in handling such a procedure and the increased risk to the crewmember in the properly functioning EMU must be carefully considered in the process of making this trade study.

2.4.3 Support Vehicles

a. Capabilities required of support vehicles versus phase of Lunar exploration. Included is possible use of telerobotic vehicles for scouting and for nominal equipment transfer/materials handling. Outcome of such a study is dependent on the assumed missions. In the present case, with missions so uncertain, the best which can be done is to examine capabilities and phasing required for various possible missions and then to note the similarities and differences in those requirements. It might be found, for example, that a crane and a truck are required immediately at mission start for all missions but that some missions require a rover immediately while others do not require it at all, or only require it well after mission start. Knowledge of these requirements will point to technology which will be required for all missions and will identify that technology which must be developed most urgently.

The possible use of robotic or teleoperated vehicles should also be examined. Again, the use of such technology will depend on the assumed mission, so similar techniques to the manned vehicle capability and phasing study, above, can be used, with similar results expected. In fact, the manned vehicle and robotic/teleoperated vehicle capability and phasing studies should be conducted together.

b. Required support vehicle performance (range, top speed, handling, cargo capacity, etc.). In this study, as many details as possible will be fleshed out for the identified support vehicles. A majority of these details should follow naturally from the above study but others can be filled in as well. Top speed of a truck, for instance, may not be a mission-dependent parameter, derivable in the previous study, but instead may depend on assumptions about suspension performance and experience with previous Lunar rovers, i.e., we may know that with current suspension technology as compared with that used on the Apollo Lunar rovers, our top speed over the rough Lunar terrain will be limited to 20 kph.

c. Requirement for manned vehicle cabin pressurization versus design, cost and operational difficulties. The uses of a pressurized cabin on each vehicle will be explored along with its advantages. Cost, in terms of development resources, launch mass, and operating requirements (power, life support expendables, maintenance) will be assessed and characteristics of operations with such a cabin will be delineated. If sufficient data are available, a recommendation for or against pressurizable cabins will be made.

d. Need for ambulance vehicle or towed ambulance module (including pressurizable trauma treatment capability, mobile hyperbaric chamber, etc.) if any, versus cost, logistics, and operational difficulties. Accident statistics for Lunar surface EVA are, fortunately, not available. The approach, then to examining utility of a Lunar ambulance is first to see what such a vehicle might be useful for and second to project the costs in terms of money, resources, and operational impacts of providing it. If it is found (in the extreme case) that all possible injuries in which an ambulance might be useful also result in the crewmember's immediate death due to suit decompression, then obviously no such ambulance need be provided. If it is found, on the other hand, that many types of injuries could result in which a crewmember's life could be saved with and only with an ambulance then it should be seriously considered. Alternately, in the absence of definite statistics, if it is found that provision of an ambulance would severely impact the primary mission then other means of dealing with physical trauma to the crew must be examined. One example of such alternate means is a pressurizable bag or "tent" to enclose an injured crewmember and another attending crewmember to allow emergency treatment. Emergency medical practice on earth will provide the direction for such research.

2.4.4 EVA Support Equipment

a. Minimum requirements for solar flare shelter design including the need for a permanent shelter at each remote worksite incorporating a pressurizable volume versus expedient type shelters requiring crew to spend up to 5 days in pressure suit. A very large portion of this study depends on the characteristics of the proposed Lunar activities and the associated EVA's. Specifically, if all activities are to be carried out within easy reach of the central base, or within easy reach of one or two major installations, then a permanent solar flare shelter can and should be constructed at each such installation. These shelters would be designed to allow the crew to spend the interval waiting for abatement of the flare in some reasonable comfort and hopefully in some form of useful activity, maintaining their productivity. If, on the other hand, activities are so widely spread that easy access to a few central shelters is not possible, then less satisfactory methods of dealing with a threatening solar flare must be investigated. The option of constructing a distributed shelter system, with all of its difficulties and costs should be explored. The alternative of hastily constructing an expedient shelter when needed should also be explored including examining the various types of expedient shelters which have been proposed. Some of these shelters would require crewmembers to spend up to 5 days in their pressure suits, certainly a last-ditch option.

b. Requirements for portable power supplies including performance requirements and basic design approach (i.e., solar, fuel cell, batteries, nuclear). These requirements are, again, dependent on the nature of the mission under consideration and so a portion of the results from this study will consist of requirements for types of power supplies for types of missions. The remainder of the results will consist of an examination of the merits of various types of power supplies and their applicability to supporting Lunar surface EVA. No examination will be made of power supplies for equipment which happens to be deployed via EVA (say, a large optical array and its instrumentation) but only of supplies for EVA tools and support equipment.

2.4.5 Airlock

a. Amount of atmosphere to recover/recycle versus cost and design difficulties and compared to logistics difficulties for pressurant resupply.

Resupply cost of airlock gases (primarily nitrogen and oxygen) in the depress/repress cycle must be weighed against the extra cost in complexity, mass, and maintenance of the lock (pumps, accumulators to store the recovered gases) and the energy required to run the pumps. The trade study must take into account transportation cost from Earth and the possibility of acquiring the required O_2 from the Lunar surface.

b. Volume of airlock/number of EVA crew to accommodate in airlock at once.

The smaller the free depress volume of the lock the less the gases expended when the lock is depressed from any given pressure. This must be weighed against opreational requirements for lock size (number of crew that can use lock at same time, operational volume required per crewmember).

c. Efficacy, utility, and characteristics of dust isolation measures. This refers specifically to those measures employed as part of the airlock system, including those on the exterior of the airlock. This study will be performed in conjunction with the study examining the use of coveralls over the suit (trade #3 under pressure envelope).

d. EVA support functions located in airlock versus dedicating airlock to environment interface (atmosphere, dust, temperature, etc.) only.

EVA support functions involve such things as airlock repress/depress, EMU servicing and checkout, and EMU resize nad maitenance. Locating these functions in the lock eliminates thier impacts on available habitation/laboratory volume. However, it causes extra requirements to be placed on the support hardware (must at least tolerate, in some cases operate at, vacuum) and increases the depress/repress volume of the lock (reference trade study 2). This trade study must weigh the relative importance of these impacts.

2.5 FY89 PLANS

The review of the Lunar mission studies and derivation of generic mission requirements will be completed in FY88. In FY89, unique Lunar EVAS drivers will be identified and strawman EVAS performance and design requirements will be defined. The major vehicle for accomplishing this work will be to perform trade studies to resolve the 19 issues and open items defined above. Current capabilities will be assessed to identify areas where technology must be developed to meet identified requirements.

In FY89, as results of the two Martian system mission studies become available, the above process will be repeated for the Phobos and Mars surface mission scenarios. Completion is expected in late FY89.

TABLE I
LUNAR EVAS ISSUES/OPEN ITEMS
FY88 STUDY

PRESSURE ENVELOPE
- Hand-in Capability
- Suit and Habitat Pressure
- Dust Protection
- Waste Handling

AIRLOCK
- Atmosphere Recovery
- Volume
- Dust Isolation Measures
- EVA Support Function Provided

EVA SUPPORT EQUIPMENT
- Solar Flare Shelter Design
- Portable Power Supply Availability

LIFE SUPPORT SYSTEM
- Regenerable vs Open Loop
- Partial Day Recharge
- Buddy System Umbilical

SUPPORT VEHICLES
- Capabilities (crane, forklift, etc.)
- Performance (speed, range, etc.)
- Mission Phasing of Capabilities/ Performance
- Pressurized Cabin
- Garage Provision

LEO ASSEMBLY OPERATIONS SUPPORT SYSTEMS AND TECHNIQUES

OBJECTIVE:
The objective of this study is to determine the feasibility of operational methods and techniques for low Earth orbit (LEO) assembly. Programmatic operational guidelines will be developed and used to assess proposed space transfer vehicle and assembly node designs received from the Integration Agents (IA). Results from these assessments will be provided to the IA's in an iterative process which will assure the influence of operations considerations on systems design and development. Once the conceptual designs reach an adequate stage of maturity, more detailed analyses will be performed to produce a more mature set of program operations guidelines, recommended crew requirements, operations support equipment needs, and recommended systems design requirements.

These activities will be augmented by the related efforts of other Special Assessment Agents (SAA) whenever appropriate. For example, coordination between the operations and the automation and robotics (A&R) areas has been initiated to effectively utilize manned space operations experience and A&R expertise on the LEO assembly problem. Data exchanges and technical discussions will ensure that these efforts are integrated and complementary.

METHODOLOGY:
A. Background: The assembly of large space transfer vehicles in LEO is an option to support manned solar system exploration. Limitations of the projected Earth-to-orbit (ETO) transportation system dictate that such assembly operations must make highly effective use of equipment, personnel, and resources launched from Earth to support these exploration missions must be effectively assessed and integrated in order to develop the support systems and techniques needed for LEO assembly operations.

Orbital assembly operations capabilities and limitations can be expected to be influenced by the design and utilization of the ETO transportation system, the assembly node facilities, the space transfer vehicles, and interfaces with the ground and other space-based facilities and systems. Due to the iterative nature of this process, assembly operations will also have an impact on these system elements. Assembly operations for Human Expeditions to Mars are representative of assembly operations required by exploration missions using similar technology in route to other solar system destinations. Extrapolation of the results to assembly operations at non-LEO sites in space is also expected to be possible.

B. Key Assumptions: Key assumptions derived from the Scenario Requirements Document (SRD) for Case Study 2 include the following:

•A LEO node is utilized to support assembly operations.

•Multiple manned missions to Mars are to be supported.

•Each mission consists of an unmanned cargo flight and a manned piloted flight.

•All space transfer vehicles are expendable and employ chemical propulsion systems.

•Advanced technology requirements for complex assembly support systems are to be minimized.

C. Approach: The LEO assembly phase of the Human Expeditions to Mars case study will be used for an initial operations assessment. Assembly operations will be a strong function of the design of the launch vehicle systems, the space transfer vehicles, and the supporting transportation node. A set of operational guidelines for LEO assembly will be developed to support an evaluation of the current proposed designs for the appropriate systems as well as the assumptions from which they were generated. Aspects of the design philosophy which negatively impact assembly operations will be identified.

An interactive relationship with the IAs, SAAs, and MASE will be established in order to develop an operationally realistic vehicle/node design and candidate assembly scenarios which will be used to expose more subtle, operationally driven system design requirements. Preliminary data requirements needed to assess the assembly scenarios have been identified and are presented in Table 1. These scenarios will identify candidate tasks for automation and robotics (A&R), establish requirements on the assembly node for the placement and capabilities of manipulator systems, and identify EVA, free-flyer, and other support equipment requirements necessary to accomplish the assembly task. The results of the Case Study 2 analyses will be extrapolated to cover Case Studies 3 and 4. Additionally, Case Study 1 will be assessed to examine the impacts on assembly operations when a LEO node is not employed.

D. Products: Two major products resulted from the fiscal year (FY) 1988 activities.

A preliminary methodology by which additional case study elements may be evaluated from an operations perspective to assess operations support systems and techniques has been developed. This process is iterative, initially supporting operations feasibility assessments of proposed assembly scenarios that eventually lead to the development of preferred operational methods and techniques for LEO assembly.

This methodology is being applied to LEO assembly operations for Case Study 2 and is utilizing ancillary studies such as the Manned Mars Mission Accommodation - Sprint Mission (LaRC). Although only portions of the case study have been examined to date, interim results obtained in FY88 include the following preliminary products: operations task breakdown, assessment criteria/categories and goals, evaluation matrices, implementation alternatives, schedule of activities, assessment tools, and FY89 study candidates.

FINDINGS:
Assembly operations in LEO will be highly dependent on the design and operational capabilities of the systems involved. The size and number of the constituent elements of the vehicles being assembled will have a major impact on the operational techniques employed, as will the amount of integration, verification, and testing required onorbit. The LEO node will strongly influence the character of assembly operations. The amount of functional support this facility provides to both the vehicle being assembled and the assembly process itself will be a major factor in the overall design of the assembly process. In addition to influencing the design of the space transfer vehicle, the capabilities of the ETO transportation system will impact assembly operations by determining the frequency of logistics support flights.

The requirements for the ETO transportation needed to support the Human Expeditions to Mars are very demanding, particularly for operations support. Current studies indicate that a large number of flights, combined with a relatively high flight rate, will be needed to launch the space transfer vehicle assembly elements and the required propellants. The high flight rate is anticipated to impact virtually all major operations phases including launch vehicle processing and cargo integration, flight planning and reconfiguration, launch preparation, and launch and mission support. The utilization of multiple launch systems, such as an unmanned heavy-lift launch vehicle and the manned Space Shuttle, can be expected to introduce additional complexities.

The assembly of the space transfer vehicle will require the development of new operations support systems and techniques. Although advanced technology development requirements are to be minimized, they may be imposed if they hold promise of significant gains in productivity or reduction in mission risk. To date, no onorbit activity has demonstrated the kinds of operations this task will involve. Depending on the design of the vehicle, however, many of the required LEO assembly operations techniques may be developed and demonstrated by the Space Station program.

An area in which the Space Station program has already demonstrated an operational constraint is logistics resupply. Space Station program analyses indicate that assembly activities will be a function, rather than a driver, of the assembly logistics flight rate. ETO transportation program considerations, such as schedule, manifest, and ground logistics capabilities, will control the number and frequency of logistics flights to the LEO node. All onorbit operations will have to conform to this schedule.

The functional requirements on the LEO node may be driven by another schedule. Program schedules for the Human Expeditions to Mars case study show periods of time, between the launch of one vehicle and the beginning of the assembly process on the next, in which no assembly will be taking place. During these periods, operational considerations may preclude a manned presence throughout the functional lifetime of the node. The possibility exists, therefore, that the LEO node might have to function in both a permanently manned and a man-tended mode. This dual capability requirement could have a strong impact on the node systems design.

ISSUES/OPEN ITEMS:
Several open items exist for LEO assembly operations. Before any meaningful, in-depth analyses can be performed, the operations capabilities and limitations of the systems involved must be defined. Minimal information about the ETO transportation and LEO node systems has been made available. In addition, while the high level assembly scenarios developed so far indicate the use of an orbital manuevering vehicle (OMV), it is doubtful that this vehicle, in its current design configuration, will be capable of handling the tasks to which it is being assigned. Therefore, a set of requirements for a more robust free-flyer system may have to be defined.

Another open item which will have a major impact on operations is the location of the LEO support crew base. System functionality requirements will differ significantly depending on whether the assembly crew is based at the Space Station and have to be ferried to the assembly node, at the LEO assembly node facility itself, or in the manned Mars vehicle. If the crew is based at the Space Station or in the Mars vehicle, the LEO node will have to be designed as a man-tended system. Basing the crew at the assembly node itself will require permanently manned capability.

Analysis of the Human Expeditions to Mars case study indicates that a significant number of the required ETO transportation flights will be dedicated to propellant delivery. At this time, the propellant handling/transfer systems and storage location have not been defined. Since the fueling process will have a major impact on the assembly and flight preparation operations, this lack of definition is an important open item.

PLANNED OR REQUIRED FY89 ACTIVITY:

•Continue LEO assembly operations assessment for Case Study 2.
•Orbital assembly operations assessments for other case studies.

TABLE 1. LEO ASSEMBLY OPERATIONS ASSESSMENT INPUTS

ASSEMBLY ELEMENT	PRELIMINARY DATA REQUIREMENTS
LEO NODE	ASSEMBLY SEQUENCE(S) -SCENARIO OVERVIEW: EVENT DESCRIPTION/TIMELINE ASSEMBLY SUPPORT EQUIPMENT (E.G. FREE-FLYERS, MANIPULATORS, EMU, MMU) NODE CONFIGURATION DEFINITION(S) -PHYSICAL LAYOUT -SYSTEMS CAPABILITIES -ORBIT PROPELLANT STORAGE FACILITIES CREW COMPLEMENT -SPACE STATION CREW -LEO SUPPORT (ASSEMBLY) CREW SCHEDULING GUIDELINES
ETO TRANSPORTATION	MANIFEST -CARGO ELEMENT DEFINITION/SEQUENCE -FLIGHT SCHEDULE -FLIGHT SUPPORT EQUIPMENT PERFORMANCE -LIFT CAPABILITY -MARGIN/RESERVES ALLOCATION -ORBITAL OPERATIONS CAPABILITIES (E.G. SURVIVAL LIFETIME, CONTROLLABILITY) PROFILE DESCRIPTION -EVENT TIMELINE
SPACE TRANSFER VEHICLE	VEHICLE CONFIGURATION THROUGHOUT ASSEMBLY ASSEMBLY TASK RESOUCE REQUIREMENT ESTIMATES SERVICE SUPPORT REQUIREMENTS -UTILITIES (E.G. POWER,THERMAL, SYSTEM MONITORING) -MAINTENANCE -ASSEMBLY -PROPELLANT FUELING

LEO ASSEMBLY OPERATIONS
SUPPORT SYSTEMS AND TECHNIQUES

OPERATIONS ASSESSMENTS

OBJECTIVE:

IDENTIFY, ASSESS, AND PROVIDE IMPLEMENTATION RECOMMENDATIONS FOR OPERATIONS SUPPORT SYSTEMS AND TECHNIQUES NEEDED TO EXTEND MAN'S PRESENCE BEYOND LOW EARTH ORBIT

SCOPE:

OPERATIONS SUPPORT SYSTEMS AND TECHNIQUES, AS APPLIED TO THE OEXP CASE STUDIES, INCLUDES ACTIVITIES REQUIRED TO DEVELOP AND UTILIZE CREW AND SUPPORTING PERSONNEL CAPABILITIES FOR PREFLIGHT PLANNING, TRAINING, REAL-TIME FLIGHT CONTROL, IN-SPACE ACTIVITIES, GROUND SUPPORT OPERATIONS, AND HAZARD CONTROL OF SPACE VEHICLES AND SUPPORTING TRANSPORTATION ELEMENTS

APPROACH:

DEVELOP AND VALIDATE AN ITERATIVE METHODOLOGY EMPLOYING INTERACTIVE EXCHANGE WITH OTHER EXPLORATION DISCIPLINES WHICH WILL IDENTIFY AND EVALUATE ALTERNATIVES FOR OPERATIONS SUPPORT SYSTEMS AND TECHNIQUES. UTILIZE THE METHODOLOGY TO DEVELOP INPUTS FOR SUBSEQUENT EXPLORATION IMPLEMENTATION PHASES.

IN-SPACE VEHICLE PROCESSING

IN-SPACE VEHICLE PROCESSING CONSISTS OF THE FOLLOWING ACTIVITIES AND FUNCTIONS: ASSEMBLY, SERVICING, VERIFICATION, AND REFURBISHMENT OF THE VARIOUS COMPONENTS, ELEMENTS AND SYSTEMS OF A SPACECRAFT OR OTHER SPACE MISSION ELEMENT(S).

PROPOSED DEFINITION FOR THE TERM "IN-SPACE VEHICLE PROCESSING", ACTION ITEM NO. ZM-88-7

IN-SPACE ASSEMBLY

IN-SPACE ACTIVITIES AND FUNCTIONS PERFORMED TO PLACE IN ORBIT, TO JOIN AND INTERCONNECT, AND TO ARRANGE IN OPERATIONAL FLIGHT CONFIGURATION THE VARIOUS COMPONENTS, ELEMENTS AND SYSTEMS OF A SPACECRAFT OR OTHER SPACE MISSION ELEMENT(S).

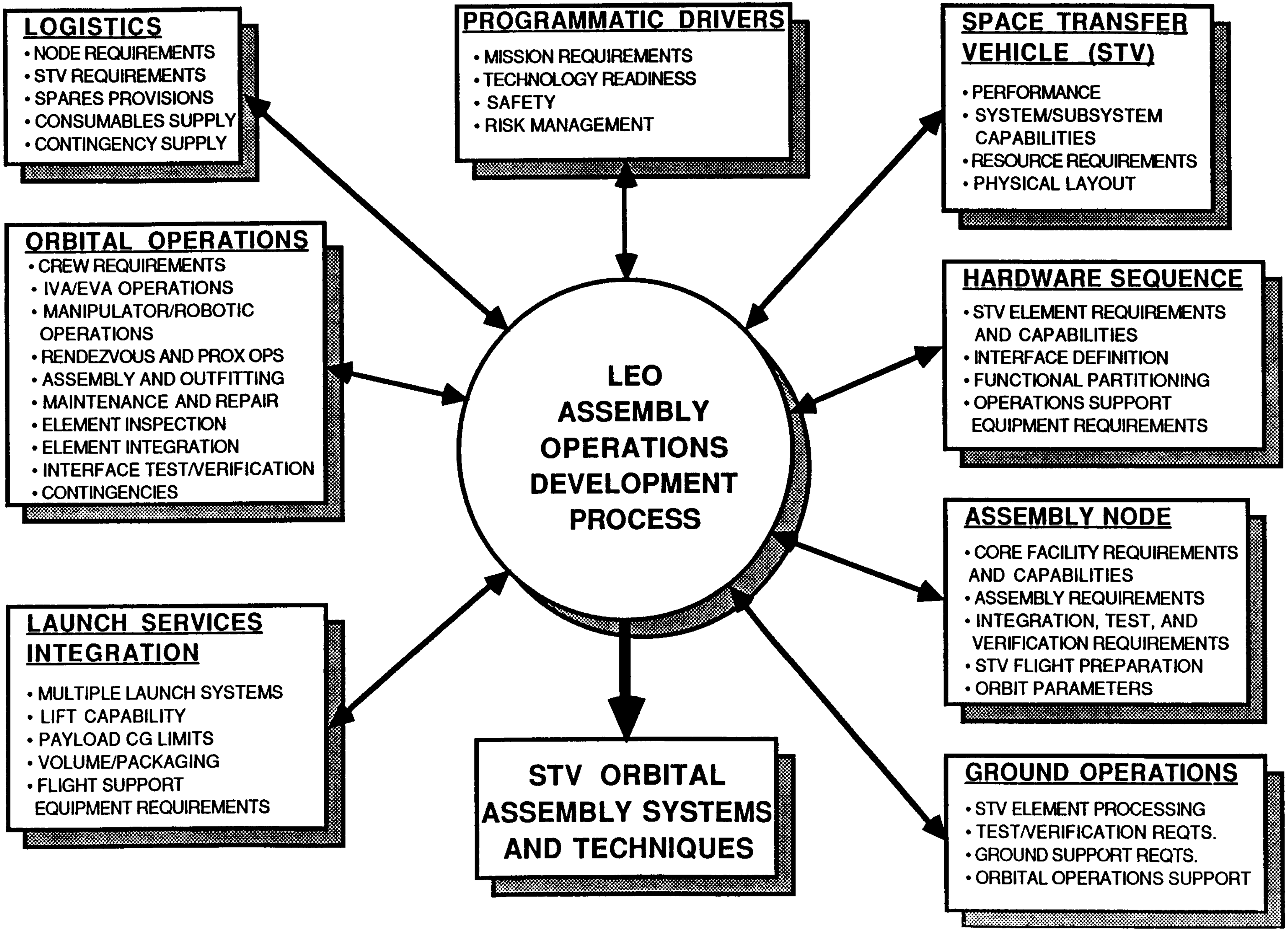
PROPOSED DEFINITION FOR THE TERM "IN-SPACE ASSEMBLY", ACTION ITEM NO. ZM-88-7

METHODOLOGY DEVELOPMENT

LOW EARTH ORBIT ASSEMBLY OPERATIONS SELECTED AS INITIAL AREA FOR DEVELOPMENT AND VERIFICATION OF ASSESSMENT METHODOLOGY

- REPRESENTATIVE OF WIDE RANGE OF OPERATIONAL ACTIVITIES
- REQUIRES INTEGRATION OF MANY COMPLEX/DIVERSE ELEMENTS
  - INTERFACES WITH MOST OTHER AREAS OF INVESTIGATION
- MUST MAKE HIGHLY EFFECTIVE USE OF EQUIPMENT, PERSONNEL, AND RESOURCES LAUNCHED FROM EARTH
  - PROVIDES MECHANISM FOR EARLY IDENTIFICATION OF AREAS NEEDING TECHNOLOGY DEVELOPMENT
- HAS COMMON FEATURES WITH OPERATIONS WHICH ARE CONDUCTED THROUGHOUT ALL CASE STUDIES

MANY COMPLEX AND DIVERSE ELEMENTS MUST BE INTEGRATED



OPERATIONS USED FOR ASSEMBLY TASKS ARE DISTRIBUTED THROUGHOUT THE FOUR CASE STUDIES

CASE STUDY 1 - HUMAN EXPEDITION TO PHOBOS

EARTH LAUNCH → MINIMUM LEO NODE ACTIVITIES → LEO NODE - MARS TRANSIT → PHOBOS/MARS SURFACE ACTIVITIES → MARS - EARTH TRANSIT → EARTH ENTRY AND LANDING

MINIMUM LEO NODE ACTIVITIES:
- o MATING AND CHECKOUT OF VEHICLES

PHOBOS/MARS SURFACE ACTIVITIES:
- o PILOTED SORTIE VEHICLE DEPLOYMENT AND CHECKOUT
- o SAMPLE CANNISTER RETRIEVAL AND STOWAGE

CASE STUDY 2 - HUMAN EXPEDITIONS TO MARS

EARTH LAUNCH → LEO NODE ACTIVITIES → LEO NODE - MARS TRANSIT → MARS ORBIT/MARS SURFACE ACTIVITIES → MARS - EARTH TRANSIT → LEO NODE ACTIVITIES → EARTH RETURN AND LANDING

LEO NODE ACTIVITIES:
- o DOCKING
- o ASSEMBLY, CHECKOUT OF VEHICLES
- o FUELING OPERATIONS

MARS ORBIT/MARS SURFACE ACTIVITIES:
- o SURFACE EVAs
- o ROVER ASSEMBLY, CHECKOUT
- o PHOBOS/DEIMOS SORTIE VEHICLE DEPLOY, CHECKOUT
- o PHOBOS/DEIMOS EVAs

LEO NODE ACTIVITIES:
- o DOCKING

CASE STUDY 3 - LUNAR OBSERVATORIES

EARTH LAUNCH → NODE ACTIVITIES → NODE - LUNAR TRANSIT → LUNAR SURFACE ACTIVITIES → MOON - EARTH TRANSIT

NODE ACTIVITIES:
- o ASSEMBLE, CHECKOUT VEHICLES AND EQUIPMENT
- o TRANSFER PROPELLANT

LUNAR SURFACE ACTIVITIES:
- o CONSTRUCT, SETUP EQUIPMENT
- o LUNAR SURFACE EVAs
- o SETUP HABITAT
- o DEPLOY, CHECKOUT ROVER
- o DEPLOY INSTRUMENTS

CASE STUDY 4 - LUNAR OUTPOST-TO-EARLY MARS OUTPOST

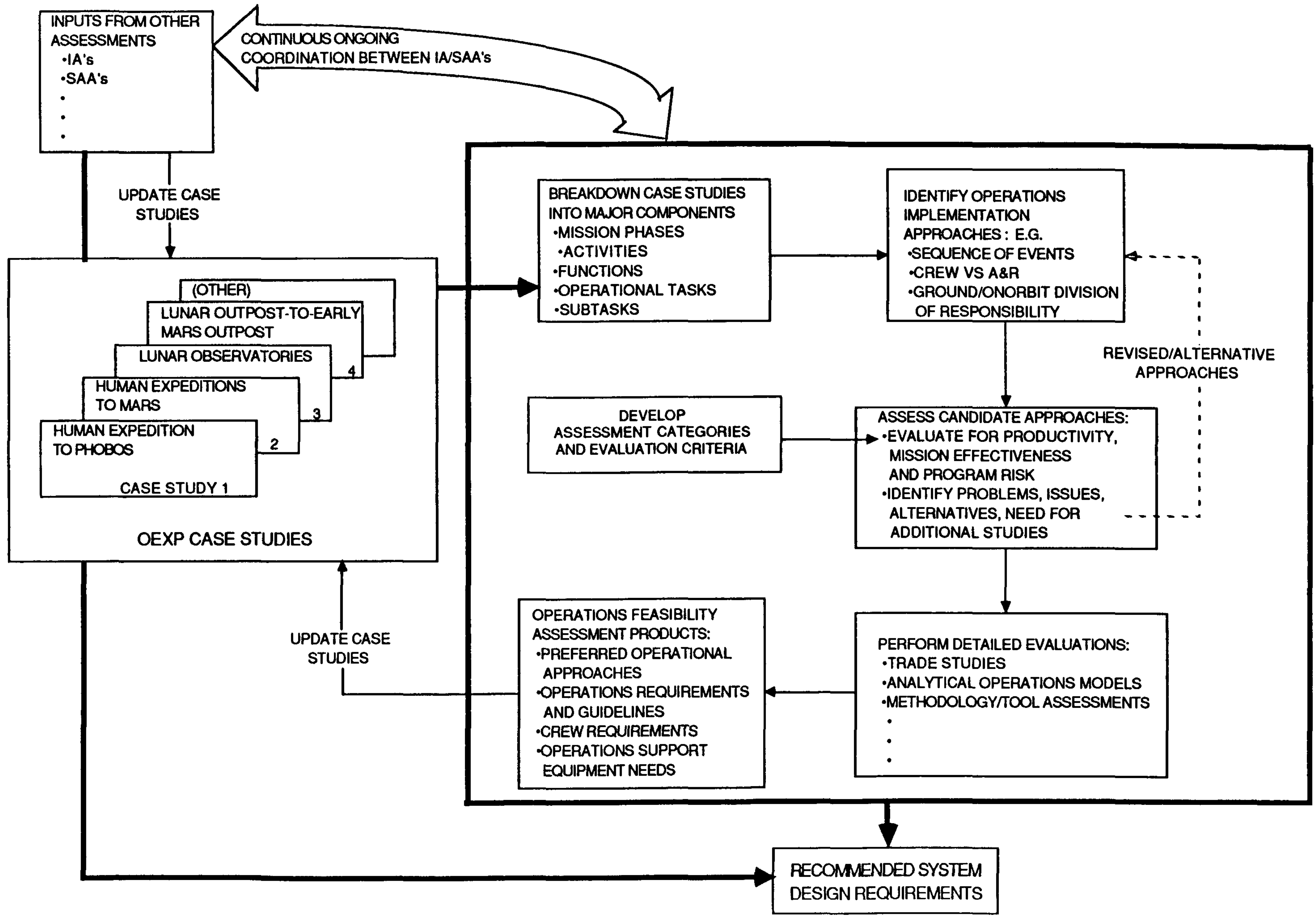
EARTH LAUNCH → NODE ACTIVITIES → NODE - LUNAR TRANSIT → LUNAR SURFACE ACTIVITIES → MOON-MARS TRANSIT → MARS SURFACE ACTIVITIES → RETURN

NODE ACTIVITIES:
- o CONSTRUCT, ASSEMBLE, SERVICE VEHICLES AND PAYLOADS
- o TRANSFER PROPELLANT

LUNAR SURFACE ACTIVITIES:
- o CONSTRUCT, SETUP, OPERATE LLOX PLANTS
- o CONSTRUCT FACILITIES
- o SERVICE, CHECKOUT, FUEL VEHICLES
- o CONDUCT EVAs
- o DEPLOY INSTRUMENTS

MARS SURFACE ACTIVITIES:
- o CONSTRUCT FACILITIES
- o SERVICE, CHECKOUT VEHICLES
- o SETUP PHOBOS LOH PLANTS

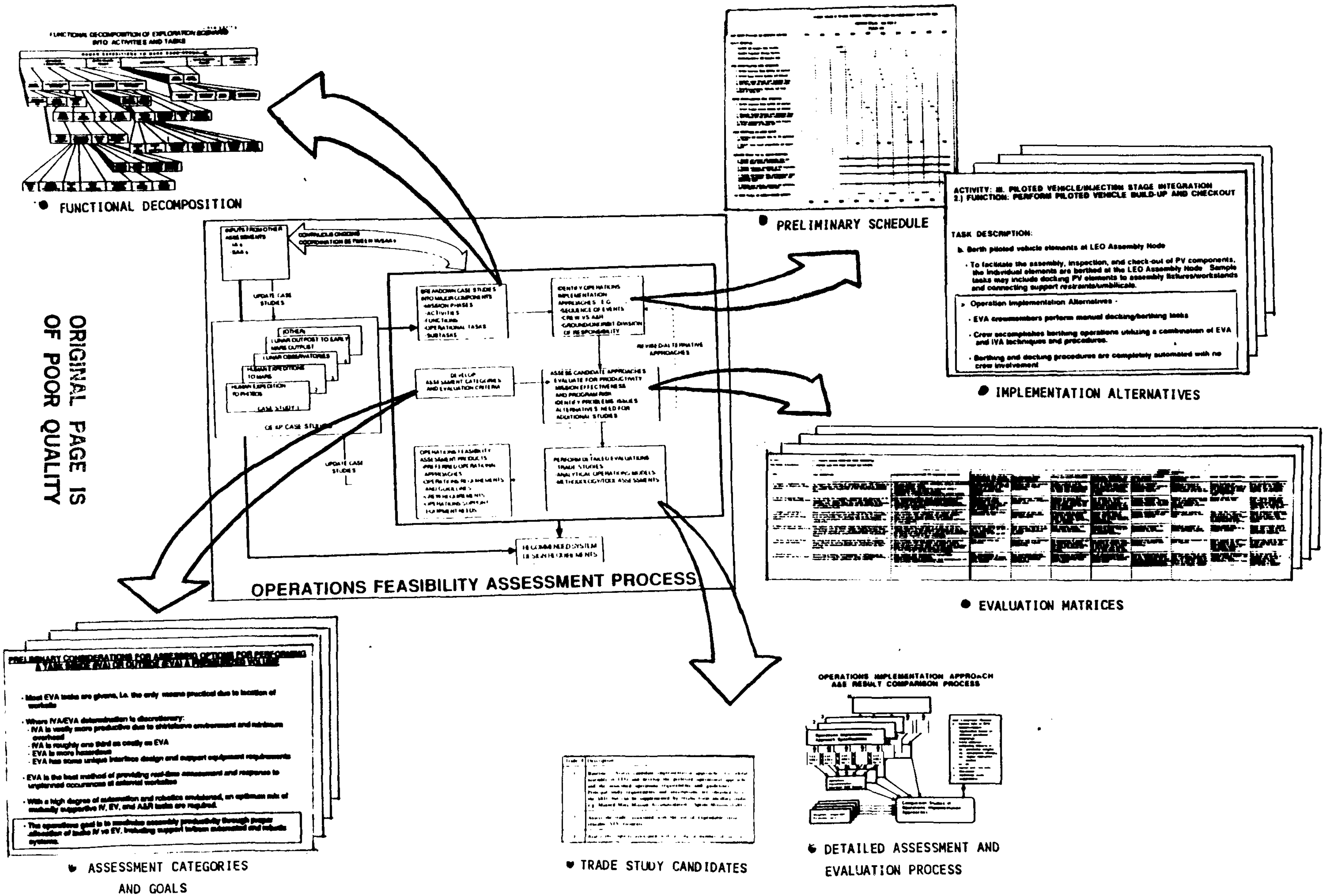
OPERATIONS FEASIBILITY ASSESSMENT PROCESS



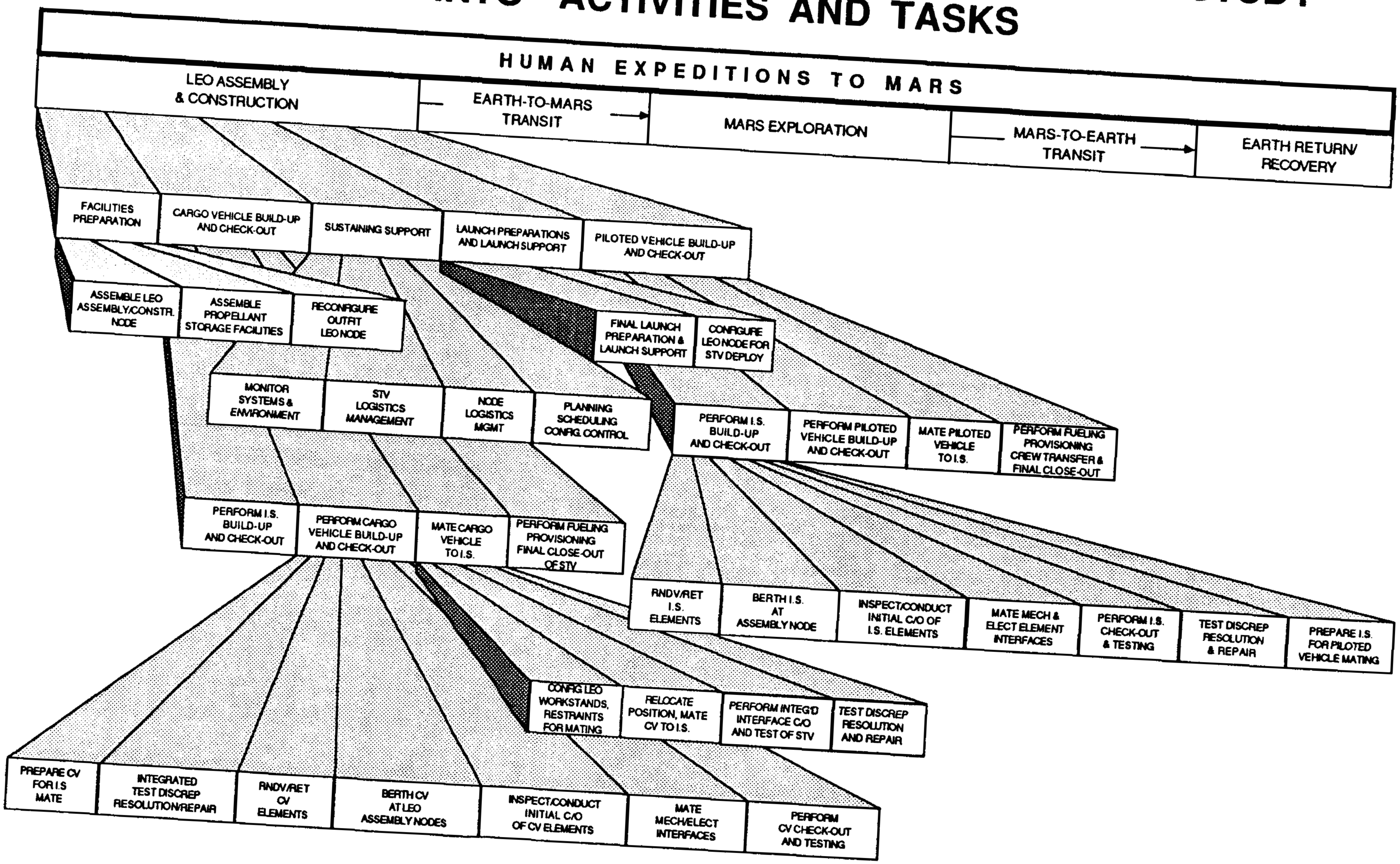
ASSUMPTIONS AND GROUNDRULES

- CASE STUDY 2 GROUNDRULED AS ASSESSMENT BASELINE FOR ASSEMBLY OPERATIONS IN LEO - REFERENCE SRD
- A LEO NODE IS EMPLOYED FOR ASSEMBLY OPERATIONS
- MULTIPLE MANNED EXPEDITIONARY MISSIONS TO MARS ARE PLANNED
- EACH MISSION CONSISTS OF AN UNMANNED CARGO FLIGHT AND A MANNED PILOTED FLIGHT
- ALL SPACE TRANSFER VEHICLES ARE EXPENDABLE AND EMPLOY CHEMICAL PROPULSION SYSTEMS
- ADVANCED TECHNOLOGY DEVELOPMENT REQUIREMENTS ARE TO BE IMPOSED ONLY IF THEY HOLD PROMISE OF SIGNIFICANT GAINS IN PRODUCTIVITY OR REDUCTION IN MISSION RISK

INTERIM PRODUCTS OF THE FEASIBILITY ASSESSMENT PROCESS



FUNCTIONAL DECOMPOSITION OF EXPLORATION CASE STUDY INTO ACTIVITIES AND TASKS



I.S. - INJECTION STAGE   C/O - CHECK-OUT   CV - CARGO VEHICLE   PV - PILOTED VEHICLE   LEO - LOW EARTH ORBIT   STV - SPACE TRANSFER VEHICLE   RNDV - RENDEZVOUS   RET - RETRIEVE

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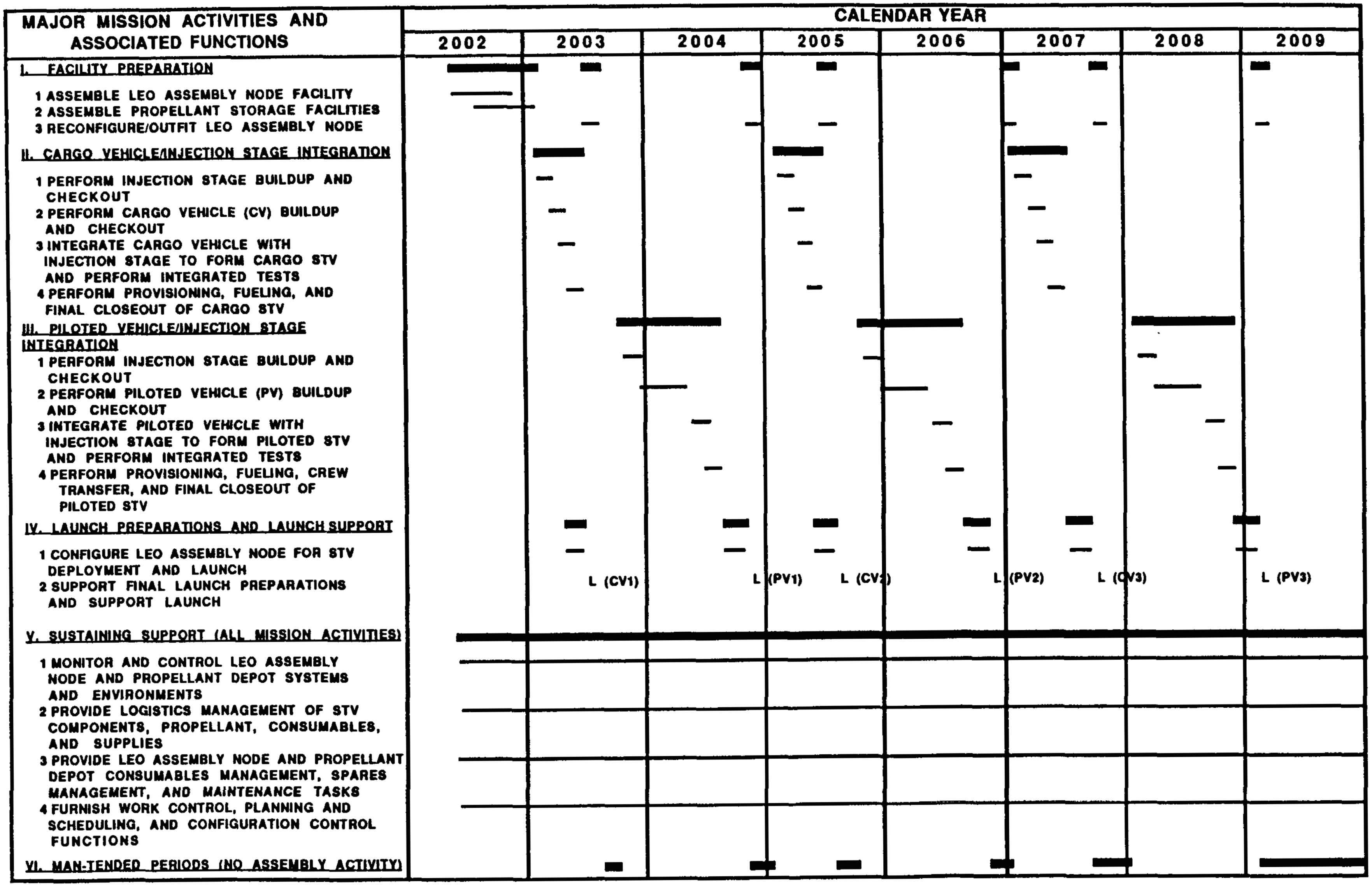
EXAMPLE OF PRELIMINARY SCHEDULE PRODUCT

PRELIMINARY SCHDULE OF TOP-LEVEL FUNCTIONAL REQUIREMENTS FOR MANNED MARS MISSION ASSEMBLY IN LOW EARTH ORBIT (ASSESSMENT BASELINE - CASE STUDY 2)



EXAMPLE OF IMPLEMENTATION ALTERNATIVES

ACTIVITY: III. PILOTED VEHICLE/INJECTION STAGE INTEGRATION
FUNCTION: 2) PERFORM PILOTED VEHICLE BUILD-UP AND CHECKOUT

TASK DESCRIPTION:

B. BERTH PILOTED VEHICLE ELEMENTS AT LEO ASSEMBLY NODE

- TO FACILITATE THE ASSEMBLY, INSPECTION, AND CHECK-OUT OF PV COMPONENTS, THE INDIVIDUAL ELEMENTS ARE BERTHED AT THE LEO ASSEMBLY NODE. SAMPLE TASKS MAY INCLUDE DOCKING PV ELEMENTS TO ASSEMBLY FIXTURES/WORKSTANDS AND CONNECTING SUPPORT RESTRAINTS/UMBILICALS.

> OPERATION IMPLEMENTATION ALTERNATIVES -

- EVA CREWMEMBERS PERFORM MANUAL DOCKING/BERTHING TASKS
- CREW ACCOMPLISHES BERTHING OPERATIONS UTILIZING A COMBINATION OF EVA AND IVA TECHNIQUES AND PROCEDURES.
- BERTHING AND DOCKING PROCEDURES ARE COMPLETELY AUTOMATED WITH NO CREW INVOLVEMENT

PRELIMINARY ASSESSMENT CATEGORIES

- INTRAVEHICULAR ACTIVITY (IVA) VERSUS EXTRAVEHICULAR ACTIVITY (EVA)
- CREW CONTROL VERSUS AUTOMATION
- SAFETY
- OPERATIONAL FLEXIBILITY
- SPECIAL SUPPORT SYSTEMS REQUIREMENTS
- AUTOMATION AND ROBOTICS REQUIREMENTS
- GROUND SUPPORT REQUIREMENTS
- COMMONALITY
- MAINTENANCE REQUIREMENTS
- NEW TECHNOLOGY REQUIREMENTS

PRELIMINARY CONSIDERATIONS FOR ASSESSING OPTIONS FOR PERFORMING A TASK INSIDE [INTRA-VEHICULAR ACTIVITY (IVA)] OR OUTSIDE [EXTRA-VEHICULAR ACTIVITY (EVA)] A PRESSURIZED VOLUME

- Many tasks must be conducted using EVA due to the location of the worksite
  - early design inputs can reduce the number of outside elements requiring EVA

- Where crew-conducted IVA/EVA determination is discretionary:
  - IVA is vastly more productive due to shirtsleeve environment and minimum overhead
  - IVA is roughly one third as costly as EVA
  - EVA is more hazardous
  - EVA has some unique interface design and support equipment requirements

- Crew-conducted EVA is the best method of providing real-time assessment and response to unplanned occurrences at external worksites

- With a high degree of automation and robotics envisioned, an optimum mix of mutually supportive IV, EV, and A&R tasks are required.

The operations goal is to maximize assembly productivity through proper allocation of tasks IV versus EV, including application of automated and robotic systems.

PRELIMINARY CONSIDERATIONS FOR ASSESSING OPTIONS FOR CREW CONTROLLED VERSUS AUTOMATED OPERATIONS

- The on-orbit crew is the most critical assembly resource.
- Automation relieves the crew of the burden of routine tasks, increasing crew efficiency.
- Crew control allows more flexibility and may provide more options for response to unforseen conditions.
- Automation is preferred for complex and repetitious tasks with many pre-identified variables
- Automation is needed for control of processes outside the human bandwith of capability
- Automation can reduce crew exposure to hazardous operations
- Automation requires increased program resources for development, implementation, and verification of the automated processes.

The operations goal is to use automation where it will enhance overall productivity or reduce risk.

PRELIMINARY CONSIDERATIONS FOR ASSESSING OPTIONS FOR SAFETY

- Any option which results in an uncontrollable hazard to crew health or safety should be removed from consideration or modified such that the hazard can be controlled.
- Any option that results in an uncontrollable hazard to equipment required for the success of the mission should be modified such that the hazard is controlled.
- Preference in selection of options should be given to those which have the the fewest and most controllable hazards.

The operations goal is to maximize crew safety and mission success probability through identification, elimination and/or control of operational hazards.

PRELIMINARY CONSIDERATIONS FOR ASSESSING OPTIONS FOR FLEXIBILITY, NEED FOR SUPPORTING RESOURCES, AND ACCOMMODATION OF CONTINGENCIES

- The ability of an option to be performed independently of other events (e.g. sunlight or dark conditions) with minimum constraints on supporting systems (e.g. data, ground command, or communications systems), and without attitude and pointing constraints, is highly desirable.

- Options which are not dependent upon completion of prior functions or tasks are preferred.

- It is preferable that options be selected which can accommodate early or delayed delivery of elements.

- Options which do not require the use of additional spacecraft (such as communication satellites, space tugs, or crew ferry vehicles) are preferred.

The operations goal is to provide the on-orbit assembly crew with the highest degree of autonomy practical, in order to provide the maximum flexibility to adapt mission plans and schedules in real-time to suit on-orbit conditions.

PRELIMINARY CONSIDERATIONS FOR ASSESSING OPTIONS WHICH REQUIRE SPECIAL SUPPORT SYSTEMS AND EQUIPMENT

- Task-unique systems and equipment requires design, development, fabrication, training, planning, and logistics support which otherwise would not be required.
- Preference should be given to options which can be accomplished with no special equipment or special tools.
- Special tools or equipment items are justified if they result in significantly improved productivity or reduced risk.

The operations goal is to drive requirements for assembly unique hardware and systems to zero.

PRELIMINARY CONSIDERATIONS FOR ASSESSING OPTIONS ASSOCIATED WITH AUTOMATION AND ROBOTICS

- Considerations for assessing assembly tasks for robotics applications include:
  - Frequency of the desired tasks
  - Hazard levels of the required tasks
  - Boredom factor of the work
  - Human bandwidth of capabilities required for the tasks
    - Required minimum dexterous manipulation
    - Continuous duration requirements of the task

- Robotics and artificial intelligence systems should be as generally applicable as possible; special purpose automation is to be avoided.

- For mission success, all robotic tasks must be able to be completed by the crew in a straight forward manner in case of system failure.

- When possible, supervised autonomy is to be preferred over tele-operated devices although both modes of operations should be available.

The operations goal is to use automation and robotics wherever their use efficiently expands crew and ground capabilities, reduces crew hazards, or enhances overall productivity.

PRELIMINARY CONSIDERATIONS FOR ASSESSING OPTIONS FOR GROUND SUPPORT REQUIREMENTS

- The ground generally has much greater resources (personnel, skills, support systems) available for operations planning and analysis, monitoring of systems performance, and troubleshooting.

- Extensive dependence on ground support requires highly coordinated ground/on orbit planning and interlinked operations.

- Experience has shown that built-in flight crew/ground interactions can enhance on-orbit efficiency.

- Use of ground support resources to perform non-critical tasks (e.g. routine system reconfiguration) can result in reduced crew workload, reduced crew skill requirements, and increased crew time availability for mission-essential tasks.

- Implementation of ground support for critical tasks requires a high degree of reliability in communications systems.

- Safety and mission success task options which depend on real-time communications should be avoided.

The operations goal is to achieve the proper allocation of functions between ground and on-board resources to maximize both overall productivity and mission success probability.

PRELIMINARY CONSIDERATIONS FOR ASSESSING OPTIONS FOR COMMONALITY

- Use of common mission elements results in reduced design, development,fabrication, packaging, spares costs, etc.

- Use of common elements reduces crew training requirements, documentation, and planning efforts

- Force-fit commonality, i.e. "pick one and go with it", can result in reduced performance and inefficient systems.

- Force-fit commonality can reduce development costs and operational risks.

- Utilization of common elements reduces logistics costs and payload to orbit requirements.

The operations goal is to promote operational efficiency through the use of effective common processes and designs, especially crew interfaces.

PRELIMINARY CONSIDERATIONS FOR ASSESSMENT OF MAINTENANCE REQUIREMENTS

- As the quantity of hardware/software systems at the assembly node increases an increased maintenance workload must be accommodated.
- Capability (including spares) to maintain safety-critical and mission-critical equipment should be available at the assembly node.
- Due to the higher overhead and lessor productivity of EVA as compared to IVA, high-maintenance systems should be within habitable pressurized volumes whenever practical.
- Proper design for assembly must include proper design for maintenance.
- Design for crew-conducted EVA or robotics as prime maintenance means (e.g. for ORU replacement) should not preclude the other as backup.

The operations goal is to have design features that minimize maintenance requirements and maximize system maintainability incorporated into the design baselines at the earliest opportunity.

PRELIMINARY CONSIDERATIONS FOR ASSESSING OPTIONS WHICH DEPEND ON TECHNOLOGY ADVANCES

- There is a natural operations bias towards proven systems, designs and processes . ("Better is the enemy of good enough.")
- Options which have been demonstrated in a space environment should be given preference, unless untested options offer substantial gains in productivity, reduced mission risk, or improved safety.
- Options which require extensive research and development efforts should be avoided unless substantial gains in productivity or reductions in risk can be realized.
- Options selected should *not* exceed the envelope of man's capability in space which is projected to exist at the time the Manned Mission to Mars will be conducted.

The operations goal is to build on proven capabilities and to expand them by identifying areas where technology advances will contribute materially to the accomplishment of the assembly mission.

EXAMPLE OF EVALUATION MATRIX

ACTIVITY: III. PILOTED VEHICLE/INJECTION STAGE INTEGRATION

FUNCTIONAL REQUIREMENT: 1. PERFORM INJECTION STAGE BUILDUP AND CHECKOUT

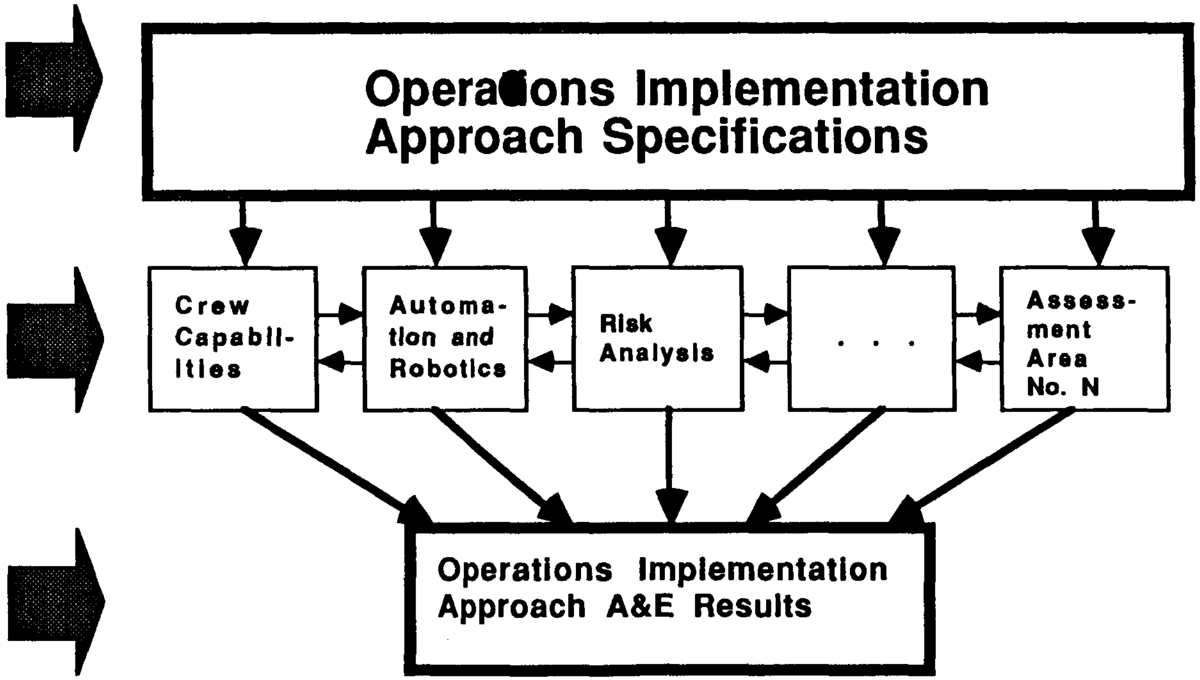
TASK	IMPLEMENTATION ALTERNATIVES FOR OPERATIONS ACTIVITIES	ASSESSMENT CATEGORIES: SUITABILITY OF TASK FOR PERFORMANCE USING IVA VERSUS EVA	ASSESSMENT CATEGORIES: CREW CONTROLLED VERSUS AUTOMATED OPERATIONS
a. Support rendezvous and retrieval of Injection Stage elements	o EVA crew performs rndzv/ret operations using a man-piloted maneuvering-type vehicle. o IVA crew performs and monitors rndzv/ret using a teleoperated unmanned ferry vehicles. o Rndzv/ret is automated and monitored by ground support.	Remotely operated latches can allow IVA operations and this approach is preferred. Ground ephemerous support required.	Automated with crew override should provide highest productivity.
b. Berth Injection Stage elements at LEO Assembly Node	o EVA crewmembers perform manual docking/berthing tasks o Crew accomplishes berthing operations utilizing a combination of EVA and IVA techniques and procedures. o Berthing and docking procedures are completely automated with no crew involvement.	NSTS experience shows berthing using remote manipulator is feasible. EVA may be required to support umbilical mating.	Crew monitoring/ control required to ensure acceptable closure rates.
c. Inspect and perform initial check-out of individual Injection Stage elements	o Crew performs visual inspection and initial check-out using on-orbit test facilities and support systems. o Crew and ground support perform joint testing o Built-in-test-equipment performs an automated self-test of each component.	CCTV and BITE should allow completion via IVA.	Combination of crew and automation should provide best combination.
d. Mate mechanical and electrical element interfaces	o Mating done using a combination of EVA and IVA crew o Mating requires minimal crew participation and can be accomplished using IVA robotics and automation techniques. o All mating and integration is automated and monitored by the ground, requires no crew involvement.	Substantial design penalty will be imposed if all EVA is to be eliminated. Combination of EVA and IVA recommended.	Crew control expected to be most desirable due to number of mating configurations expected.
e. Perform Injection Stage check-out and testing	o Crew performs integrated systems testing and check-out using on-orbit test facilities and support systems. o Crew and ground perform joint integrated systems testing o Built-in-test-equipment performs an automated self-test of each integrated system and mated interface.	BITE should allow IVA operation.	Automated with crew oversight should provide highest productivity.
f. Resolve test discrepancies, perform repairs	o Crew resolves test discrepancies and performs repair/replacement activities. o Ground troubleshoots and resolves test discrepancies and crew performs repair/replacement tasks.	EVA, teleoperator, or robotics required if flight with inoperative elements is to be avoided.	Crew controlled provides necessary ability to tailor response to individual discrepancies.

TYPICAL TECHNICAL ASSESSMENT AND EVALUATION (A&E) PROCESS

Details of Approach:
- Level of onorbit mating
- Ground/onorbit partitioning
- Assembly sequencing
- Etc.

Specific A&E Areas:
- Crew operations required
- Level of automation
- Derived OSE requirements
- Timeline analysis
- Mission effectiveness
- Program risk
- Etc.

Overall A&E Results:
- Assign comparison parameter values
- List issues and concerns
- Additional study suggestions
- Etc.



TRADE STUDY CANDIDATES

Trade #	Description
1	Baseline - Assess candidate implementation approaches for assembly in LEO and develop the preferred operational approaches and the associated operations requirements and guidelines. Principal study requirements and assumptions are obtained from the SRD, but can be supplemented by results from ancillary studies, e.g. Manned Mars Mission Accommodation - Sprint Mission (LaRC).
2	Assess the trades associated with the use of expendable versus reusable STV elements.
3	Assess the impacts associated with a reduced number of missions.

PRELIMINARY RESULTS AND FINDINGS

- A PRELIMINARY METHODOLOGY TO ASSESS OPERATIONS SUPPORT SYSTEMS AND TECHNIQUES HAS BEEN DEVELOPED

- THIS METHODOLOGY IS BEING APPLIED TO LEO ASSEMBLY OPERATIONS FOR CASE STUDY 2. FY88 INTERIM RESULTS INCLUDE THE FOLLOWING PRELIMINARY PRODUCTS:

  -OPERATIONS TASK BREAKDOWN
  -ASSESSMENT CRITERIA/CATEGORIES AND GOALS
  -EVALUATION MATRICES
  -IMPLEMENTATION ALTERNATIVES
  -SCHEDULE OF EVENTS
  -IDENTIFICATION OF ASSESSMENT TOOLS
  -IDENTIFICATION OF FY89 STUDY CANDIDATES

PRELIMINARY RESULTS AND FINDINGS (CONCLUDED)

- ONORBIT TECHNIQUES CONDUCTED TO DATE FALL FAR SHORT OF DEMONSTRATING THE TECHNOLOGY REQUIRED FOR MANNED MARS MISSION LEO ASSEMBLY OPERATIONS
- MANY REQUIRED ASSEMBLY OPERATIONS TASKS WILL BE DEVELOPED/DEMONSTRATED BY SPACE STATION PROGRAM
- LEO ASSEMBLY OPERATIONS WILL BE CONSTRAINED BY ETO TRANSPORTATION AND LEO NODE OPERATIONS
- GAPS IN ASSEMBLY ACTIVITIES AT THE LEO NODE COULD DRIVE LEO NODE SYSTEMS TO HAVE BOTH MANNED AND MAN-TENDED CAPABILITIES

OPERATIONS SAA PROPOSED FY89 ACTIVITIES

- ASSEMBLY AND CONSTRUCTION
  - CONTINUE LEO ASSEMBLY OPERATIONS ASSESSMENT FOR CASE STUDY 2
  - ORBITAL ASSEMBLY OPERATIONS ASSESSMENTS FOR OTHER CASE STUDIES

SPECIAL ASSESSMENT STUDIES:

AUTOMATION AND ROBOTICS / HUMAN PERFORMANCE FOR EXPLORATION CLASS MISSIONS

JULY 1988

AMES RESEARCH CENTER
NATIONAL AERONAUTICS AND
SPACE ADMINISTRATION

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INTRODUCTION

This Special Assessment Study has been conducted for the NASA Office of Exploration (NASA Code Z). In the context of future Exploration Class Missions, this study covers the automation and robotics (A&R), and human performance (HP) OEXP evaluations.

The scope of this Study is based on the Charter for Special Assessment Agents (SAA) which has been published by the Office of Exploration in Section 3.3.4 of the Exploration Requirements Document (ERD). This charter document requires the SAA to focus on the big problems and high leverage opportunities of the initiatives. In these efforts, the SAA is encouraged and licensed to identify and evaluate conventional or unconventional systems, technology, configuration, and technique options that potentially have a high leverage or major impact on various human exploration scenarios - "[the SAA is to] analyze the exploration scenarios from the vantage point of 'chief engineer' for the parochial viewpoint of their specific assignment". Specifically the A&R/HP SAA has been chartered to examine the use of man and machine as tools in the Case Studies. This includes assessment of the artificial intelligence and robotics requirements as well as human performance and man/machine trade-offs.

The study and evaluation process for automation, robotics, and human performance is schematically presented in Fig.1. Based on ongoing research, and current knowledge in industry, NASA Centers, and universities, the current Code Z scenario documentation has been reviewed and evaluated. An Inter-Center Working Group was formed to ensure appropriate technical support and input to detailed technical studies and evaluations as the exploration scenarios mature. In order to establish a reliable base for automation and robotics technology forecasting to the Year 2000 era, a workshop was held involving about fifty experts from universities, industry, and NASA (see appended Report on the Workshop: Robotic Needs for the Human Exploration of the Moon and Mars). Furthermore, continuing contributions by, and interactions with, the academic community are promoted.

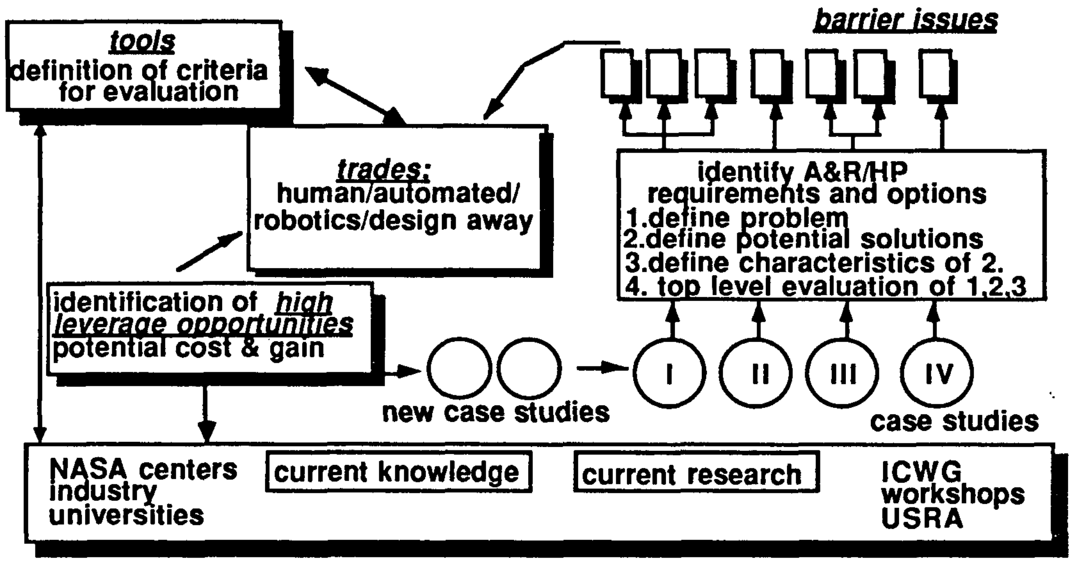

figure 1

II. OBJECTIVES OF A&R/HP ASSESSMENT

The exploration scenarios under consideration require complex operational efforts in space and on distant planetary surfaces to assemble, operate, and maintain systems of various kinds. These will involve humans and machines in appropriate combinations to achieve the best possible effectiveness of the overall system. The objectives of the A&R/HP special assessment studies are to review critically the OEXP case studies and:

- assess the feasibility of the automation, robotic, and human performance assumptions;
- identify A&R system options;
- estimate the appropriate level of automation and robotics to accomplish feasible and man-machine balanced, cost-effective operations in space;
- identify areas where conceptually different approaches to the use of people and machines can leverage the benefits of the scenarios;
- recommend modifications to scenarios or new scenarios that will improve the expected benefits;
- develop analytical tools for future assessments.

III. RATIONALE FOR A&R/HP ASSESSMENT

The appropriate employment of extra-vehicular activities and/or robotics in space activities is a critical consideration for the success of large-scale operations including the assembly, construction, and maintenance of space stations, space platforms, and space transportation vehicles. To accomplish these tasks entirely by EVA is almost doomed to failure, not only from a technical point of view, but also because safety and the relatively high cost to keep people on an EVA. On the planetary surface of Mars long time delays will necessitate man's presence or the use of automation to meet the required mission objectives. Man and machines, in both of these situations, are tools that can perform the different required tasks. Due to safety, cost, and reliability, there will necessarily be a combination of man/machine use in performing the various tasks of the future manned space missions. An analysis and assessment of the operational aspects of the Exploration Scenarios should point to major areas where automation and robotics can provide appreciable benefits when traded against human performance operations. These areas should then be isolated, to the degree possible, and defined and analyzed in detail to identify and specify realistic combinations of A&R/HP within the context of the mission scenario.

This analysis and assessment process aims at improving the understanding of automatic and robotic machine capabilities by space systems designers. The technologies for these space system capabilities are still in their infancy, and very little in-space operational experience has been accumulated to date. Furthermore, todays technologies in automation and robotics will surely be obsolete by the year 2000, at the time when the systems for the Exploration Class Missions will be designed. This assessment should therefore give insight into the types of automation and robotics technologies that will be of key significance for the Exploration Class Missions and which should be developed during the next decade.

IV. PRECURSOR MISSIONS AND TECHNOLOGIES

In assessing the Exploration Scenarios we must project at least to the Year 2000 and take into account the state of technology and operational experience in space at that time. The OEXP Case Studies state the the allowed technology assumptions that include current technologies and some future system. The current capabilities of Space Shuttle and the Manned Maneuvering Unit are considered baseline for all low earth orbit space activities. In addition, Phase I space station, a Heavy Lift Launch Vehicle with a twenty-five foot shroud and capable of transporting ninety-one metric tons to low earth orbit, the Flight Telerobotic Servicer, an Orbital Maneuvering Vehicle, and an Orbital Transfer Vehicle are often included in the OEXP assumptions.

Precursor missions that will affect the A&R/HP assessments that are often assumed include the Mars Observer, the Lunar Observer, and the Mars Rover and Sample Return Mission. The Mars Observer, planned for 1992, will provide the necessary survey of the Martian surface to establish appropriate landing and habitability sites for the Exploration Class Missions. The Lunar Observer, planned for 1994, will provide the necessary survey information for locating lunar science bases and/or mining and propellant production facilities. The Mars Rover and Sample Return mission, planned for 1998/01, will give pertinent science information about Martian surface conditions, material properties, and the environment. But most important, from an operational point of view, it will utilize automation and robotics technology at a relatively advanced level under conditions of large communication time delays. The robotic operations on the surface of Mars include navigating, traversing, rendezvousing, proximity operations, docking with another body, inspection of samples, handling and manipulating, analysis of samples, coordinating among systems, self maintenance, monitoring, and recovery from faults.

In summary, by the Year 2000, when the design process for the Exploration Class Missions is under way, the in-space operational experience in human performance, automation, and robotics will have reached a considerable level of accomplishment in the following areas:

- Extra Vehicular Activities in Earth orbit
- In-space assembly and construction
- Handling and inter-orbit transport of large objects
- Autonomous self-maintenance of space systems
- Remote sensing and navigation in interplanetary space
- Telerobotic operations with large communication time delays including:

  Planetary surface navigation and mobility
  Sample acquisition, handling and analysis
  In-space rendezvous and docking

From this list of capabilities, we can deduce that most automation and robotics technologies required for the Exploration Class Missions will have been exercised to a limited degree by the Year 2000. This will provide a technology base, which, together with the parallel developments in research laboratories, will satisfy the special system design needs of the Exploration Class Missions.

V. GENERAL PRINCIPLES FOR ASSESSMENT

The characteristics of the Exploration Class Missions considered in these assessment studies are such that at least some automation and/or robotics capabilities are required to accomplish the mission objectives. These capabilities are generally necessary because of the high cost and safety concerns in the space environment. In addition, A&R can reduce the operational complexity by reducing the number of humans involved in a task and in coping with large communication distances.

These statements imply that an idealized operational cost-effective function in terms of increasing A&R would be unacceptably large for zero A&R. This function would sharply drop with increasing A&R to some minimum value and would then start rising again. The rise of this cost function signifies the fact that for ever increasing A&R the technologies may not be available, or the automated and robotic system may become technically too complex to maintain adequate reliability and safety, or acceptable cost. It follows that to aim at too much A&R may be counter-productive. It appears extremely important to recognize that as long as at least one human is around to handle the occasional unexpected, such as exchanging a module, undoing an entanglement, etc., the system can be designed much simpler from an A&R point of view.

The assessment of A&R/HP issues requires consideration of a wide set of design variables which are used to describe the physical and operational characteristics of the system.

A. Performance Parameters for A&R/HP Design

The A&R performance parameters describe the physical and operational characteristics of the system. They include the attributes specifically describing the implementing agents, either human or machine and the unique characteristics of the tasks, irrespective of how they are implemented.

Human Performance Characteristics - The characteristics distinguishing humans from machines include generally those associated with intelligent, intuitive behavior, such as selective recall, judgment, improvisation, situation adaptation, generalization, pattern recognition, ambiguity tolerance, global awareness, etc. However, in space operations, particularly EVA, the human has severe limitations in handling large-scale systems, in long-term physical endurance, and in small-scale dexterity because of the gloves of the space suit. It follows that most EVA are done using various (hand) tools.

Machine Performance Parameters - Some of the typical, mostly computer-based machine performance parameters include large data storage, precise repetitive operations, quick response, different types of sensors and broad sensor ranges, control of large forces, endurance, no distraction, rapid complex and preprogrammed analysis, no physiological bounds, except those imposed by technological limitations.

Task Parameters - For simple tasks, speed of performance and accuracy of implementation are usually foremost in trade-off considerations. The human is the measure of all things, however, in space we may have to deal with physically very large systems and over very long time periods. The scales both in physical and time dimensions relative to "human scales" then become important task parameters. As tasks become varied in kind, manipulative dexterity enters into the picture, and as the overall task complexity increases with an increasing number of of elementary tasks with different starting times, the timely and economical scheduling of tasks becomes an increasingly more important performance parameter. Tasks or operations requiring planning into the future presuppose increasing intelligent capabilities with increasing complexity.

There are numerous criteria for evaluating a system. For manned systems in space, safety usually ranks highest. This is followed by reliability and life-cycle cost. In addition to task parameters, many auxiliary, technical parameters are used to describe system performance, capability, adaptability, and the like. At a particular time, these parameters reflect the state of technology as the basic input for system design. The usual approach to system design consists of an iterative process in which first the functional objectives are specified and then, based on a given state of technology and minimum required safety and reliability, the life-cycle costs are minimized.

B. System Operations

Automatic and robotic systems must perform their assigned tasks within their task environment. All the while, these systems must be maintained in operating conditions through maintenance and occasional repair. This requires capabilities of self-maintenance and the performance of functional tasks for which the system was designed.

System Self-Maintenance - Automated and robotic systems should maintain operating conditions autonomously through fault tolerant designs, graceful failure techniques, modular designs with hierarchical structure where possible,and fault alarms to the module exchange level. Trade-offs between redundancies in the design, spare modules to be carried along, and built-in reliability are of prime concern. Manual maintenance should be restricted to module exchange operations.

Functional Task Performance - The required functional task performance operations can be put into a few categories. At a high level of abstraction, these categories are distinguished from each other by the kind of subsystems they require for task implementation.

Movement from the present position to another area of interest by free flight in space, or by driving, walking, rolling on a planetary surface, or by atmospheric flight, requires appropriate capabilities of navigation and mobility. Rendezvous, proximity operations, and docking with another body require proximity sensors, finely adjustable propulsion and mobility, and docking and attachment mechanisms. Inspecting objects with respect to given criteria requires appropriate sensors and identification and recognition capabilities, the latter being provided by artificial intelligence techniques. The handling and manipulation of such objects requires, in addition, grasping (docking) capabilities and objective oriented coordination (hand-eye coordination) of actuators and sensors. Finally, if there are more than one robot systems, they should be able to coordinate their activities with each other in a cooperative fashion.

C. Task Performance Operations

Many functions in space could be performed by astronauts in space suits at the task site. However, there are many tasks which require larger, more powerful systems and with greater performance duration. Such systems may be tele-operated from a control station or they may be robotic systems which require only intermittent supervisory control.

EVA Functions - EVA operations require at least two astronauts with at most moderate communication time delay (up to lunar distances) from a control station where at least one control operator is in continuing attendance. The control station may be on Earth, in orbit, or on a planetary surface. It requires display feedback systems about the remote operations and a voice input system for uplink communication. The astronauts at the worksite have suitable life support systems and tools to handle and manipulate objects which should be designed "astronaut friendly." The size of the handled objects is limited to "man size," and the operating cycle may reach a duration of not more than six hours.

Teleoperated Functions - Many tasks require two cooperating systems, such as two astronauts or two robotic devices, at the work site. For two tele-operated robotic devices, we require in the control station one supervisory operator and two dedicated operators with display feedback, force feedback, and dedicated manipulative input systems. The communication time delay is restricted to less than one second, preferably to less than one half of a second to avoid control instabilities. For tasks which require only one tele-operated robotic device, at least one operator and some time of the supervisor in the control station can be removed. If the operators can be rotated in shifts, the operating cycle is unlimited. In any case, the objects to be handled should be designed robot friendly. Both, astronauts and robotic devices, are expected to use tools. The objects to be handled should accordingly be designed tool friendly, while the tools should be designed either astronaut or robot friendly.

Telerobotic Functions - If autonomous capabilities are added to the robotic devices at the remote site, the communication time delay can be increased, accordingly. The operators in the control station then take on functions of supervisors providing intermittent supervisory monitoring and control and the time or number of operators may be reduced.

D. General Design Guidelines

Before considering and assessing the individual Exploration Class Missions and the corresponding scenarios, we are able to derive general design guidelines for automation and robotics from the preceding discussions and from the results of the previously mentioned Robotics Workshop.

Modular designs and self deployment techniques for structures, substructures, vehicles, machinery, etc. should be used to the degree possible. In these considerations, automated beam and truss building in space is a form of self deployment. In-space assembly is generally easier with fewer and larger objects than with many and smaller objects. Rendezvous and mating techniques with standardized interfaces should be used where possible, and complexity should be avoided.

All objects that need to be manipulated should be designed EVA and robot friendly. They should be clearly labeled and easily identifiable from arbitrary orientations. They should be designed for easy grasping, which is important for items to be picked up and for crawling robots and astronauts.

Unique operations and tasks requiring circumspection should generally not be automated. This task category includes replacements of modules, repairs, etc. Unique operations are relatively expensive to automate, and circumspection requires artificial intelligence technologies which are sometimes not readily available.

Recurring operations and definable tasks, such as system monitoring, fault diagnosis,house-keeping,etc. should be automated to the fullest degree possible. Such autonomous capabilities can help the system reliability and can support system self maintenance.

Software engineering and maintenance are lead items for all automation and robotic systems. Software systems should be conceptualized and designed in conjunction with the associated operational hardware systems.

At the remote site, systems autonomy allows us to limit the amount of required communications between the control station and the remote system, because the system is able to make many operational decisions on its own. The desire to reduce the communication traffic between control station and remote system may have numerous sources, such as planetary occultation, two-way light time, limited bandwidth, error rate, response time of equipment, etc.

VI. BROAD ANALYSIS OF EXPLORATION SCENARIOS

The following preliminary, high-level analysis is based on the above assessment and on material published by NASA Code Z about the Exploration Class Missions. The specific Exploration Scenarios are analyzed from the point of view of automation and robotics or human performance in all operational phases. The key operational elements pointing to significant involvement of automation and robotics and the issues requiring in depth study are identified.

A. Human Expedition to Phobos

Requirements for A&R and HP for this Case Study are in Earth orbit, where the various subsystems of the cargo mission and the piloted mission will be assembled using automated rendezvous and docking operations. On the Martian surface, two landers with rovers will be automatically deployed. From the piloted Mars orbiting vehicle with a crew of four, the rovers will be tele-operated performing surface science investigations and sample return to the landers for lift-off and recovery by the orbiter. The surface of Phobos will be explored through EVA operations.

The required automation and robotics technologies are expected to be available by the Year 2000 primarily through preceding missions. The Space Station and its adjuncts will provide the technical basis and testing ground for the assembly operations in Earth orbit. The MRSR mission will use and qualify automation and robotics technologies required for the Mars surface operations.

The crew of four in the piloted Mars orbiter requires the optimization of performance and scheduling when teleoperating the two rovers on the Martian surface. As a consequence, the rovers may require substantial autonomy for traversing, sample acquisition, and science performance. It may be required to share control of the rovers with the operations center on Earth. For example, long term plans and analyses could be done on Earth, while short term control operations could be performed by the orbiting crew.

B. Human Expedition to Mars

The requirements for A&R and HP in this Case Study emerge at several places. In Earth orbit, various systems for both the cargo mission and the piloted mission are assembled using automated rendezvous and docking operations. To the degree possible self deployment techniques will be used as might, for example, be necessary for the fifty foot diameter Martian entry aeroshell. Fluid transfer operations will be made in LEO and in Mars orbit using specialized autonomous transfer mechanisms. Landing of the cargo vehicles and site preparation on the Martian surface will be done automatically primarily with self deployable systems. Two rovers will be tele-operated from a Martian based control station.

Most of the required automation and robotics technologies will be available by the Year 2000 through preceding missions. Although, the assembly operations are more elaborate than for the Phobos Case Study, the demands on automation and robotics technology are about the same. The autonomous technologies for the teleoperation of the two rovers on the Martian surface are estimated to be somewhat less advanced than for the Phobos Case Study.

There are however two notable exceptions which require special consideration. First, the fluid transfer technology in zero gravity is not in hand and requires special study to determine the core of the problem and possible solution characteristics. Second, the fifty foot diameter Mars entry shell must be brought into Earth orbit in several pieces which must be deployed or assembled in space. This process appears to have special problems because of the required precisions for mating at the outer shell surface.

C. Lunar Observatories

As in the Phobos and Mars cases, this Case Study scenario requires Earth orbital assembly operations, including A&R and HP technologies for automated rendezvous and docking. Fluid transfer operations in LEO will use specialized autonomous transfer mechanisms. Emplacements of instruments and other machinery on the lunar surface will be done primarily through self deployable systems with some EVA. The emplaced science packages will then operate autonomously, except for occasional repair functions,if required. The necessary automation and robotics technologies will be available by the Year 2000. Much of the lunar surface instrumentation technology has been used on the Apollo project, although at a less elaborate level. However, similar as for the Mars case, the fluid transfer technology for zero gravity operations still needs study.

D. Lunar to Early Mars Outposts

Because of the complexity of this exploration scenario, the A&R and HP requirements in this Case Study are generally more extensive than for the previous cases. Systems will be assembled in Earth orbit at a LEO servicing node using automated rendezvous and docking operations. For the Martian entry aeroshell, self deployment or special assembly procedures will be used. On the lunar surface, telerobotic techniques for mining and hauling, and autonomous techniques for propellant production and storage will be employed. The deployment of the corresponding machinery will be done automatically to the degree possible, and with the help of EVA. Propellant transfers in earth orbit, on or near the Moon, and in the vicinity, or on the surface, of Mars will be accomplished using specialized autonomous transfer mechanisms. Proximity operations at, and landing and operating on, Phobos or Deimos will partially be done automatically by self deployable equipment and partially by teleoperation.

Many of the automation and robotics technologies will be available by the Year 2000, as stated for the previous Case Studies. Although, the overall extent of the assembly operations are here considerably more elaborate, the required automation and robotics technologies are about the same. Similar statements can be made about the operations on the lunar surface and those in the vicinity of Mars.

Nevertheless, there are several areas that require special consideration. The fluid transfer in zero gravity requires special study to define the problem and identify possible solutions. The 50 feet diameter Mars entry shell must be assembled or deployed in Earth orbit, which requires a feasibility study. The whole process on the Moon, starting with equipment deployment, tele-operated mining, automated propellant production, and automated propellant storage requires a system study based on previous related work by NASA. Finally, the details of autonomous or tele-operated rendezvous, docking, and surface operation on Phobos and Deimos require study and definition.

E. Relative Evaluation of Benefits and Risks

All of the Case Studies call for some level of automation and robotics. This is because the application of automation and robotic technologies can provide certain benefits. Exactly how much benefits in terms of higher reliability, more safety, greater performance,and lower costs is at the current level of scenario descriptions hardly possible to determine. Similarly, it is not possible to give precise indications about the technological risks involved in applying automation and robotics technologies that will not have had an appreciable track record. Nevertheless, taking into consideration the overall picture of each exploration scenario, heuristic, qualitative statements can be made about the relative benefits and risks involved (figure 2).

Case Study: Benefit / Risk Analysisis
(scale: lowest, low, high, highest)

Case Study	A&R Benefits	A&R techn. risk
Human Expedition to Phobos	high	high
Human Expedition to Mars	high	low
Lunar Observatories	high	lowest
Lunar Outpost to Early Mars Outpost	highest	highest

figure 2

VIII. CONCLUSIONS AND RECOMMENDATIONS

The above high level evaluations indicated those areas that required further examination. Listed below are those areas of critical importance that have been addressed in the past year.

A. Phobos Expedition Mars Rover

In Case Study I, there is a question as to the effectiveness of the crew and the remote operations of rovers on Phobos and Mars. A study addressing this concern was performed by a study team at JPL (Appendix II).

B. Mining, Processing, and Propellant Storage on the Moon

Case Study IV calls for the set-up and operation of a Lunar Oxygen Plant with little to no human interaction. A study addressing simply the question - can a LLOX plant nominally operate autonomously? was performed at ARC. Additional studies concerning the more difficult questions of plant set-up and contingency operations are planned for FY89.

C. Assembly in Earth Orbit

All of the Case Studies call for some degree of assembly in space, from the automated rendezvous and docking of Case Study I to the continual construction of vehicles in the infrastructure in Case Study IV. Due to the broad nature of this problem several efforts have been undertaken in this past year. The Robotics workshop identified the robotics systems options for in-space assembly. A study developing a possible scenario for the assembly of the vehicles described in Case Study II was performed by Boeing. This study defined the problem to a sufficient level of detail to begin performing A&R/HP assessments of the process. Issues of path planning dictated by the complex operations of in-space assembly are being examined at JPL. A last study is developing a database of the data from the Flight Telerobotic Servicer (FTS) providing a metric of comparison of robotics systems in in-space assembly.

D. Extended Automated Planetary Site Preparation

Site preparation on the Moon and on Mars for temporary or permanent human habitability, as a scientific outpost, and for propellant production will require a substantial contribution from automation and robotics technologies. The objective of this area is to investigate the potential benefits to be derived using robotics for the preparation of planetary sites. This is an area of potentially high leverage as it allows for a more optimal use of man in these future exploration missions.

E. Automated Rendezvous, Docking and Deployment on Phobos and Deimos

Rendezvous, docking, and deployment of equipment on Phobos and/or Deimos poses special problems because of the low local gravity field. The objective of this area is to establish a conceptual approach for automating these operations.

F. Automated Propellant Transfer Techniques in Zero Gravity

The transfer of propellants and other fluids in a micro-gravity environment is indicated by all of the OEXP Case Studies. For safety, cost, and efficiency a feasible standardized approach to automated docking, propellant transfer, and disengagement of propellant modules in Earth orbit is desired.

G. In-Space Refurbishing and Turnaround

The turnaround and refurbishing of spacecraft on orbit could prove to be a considerably more difficult task than even the initial, in-space, assembly of the same vehicles. Several studies in this areas are proposed. The first being performed at JPL, is examining the issue of structural integrity specifically the issue of automated check-out.

These first studies performed by the A&R/HP SAA have the primary goal of establishing conceptual feasibility for automation, robotics, and crew application in the OEXP Exploration Class Case Studies. Future studies should continue to address the A&R/HP SAA objectives allowing automation and robotics to be incorporated into these missions during the design and systems integration phase. In this manner the utilization of both man and machine can be optimized.

APPENDIX 1

REPORT ON THE WORKSHOP :

ROBOTIC NEEDS FOR THE HUMAN EXPLORATION OF THE MOON AND MARS

Held in Palo Alto, California,
on May 10-11, 1988

September 1988

Ames Research Center
National Aeronautics and
Space Administration

FOREWORD

In June 1987, the National Aeronautics and Space Administration established the Office of Exploration (OEXP) headed by John Aaron. This organization will provide recommendations and viable alternatives for an early 1990's national decision on a focused program for human exploration of the solar system, in particular to the Moon and Mars. OEXP appointed Special Assessment Agents (SAA) for several key technical areas. The role of the SAA is to, "analyze the exploration scenarios from the vantage point of 'chief engineer' for the parochial view point of their specific assignment". Specifically, the SAA should identify and evaluate systems, technology, configuration, and technique options that potentially have a high leverage or major impact on human exploration. The OEXP Robotics Workshop was a mechanism through which the SAA could draw on robotics expertise in the agency, Universities, and industry to explore realistic long-term robotics capabilities and to gain some understanding of the research required to obtain these capabilities. This will provide the basis for a realistic assessment of the robotics needs for future manned exploration.

AGENDA
NASA OFFICE OF EXPLORATION'S
WORKSHOP ON
ROBOTIC NEEDS FOR THE HUMAN EXPLORATION OF THE
MOON AND MARS

May 10-11, 1988

May 10, 1988

8:30 a.m.	Welcome - Vic Peterson
8:45 a.m.	Introduction - Michael Sims
9:00 a.m.	Remarks on the Office of Exploration - Ed Gabris
9:30 a.m.	Remarks on Project Pathfinder - Peter Friedland
9:45 a.m.	Workshop Objectives/Structure - Michael Sims
10:00 a.m.	Mission Categories and Concepts • In-Space Assembly - Brian Pritchard, presented by Stephen Katzberg • Transportation Requirements - Fred Huffaker, presented by Harvey Feingold • In Situ Resource Utilization - John Alred, presented by Tom Dollman • Planetary Rovers - Michael Sims • Human Factors Considerations - David Nagel
11:30 a.m.	Discussion focusing on top level evaluations
12:00 noon	Lunch
1:30 p.m.	Organize into workshop subgroups
2:00 p.m.	Break into subgroups for detailed discussions
6:00 p.m.	Dinner
8:00 p.m.	General meeting and discussions

May 11, 1988

8:30 a.m.	Subgroup reports - Subgroup Chairs
10:30 a.m.	Task integration discussions
12:00 noon	Lunch
1:00 p.m.	Overview discussions
3:00 p.m.	Summary/conclusions - Michael Sims
4:00 p.m.	Proposal for a Remotely Manned Space Station - Marvin Minsky
5:00 p.m.	Adjourn

LIST OF WORKSHOP PARTICIPANTS:

Harold Alexander, Stanford University
Charles Armstrong, Johnson Space Center
Bill Berry, Ames Research Center
John Bull, Ames Research Center
Bob Cataldo, Lewis Research Center
Dave Criswell, University of California, San Diego
Tom Dollman, Ames Research Center
Mark Drummond, Ames Research Center
Harvey Feingold, Science Application International Corp.
Marty Fischler, SRI International
Peter Friedland, Ames Research Center
Ed Gabris, NASA Office of Exploration
Kathy Healey, Johnson Space Center
Ewald Heer, Heer Associates, Inc.
Tom Jackson, Aerojet Electro Systems Company
Vicky Johnson, Ames Research Center
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Monte Zweben, Ames Research Center

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INTRODUCTION

During the decade of the 1960's, humans expanded their horizons far beyond their natural Earth environment with both manned and unmanned spacecraft. In the former, man was placed in a benign environment of a spacecraft or spacesuit. In the latter, humans remained on earth, while they communicated with the unmanned spacecraft, receiving data remotely. This latter case is a simple example of teleoperations; man received data from space, through a network of sensors and telecommunications. The newly installed space telecommunication system enabled the operation of remote spacecraft as far as the moon, Mars, Venus, and later the outer planets. During the 1960's, such activities became known as teleoperation - teleoperation being defined as general-purpose, remotely controlled, cybernetic, dexterous, man-machine systems that augment and extend human sensory and manipulative abilities to remote places.

The early spacecraft had only the most rudimentary physical control execution devices (special purpose actuators) that would, e.g., cause midcourse trajectory corrections, rotate solar panels toward the sun, and reorient instrument platforms for better viewing. General purpose, but still crude, manipulative, handling, and mobile vehicle devices were introduced early into lunar surface missions and have been improved upon ever since.

Teleoperation in space has severe limitations that have become obvious as the operations with remote spacecraft have become more complex and the time delays between earth and the spacecraft have grown in orders of magnitude. For most operations a remote system should, at least to some degree, be able to sense, make decisions, and act autonomously, i.e., the remotely placed machine should have some degree of intelligence. As early as 1967 this became obvious when the manipulator of the Surveyor III spacecraft was remotely operated from Earth to dig a small trench into the lunar soil and perform a series of simple tests. This process required attention by scores of operators and scientists waiting for video and other feedback information to confirm or negate every minute motion. Commanding the remote device at the level of a single degree of freedom proved extremely inefficient, even for the relatively short communication distance of seconds to the Moon. With telecommunication links that range from several minutes to hours in signal travel time, these inefficiencies will be compounded. For example, it has been estimated that a Mars roving vehicle would be usable only 4% of the time in a strictly teleoperated mode of operation, while a technically feasible intelligent robot system, needing but a minimum of support from Earth-based control stations, could operate at least 80-90% of the time.

Perhaps the most complicated space robot system built, to date, was the Viking Lander that was sent to Mars in 1976. It was reprogrammable from Earth and an autonomously working laboratory performed a series of experiments on anything it could reach with its manipulator. In a similar category of complexity and autonomy were the two Voyager spacecraft, launched in 1977 to flyby Jupiter in 1979, Saturn in 1980/81, and Uranus in 1986. These still relatively crude intelligent robot systems included appreciable capabilities of intelligent decision making and action in response to uncertain environments and unanticipated events. They were designed for more than eight years of operational life, long range communications, high precision navigation, and reprogrammability in flight to support changing science requirements. The onboard computers could sequence onboard operations to adapt to differing data acquisition requirements and to isolate and replace faulty system elements by electronic switching.

The incorporation of progressively more autonomous capabilities in spacecraft has been possible because of the tremendous advancements in electronics, sensors, communications, and computers since the 1960's. As a consequence, space missions have been able to cope with ever increasing complexity and data rates. For example, the data rates that could be handled by planetary missions have increased by about six orders of magnitude, while for Earth orbital missions, they have increased about three-and-one-half orders of magnitude. These trends are expected to continue at least through the late 1990's.

The space program entered a new era on April 12, 1981, with the initial launch of the Space Shuttle, the first manned spacecraft designed for reuse. Planning of the Space Shuttle was started in the late 1960's in anticipation of a permanently manned space station. From the beginning it featured one or possibly two manipulator arms each about fifty feet long with seven degrees of freedom and controlled in a teleoperated/semiautomated mode with the human operator in the control station in front of the cargo bay. The manipulator arm tasks have included handling cargo, aiding the retrieval and deployment of satellites, and assisting astronauts in the repair of spacecraft.

The resurrection of U.S. space travel with the Space Shuttle launch in late 1988 is the next important step towards constructing and operating a Space Station in Earth orbit during the 1990's and beyond. The space station has necessarily spawned a number of auxiliary transportation and operational systems which are now in the planning or design stage for operation in the middle to late 1990's. These systems include the EVA Retriever, the Flight Telerobotic Servicer (FTS), the Orbital Maneuvering Vehicle (OMV), and the Orbital Transfer Vehicle (OTV). A Mars Rover and Sample Return (MRSR) mission is planned for 1998/2001. In addition, a Heavy Lift Launch Vehicle (HLLV) is planned to be available by the year 2001 with a delivery capacity of approximately 90 metric tons to low Earth orbit.

Robotics and teleoperated systems exist on earth and are, in general, more complex and more developed than the systems that have been used in space. To assess the robotics capabilities required in the next several decades, it will be important to examine earth systems for their success, failures, techniques, and final design. In particular, the Remote Operated Vehicle (ROV), and other similar robotics, designed for underwater construction and repair will have design criteria that carry over to space robotics design. The underwater environment is especially applicable as it is dangerous in much the same way space is to man, and as the tasks are similar to those required for in-space assembly.

OEXP: Case Studies in Manned Exploration

The Office of Exploration has developed four Case Studies - possible scenarios of the future manned exploration of space. These Case Studies define more specific domains and activities that may occur in the future manned space program. The mission concepts (Case Studies) currently under consideration by the NASA Office of Exploration include:

1. **Human Expedition to Phobos** - This mission will bring the first humans to the Martian moon Phobos to explore, conduct resource surveys, establish a science station, and conduct enhanced robotic explorations of Mars from Phobos. This Case Study strives to minimize the enabling technologies, life sciences research, and use of space station.

2. **Human Expedition to Mars** - This mission will bring the first humans to the Martian surface to conduct local geological reconnaissance, emplace long-lived geophysical instruments, collect samples for return to Earth, and perform ancillary exploration of the Martian moons, Phobos and Deimos. This Case Study requires the in-space assembly of the Mars-Earth transit vehicles, yet still strives to minimize the enabling technologies.

3. **Lunar Observatory** - This mission will establish a long-duration human-tended astronomical observatory on the far side of the Moon and conduct regional lunar explorations. This Case Study develops an infrastructure with a focus on surface science and also attempts to minimize the impact on low earth operations.

4. **Lunar Outpost to Early Mars Evolution** - This mission begins with the development of a lunar science and resource outpost comprised of a lunar oxygen plant, local-to-regional geological exploration facilities, an astrophysical observatory, and a life sciences laboratory facility. Once the lunar outpost is operational, manned flights to Mars will be undertaken to perform local-to-regional geological explorations of the surface of Mars using manned and unmanned mobility systems, and to perform explorations on Phobos and Deimos for propellant extraction possibilities in support of subsequent missions. This mission will lead to a self-sufficient, sustained human presence beyond earth orbit providing the basis for continuing technology development and a broad infrastructure for growth.

BACKGROUND MATERIAL

The presentations and discussions served as the basis for the subgroup deliberations. They provided the background information on the OEXP program, other current NASA programs(Pathfinder), the workshop objectives and structure, and the the barrier issues in the current OEXP plans.

Ed Gabris of the Office of Exploration, HQ, outlined the Exploration Themes and Scenarios. Referring to the Sally Ride Report, "High Frontier," as a background document, he presented the main goals of the national space policy and proceeded to state the goal of the Office of Exploration as "to provide recommendations and alternatives for an early 1990's national decision on a focused program for human exploration of the Solar System." Precursor missions including a Mars Observer, a Lunar Observer, and a Mars Rover & Sample Return are important steps towards the Exploration Type Missions that include human expeditions to Mars and Phobos, extraterrestrial science outposts, and evolutionary expansion towards a permanent human presence on the Moon and Mars.

The Project Pathfinder was presented by Peter Friedland (ARC). Project Pathfinder is organized around four major thrusts: (1) Exploration, (2) Operations, (3) Humans-in-Space, and (4) Transfer Vehicles. Each thrust focuses on a set of key technology elements to support critical mission capabilities. These elements include such items as planetary surface sample acquisition and analysis, power and propulsion, optical communications, autonomous rendezvous and docking, extraterrestrial resources, in-space assembly and construction, EVA, human performance, autonomous landers, fault tolerant systems, lunar and planetary operations using robotic systems, and the like.

Michael Sims (ARC) pointed out that generally the problems of the Exploration Type Missions are broad and complex. However, in this Workshop, the concentration is only on questions involving robotics within the scope and objectives as laid out for the Special Assessment Agents of the Office of Exploration.

Stephen Katzberg (LaRC) presented, for Brian Pritchard(LaRC) an overview of issues as they pertain to on-orbit assembly, servicing, and check-out (attachment). The concept of operational nodes in Earth orbit, in the vicinity of the Moon, and in a Martian orbit was described. Various options in connection with the Space Station were identified, and tradeoff parameters of human versus teleoperation or robotics were discussed. Some big issue items were identified, such as mechanisms, propellant transfer in space, and multiple interconnects during docking and assembly. Some of these issues were amplified in the following presentation.

In particular, Harvey Feingold (SAIC) discussed material from Fred Huffaker (MSFC) on early automation and robotics demands driven by low Earth orbital transportation requirements (attachment). Specific demands for the Phobos Expedition, the Mars Expedition, the Lunar Observatories, and the Evolutionary Expansion were identified, and requirements for low Earth orbital operations were presented. Some of the transportation related automation and robotics issues are summed up as follows: (1) A&R as a substitute for human presence during low Earth orbit assembly and check-out; (2) criteria for the use of automation versus telerobotics; (3) risk versus cost drivers for A&R developments; (4) enablement and enhancement of transportation activities by A&R.

Tom Dollman (MSFC, ARC) representing John Alred (JSC) presented barrier issues for planetary surface systems and provided a lunar outpost scenario and a lunar base scenario with early self-sufficiency (attachment). The scenarios illustrated concepts for the utilization of lunar soil as radiation protection of habitations and as structural material. Various concepts for surface construction, material processing, mining, and transportation were discussed. The A&R impacts in terms of humans versus telerobots and robots were pointed out for consideration.

Finally, David Nagel(ARC) gave an insightful talk on human factors considerations based on cockpit conversations during critical flight operations. Various examples of failures of human-machine integration were pointed out. A case was made that the problems associated with the coordination of multiple automated agents is far from being solved and requires that humans and their system characteristics be considered early in the design process.

WORKSHOP OBJECTIVES

Space missions to be launched about 15 years from now will utilize state-of-the-art technologies of ten years from today. Automation and robotics technologies have developed at a tremendous rate in the past and are expected to develop at an even greater rate in the future. Since analytical technology forecasting models are lacking, it is extremely difficult to make accurate predictions about the technical capabilities a decade hence. Nevertheless, in planning future manned missions to the Moon and Mars, it is important to make judgments concerning what is and what is not a feasible technology in one to two decades from today. For example, how realistic is it to assume that a particular in-space assembly task will be doable in the year 2002, or that propellant production can occur on the Martian surface in the year 2005?

The Workshop on Robotic Needs for the Human Exploration of the Moon and Mars, held May 10-11, 1988, in Palo Alto, California was designed to assess mission needs, estimate technology adequacy, and appraise the feasibility of required advancements for the OEXP defined advanced missions. Specifically, the workshop utilized experts in robotics to:

1. identify the research and development required for these missions
2. identify the system options
3. identify the potential barriers in the robotics assumptions of the missions
4. identify the areas of potential high return in the missions relative to robotics
5. communicate to the University community the objectives and mission requirements of the Office of Exploration

WORKSHOP METHODOLOGY

Leading experts in robotics were invited from NASA, academia, and industry. In addition there were OEXP and other NASA personnel present to provide information as needed on the assumptions, background and programmatic aspects of space mission planning and implementation.

The workshop began with overview presentations about the Office of Exploration program, the Case Studies, and some of the barrier issues to be addressed.

After the presentation of the background material, to better approach the problems at hand the participants broke into five subgroups. Each of these subgroups forms a representative subsection of the whole of robotics.

1. **Mobility** - wheeled, legged, free flight in space, movement around space structures, hopping, balloon flight, etc.

2. **Manipulation of Objects** - picking up objects (tools,rocks, etc.), manipulating beams in orbit, manipulating samples on a planetary surface, performing electrical and hydraulic connections, bolting, welding, digging, excavating, etc.

3. **Perception and Environmental Interpretation** - object interpretation, fault diagnosis, instrument analysis, understanding of soil conditions, etc.

4. **Navigation and Position Information** - knowing where you are and determining how to get somewhere else by free flight, space structure crawlers, planetary surface mobility, etc.

5. **System Integration** - getting everything to work together, interface issues, overall intelligent control, man-machine interfaces, computing hardware, coordination between intelligent agents, etc.

Each of these subgroups addressed the objectives described above with respect primarily to in-space assembly, planetary exploration, and planetary mining/sitework. The results included:

1. problem definition
   - the domain
   - the mode of operation
   - the options and characteristics
2. potential problems
   - level of complexity
   - research issues
   - design suggestions

Mobility:

Mobility involves more than simple locomotion. It includes also reliability and redundancy, hotel functions and support, stowing and deployment, control and models, platform stability and reaction, stabilization and smooth motion, miniaturization and scaling, and mass volume and power of the locomotion of payloads and the robotic element itself. The options chosen for each of these are highly dependent on the environment, level of structure and the drivers for motion(figure 1). In very general terms, in-space assembly will take place in a more structured environment in a more recipe driven mode, while planetary exploration is in a less structured environment and consequently requires a more event driven mode of operation. Sitework exists at the intersection of assembly and exploration having features of each in both environment and probable mode of operation.

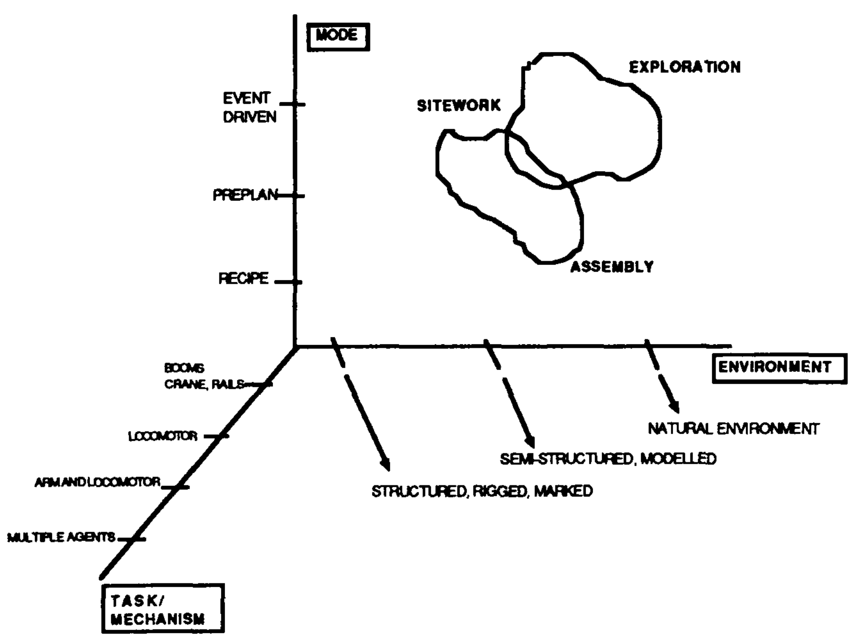


figure 1
adapted from Mobility Subgroup viewgraph

Due to the strong influence of environment and mode of operation, mobility can be broken into four subtasks. These include, in the space environment, structure traversal and flying and docking and , on the planetary surface, exploration and sitework and support (figure 2). It is at this level of detail that the characteristic modes of operation and tasks can be defined(figure 3).

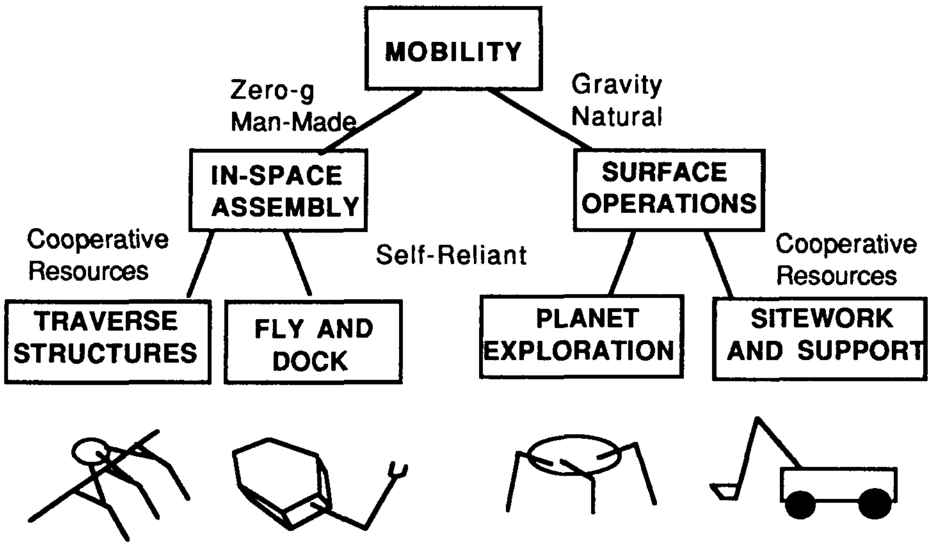


figure 2
adapted from Mobility Subgroup presentation

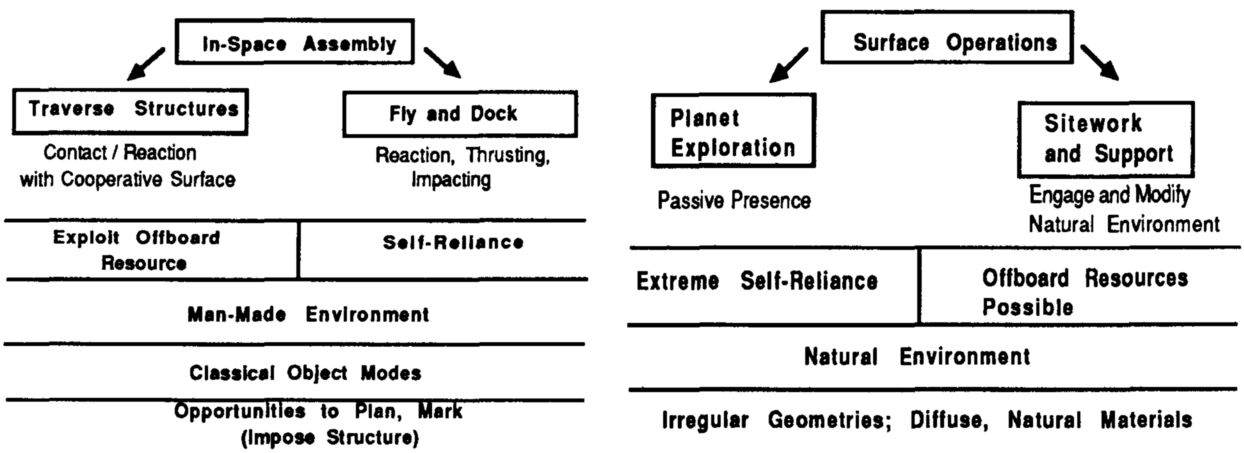


figure 3
adapted from Mobility Subgroup presentation

With the domain of the problem well defined, the subgroup conclusions were alternative and enabling technologies for the different modes of mobility. In summary:

Structure Traversal:

Alternatives | Enabling technologies

- prehensile locomotion
- rails, trolley
- booms, cranes
- serpentine locomotion

- grasping
- proprioceptive contact
- targeting grasp points
- tether management
- connector mating
- bracing by holds or leaning
- actuator conflict, n-chains

Fly and Dock:

Alternatives
- thrusters
- springers, hoppers
- grapple and reel

Enabling technologies
- reaction mass and resupply
- real-time,dynamic interaction for auto-dock
- marking and rigging landing sites
- preclude damage to self and structure
- task model includes all physics
- teleoperations assistance

Planetary Exploration and Site Mobility:

Alternatives
- *rolling, tracks*
- rolling, wheels
- walking
- serpentine
- bounding

Enabling technologies
- generate reaction by looking to terrain
- complete predictive models of intended motion for planning,control, safeguard
- full models of contingency and recovery
- complete data for state model

In addition the mobility subgroup identified the concerns across configurations relative to mobility:

- control of complex plans
- mechanism model of multiple closed chains (walking and prehension)
- degeneration to crippled locomotion - graceful degredation to alternative mode
- hotel and housekeeping
- safeguard at all cost
- predictability of achieving intended locomotion goals
- completeness of diagnostics, contingency
- Swiss army knife versus specialized
- power

Manipulation:

Manipulation includes the devices, the materials, the control, the mechanics, and the planning of the manipulation of objects. These manipulations may be gross movements or, perhaps more interestingly, fine interactions with physical systems. Unlike mobility, the background environment has a very limited effect on manipulation, except in the engineering attributes required of, for example, a space versus a planetary system. Instead, the local background - which can be influenced early on in the design process plays a more important role.

The largest barrier issue for robotic manipulation was perceived to be the importance of teleoperation. "Unless NASA makes a significant and continuing commitment for the development of telerobotic technologies, exploration missions are unrealistic in terms of crew safety, cost, and schedules". The importance on teleoperations comes from the fact that it, in many situations, is the least complex mode of operation while simultaneously providing the greatest range of operational capabilities. Even so, there is currently no centralized commitment to developing telerobotic assembly capabilities.

A second potential barrier issue is that there are elements of environmental engineering that must be included from the very beginning of the design and development process. In fact, " [you] cannot separate the design of the task and the design of the agent (manipulator, human,...)". This influences the requirement for modularity in the structure and the agent as well as such characteristics as commonalty, and verification means,...thus, any manipulation system being examined for future manned space exploration should not be disjoint from the development of space hardware.

Basic research needs for the exploration class missions in the next decade in the area of manipulation were identified as the following areas including modes of operations, mechanical systems, software, and materials.

Teleoperation

- Variable delay times
- Force/non-force feedback
- Tactile/non-tactile feedback
- Visual feedback (resolution, depth, color, contrast)
- Anthropomorphic issues
- Dynamic workcell
- Task scaling
- Simulation

Mechanics

- Contact
- Impact
- Adhesion
- Mating
- Kinematics

Planning

- Uncertainty and unpredictability
- Task characterization
- Sensor based strategy
- Global versus local strategies

- Learning

Devices
- Actuators
- Arms
- End Effectors
- Sensors

Materials
- Joint-seals
- Adhesives
- Lubricants
- Super conductors

Controls
- High degree of freedom manipulators
- Flexible structures
- Small devices coordination
- Non-collocated entities coordination

Perception:

Perception involves the sensing of the environment and objects in the environment by a robot for either local or remote use. The information is used to model the local environment. The data for such modeling may be in the robot computer, may be obtained through observation, or may be a combination of both. It may include geometrical, physical, operational, and semantic aspects. The techniques for sensing and the data required from the sensors varies with the different environments of free space and planetary surfaces. In the free space environment, one can generally assume that there are good models available and that there is adequate instrumentation to establish spatial relationships. On a planetary surface, models are more dependent on precursor science missions and the quality and types of sensors.

The key problems in perception in a free space environment is the inability to quantify performance. On planetary surfaces one has to cope with additional problems, such as: (1) recognition of geological characteristics, surface features for navigation, and task specific features; (2) dynamic acquisition of knowledge; (3) self-understanding and internal calibration; and (4) tasking language for reprogramming.

In addition to these environment specific concerns there are two potential barrier issues in perception. First, there is no general purpose vision system. Second, environmental modeling is required. Research is being performed in the first, however, to perform the Case Studies as prescribed, the vision systems will need to be greatly expanded and refined. The vision system is one form of sensor that can allow man to remotely operate. Operationally, a well-developed vision system will save on cost, increase crew and remote vehicle/robot safety, and allow for a greater scientific return. Environmental modeling is an inexpensive means to prepare the robotic systems being developed. In the area of perception, such models are required in order to develop the software to a sufficient level to achieve the mission goals.

The inability to quantify performance, recognition of features, knowledge acquisition, vision systems, and environmental models are all general areas that require research prior to embarking on these future space missions. These areas have been examined to a sufficient level of detail to identify a few of the more specific enabling capabilities for perception. Table 1 lists the enabling capabilities for various space activities and evaluates them with respect to need and time. Note that in this list, all of the enabling capabilities are required in the next three to five years, and two-thirds are required in the next one to two years to achieve the prescribed missions.

ENABLING CAPABILITIES FOR SPACE ACTIVITIES	FLY/DOCK/ GRAPPLE	IN-SPACE ASSEMBLY	INSPECTION	AUTO-LANDING	SURFACE NAV. EXP-LORATION.	SURFACE OPS
NEW SENSOR TECHNOLOGY	M 2		M 1-3	R 2	R 2-3	R 2-3
SENSOR SUITE CONFIG.			R 1		R 1	R 2
SENSOR MODELING FOR SELF CALIBRATION					R 2-3	R 1-2
PERCEPTION TASK SPEC. LANGUAGE	R 1/2	R 2	M 3		M 1-2	R 3
SENSOR DATA FILTERING TO FOCUS ATTENTION			R 2		R 2-3	R 2-3
ACQUISITION OF NEW MODELS BY ROBOT			M 2-3		R 1-2	R 1-3
RECOGNITION OF KNOWN OBJECTS	R 1/2	R 1/3	R 1/2	R 1/2	M 2-3	R 1/3
MODELING MOTION OF OBJECTS	R 2/3			R 2		
MODELING TERRAIN SHAPE AND TYPE					R 1-2	R 1-3
MONITORING ROBOT TASK EXECITION		M 2			R 1-2	R 2/3
PERCEPTUAL SERVO-LOOPS	R 1/2	R 1/2		R 1	M 1	R 1/2
DISPLAY INTERNAL MODEL TO OPERATOR	R 1-2	R 1	R 2-3		R 2	R 3
COMPUTATION POWER IMPROVEMENTS	M 2	M 2	R 2	M 2	R 2	R 2
RAPID REMOTE REPROGRAMMING			M 2-3		R 2	R 2
PERFORMANCE ASSESSMENT OF PERCEPTION			R 2-3		R 1-2	R 2/3
DIFFICULTY OF DOMAIN ENGINEERING	1 - 2	1 - 2	3	1 - 2		1 - 2

CODES:
R = required
M = might be required

1 = 1-2 years
2 = 3-5 years
3 = science needed first

A/B:
"A" with domain engineering
"B" without

table 1
adapted from perception subgroup presentation

Navigation and Position Information

As with the other subtasks of robotics, navigation is dependent on the environment and mode of operation. The classification spectra across which the differences in navigation range can be listed as follow:

environment:

structured environment (dumb robot)	------->	unstructured environment (intelligent robot)

mode of operation:

process-based (roach)	------->	map-based (cat)
teleoperated	------->	local control
construction tasks	------->	roving tasks
surface operations (gravity)	------->	3-space (micro-g)

figure 4
adapted from Navigation subgroup presentation

For navigation, onboard computing power, in terms of fast array processors capable of putting data structures on ill-structured domains and developing mechanisms for knowledge acquisition and model revision, will need to be available.

The near term navigation capability development tasks should concentrate on:

- integrating route planning with large terrain databases, manipulator planning, and science planning
- execution monitoring
- uncertainty information
- expectation generation
- feature information
- automation of landmark selection
- automation of planetary surface navigation
- navigation aids for teleoperation

As a summary, table 2 provides evaluation results, where the extremes of some parameter spectra are evaluated with respect to various operational situations on planetary surfaces and in three-dimensional space. The associated key symbols give an indication about relevance, complexity, and importance of the chosen parameters in various domains.

activity/task / environment/operation	"non-teleoperable" surface roving	"teleoperable" surface roving	surface construct.	3-space roving	3-space constr.
structured environment	NA	NA	V 7	V 5	V 8
unstructured environment	V 4	V 2	V 4	V 5	V 6
process-based	G1,E,8	G1,E,6	G1,E,5	G1,E,6	G1,E,7
map-based	G1,E,3	G1,E,2	G1,E,5	G1,E,5	G1,E,7
teleoperation	NA	G1,E,3	G1,E,7	G1,E,4	G1,E,5
local control	V 5	V 3	V 7	V 3	V 5

KEY:
V = vital
E = enhancing
# = complexity,1 low -> 10 high

G1 indicates that one of the items marked G1 in a given vertical column is required

table 2
adapted from navigation subgroup presentation

System Integration:

The Systems Integration subgroup attempted to structure the problem along dimensions of location: where the primary system operates, process: what the system does, and agent: what type of system it is. Parameters entering these considerations were system availability, response time, number of repetitive actions, communication delay, completeness and certainty of environment determination, degree of the structure, and to some degree, cost.

Basic problems in robotics system integration for exploration-type missions and *design guidelines for those areas requiring research:*

- functional organization
- structural organization
- function to structure mapping
- communication (machine-machine. man-machine, man-man)
  - bandwidth
  - protocols (data, images, video, actions, goals, plans)
- world modeling, representations
  - geometry, image, physical, goals, plans, state, actions, model building from sensed information, parameter identification
- intelligent control
  - definition
  - coordination (sequential, parallel, shared - multi-level), division of primitive operations
  - mode switching
  - strategy building
- design for assembly, maintenance, evolution
- tolerances/sensing uncertainty, control errors/strategy relation
- scale

- software: realtime, distributed, large, production environment, debugging, maintenance, AI support

A distillation of these issues to eight major research issues, which were evaluated with respect to the application domains in-space assembly, planetary exploration, and lunar mining, are presented in Table 3.

	in-space assembly	planetary exploration	lunar mining
human/machine integration	E 1	N 1	N 1
design for assembly and maintenance	V 2	V 2	E 2
intelligent communications	E 3	N 3	N 3
tele-automation	H 2 (colocated) E 3 (ground based)	V 3	N 2
deep modeling of situations & actions	E(physics) 2 (?)	V 3	N 2
architectures	V 3 (?)	V 2(?)	V 2
on-board computation	V 1 (?)	V 2(?)	E 2 (?)
manual & automatic lifecycle evolution	H 3	E 3	N 3

KEY:
1 = 1-2 years
2 = 3-5 years
3 = 6-10 years

V = vital
H = enhancing
N = not applicable

table 3
adapted from system integration subgroup presentation

Conclusions

One of the most important conclusions that came from the different subgroups was the identification of those areas of robotics that need top be at a greater level of development in the next decade for the successful embarkment upon manned exploration class missions as defined by the OEXP Case Studies. Specifically these technologies are required for in-space assembly and planetary operations including mining and exploration. Some of the robotic technologies that will be required to be available within the next one-two years, in preparation for these future manned space missions include are the following:

- on-board computation improvements
  inspection
  fast flight processors
  rapid remote programming
- sensors
  suite configuration - inspection, navigation, surface operations
  new sensor technology - auto-landing,navigation, operations

- navigation aids for teleoperations (LEO)
- techniques for structuring the 3-space domain, i.e. bar codes
- telerobotics
  force/nonforce feedback
  visual fidelity (resolution, depth perception, colour, contrast)
  task scaling
- simulation (LEO assembly)
- position calibration via navigation aids, grip locations, docking locations, part identification
- reliability, redundancy, predictability
- control/models
- rapid remote programming
- hotel and housekeeping, safeguard at all costs (MRSR)

Additional areas that will require development within the next three to five years are as follow:

- artificial intelligence:
  integration of route and manipulator planning
  performance assessment, inspection, execution monitoring
  expectation generation (in-space assembly)
  mechanism for knowledge acquisition and model revision
  planning: uncertainty and unpredictability, task characterization, sensor based strategy, global vs local strategies, learning
  control of: high DOF manipulators, flexible structures, small devices(coordination), non-collocated entities, complex plans
  automation of landmark selection, planetary surface
  automation of planetary surface navigation, including planning, control, safeguard, contingency and recovery
  Swiss army knife vs specialist robotics
  integration of route and manipulator planning
  integration of route and science planning
  architectures (planetary surface)
  perception task specification language
  perceptual servo-loops
  execution monitoring (in-space assembly)
  expectation generation
  complete task decomposition/execution from hi-level prescription
  recognition of known objects

- sensor technologies:
  sensor data filtering to focus attention (LEO)
  sensor modeling for self calibration
  proprioceptive contact

- models:
  modeling terrain shape and type
  mechanism model of multiple closed chains
  non-semantic models of the terrain

- engineering:
  design for assembly and maintenance
  display internal model to operator (i.e. heads-up display)
  array flight processors (in-space assembly)
  devices: actuators, arms, end effectors, sensors

materials: joint seals, adhesives, lubricants, superconductors
mechanics: contact, impact, adhesion, mating, kinematics
miniaturization/scaling

- telerobotics:
telerobotics - variable time delays
telerobotics - anthropomorphic issues
telerobotics - tactile/nontactile
telerobotics - dynamic work cell
tele-automation

These two lists of cross-scenario robotics requirements within the next decade indicate 1)the large number of areas that require further development in order to carry out scenarios such as those presented by the Office of Exploration, 2)some of the broad areas that should be addressed.

Other, broad, conclusions arrived at by the participants of the OEXP Robotics workshop are as follow:

1. The in-space assembly process is possible with reasonable extensions of technology - depending on appropriate design and structure (i.e., size, interconnects, etc.) of the components. There was a strong feeling that with appropriate care taken in the design of components and the assembly process, the tasks are doable with robotic technology. One can not separate the task, the component design, and the design of the effecting agent. It is unreasonable to assume that robots in space will take over all assembly tasks currently done on the Space Shuttle at KSC. However, robots may ease the assembly burden for well designed processes.

2. All parts, which must be manipulated, should be designed robot friendly. For example, all parts should have clear machine readable labels, and should be designed for easy robot grasping. Designers should strife for compatibility between robot friendly and EVA friendly designs.

3. Design of the task/design of the agent should be performed simultaneously

3. Technology options for the Office of Exploration missions

4. Greater computing power in space; faster, allowing rapid remote programming

5. Teleoperations will be required due to the crew safety, cost, and schedules

The Robotics Workshop also opened communications and interchanges with the university community and industry and established a broader understanding by the participants of the OEXP missions.

An Overview of On-Orbit Assembly/Servicing/Check-out Issues
Stephen Katzberg, LaRC

MANNED MARS/LUNAR BASE INITIATIVES

AN OVERVIEW OF ON-ORBIT ASSEMBLY/SERVICING/CHECKOUT ISSUES

PRESENTED TO

OFFICE OF EXPLORATION WORKSHOP

ON

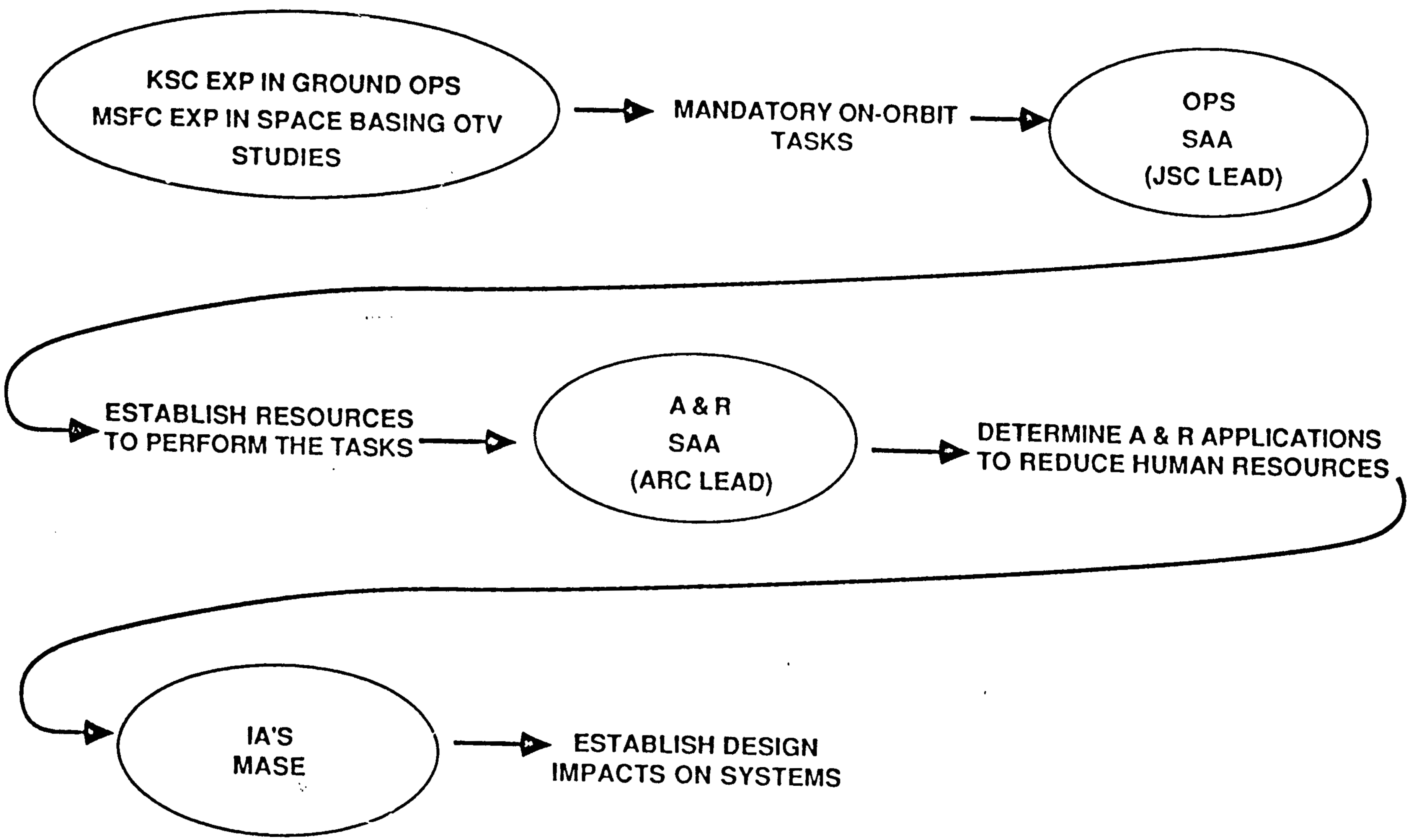
ROBOTIC NEEDS FOR THE EXPLORATION OF THE MOON AND MARS

MAY 10, 1988

DR. STEPHEN J. KATZBERG
LANGLEY RESEARCH CENTER

LaRC Transportation Node Integration Agent

ON-ORBIT ASSEMBLY/CHECKOUT/SERVICING REQUIREMENTS

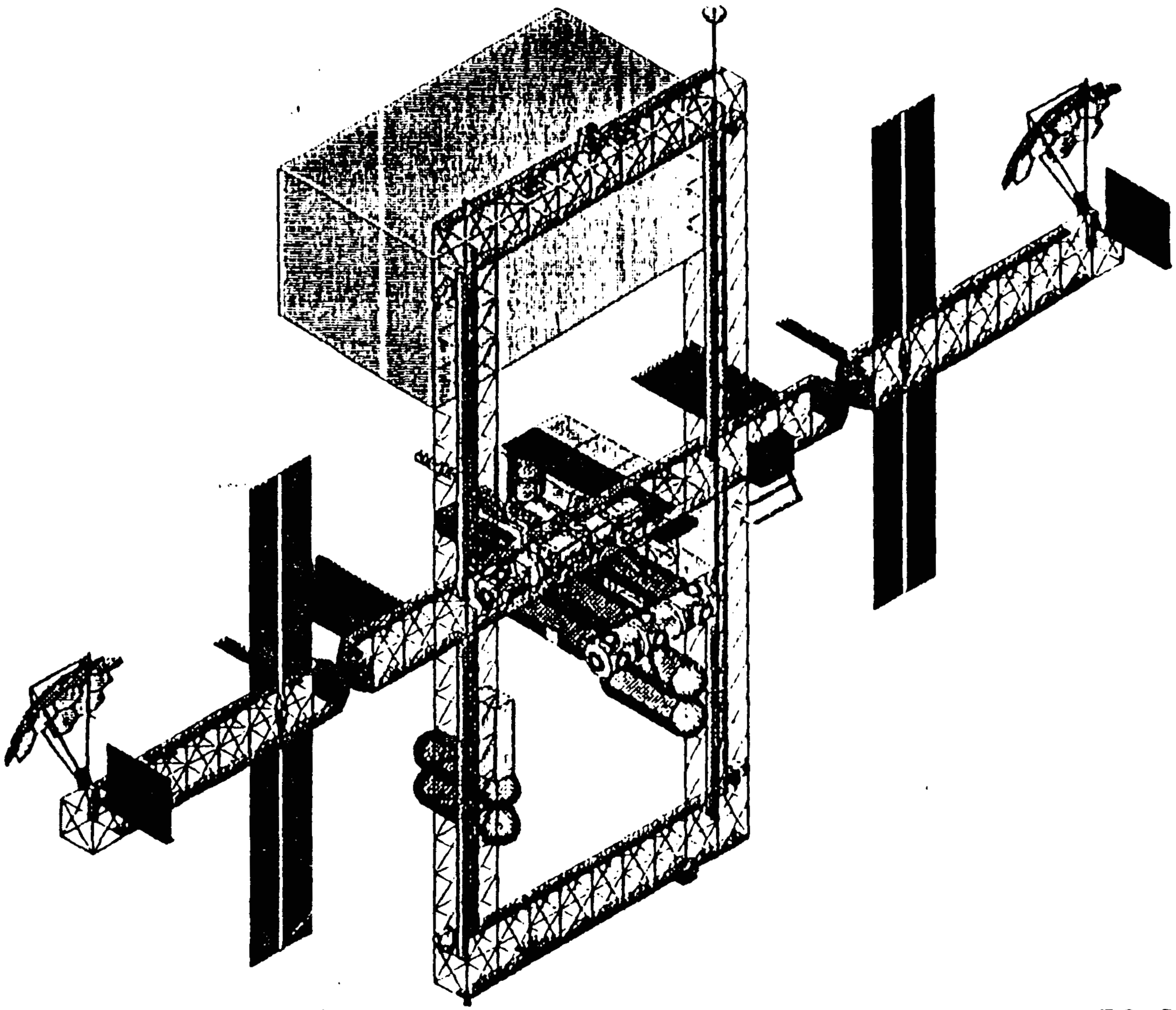


VEHICLE ACCOMMODATION OPTIONS

OPTION #1:	STATION BASED	- ALL VEHICLE ACCOMMODATIONS BASED ON STATION
OPTION #2:	STATION BASED w/PTF	- VEHICLE ASSEMBLY AND REFURBISHMENT FACILITY IS ON-STATION. PROPELLANT IS LOCATED ON A CO-ORBITING PROPELLANT TANK FARM (PTF)
OPTION #3:	TRANSPORTATION DEPOT (MAN TENDED)	- VEHICLE ACCOMMODATIONS ARE KEPT ON A CO-ORBITING PLATFORM, BUT CREW IS BASED ON STATION
OPTION #4:	TRANSPORTATION DEPOT (PERMANENTLY MANNED)	- A SEPARATE FACILITY IS PROVIDED FOR VEHICLE AND CREW

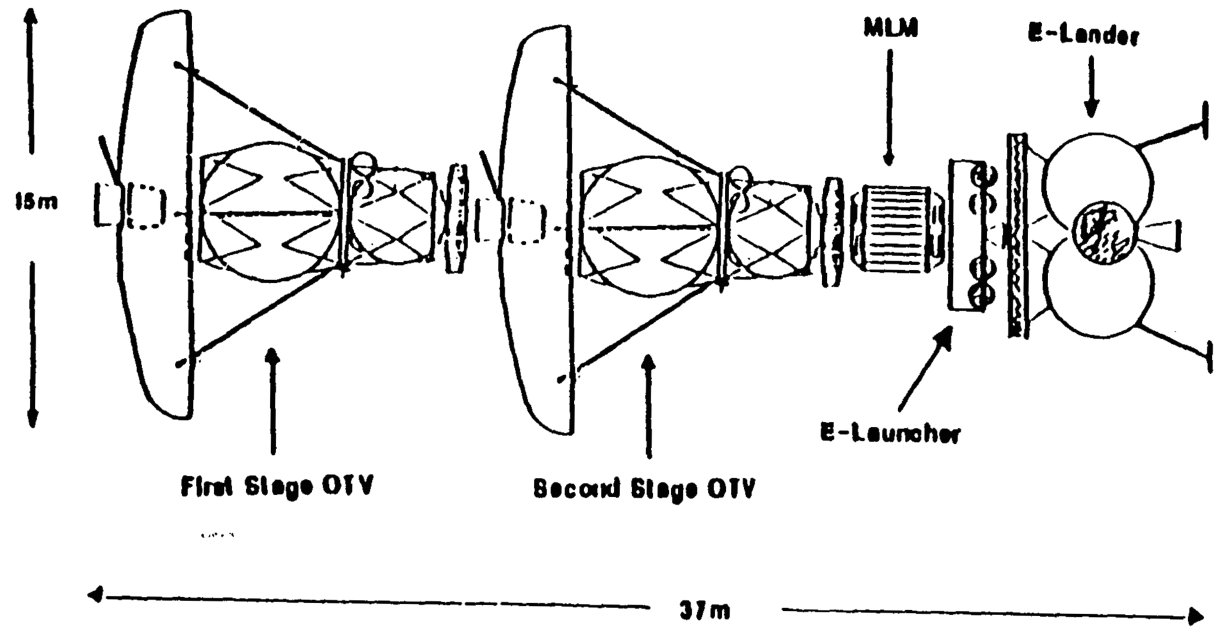
Lunar Base Accommodation Study

STATION CONFIGURATION (OPTION 1A)



Lunar Base Accommodation Study

LUNAR VEHICLE STACK

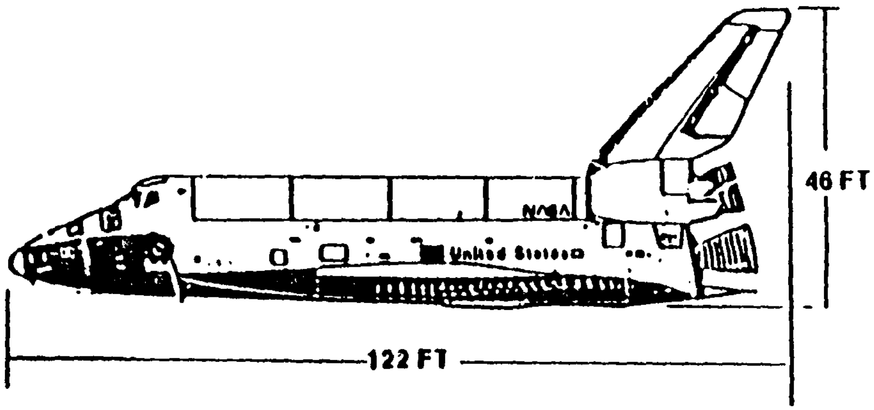


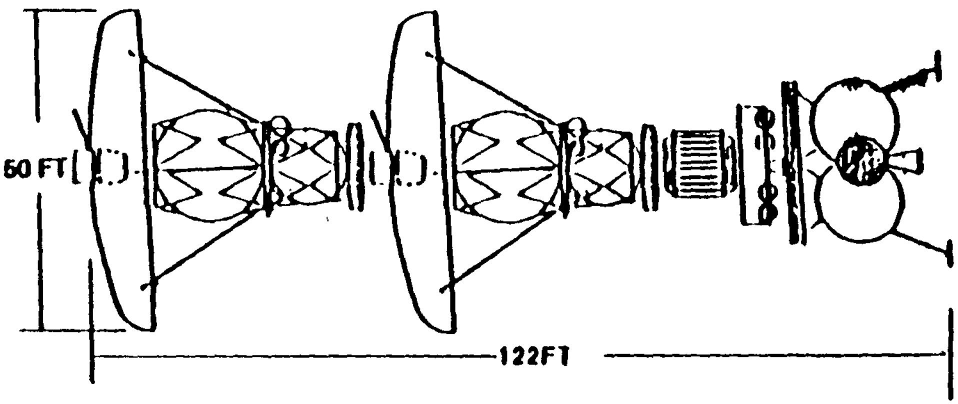
LUNAR VEHICLE STACK WEIGHT (LBS.)

FIRST STAGE	= 87,000
SECOND STAGE	= 87,000
MLM	= 13,200
EXPENDABLE LANDER	= 38,200
EXPENDABLE LAUNCHER	= 16,720
TOTAL VEHICLE WEIGHT	= 242,200

Lunar Base Accommodation Study

VEHICLE COMPARISON

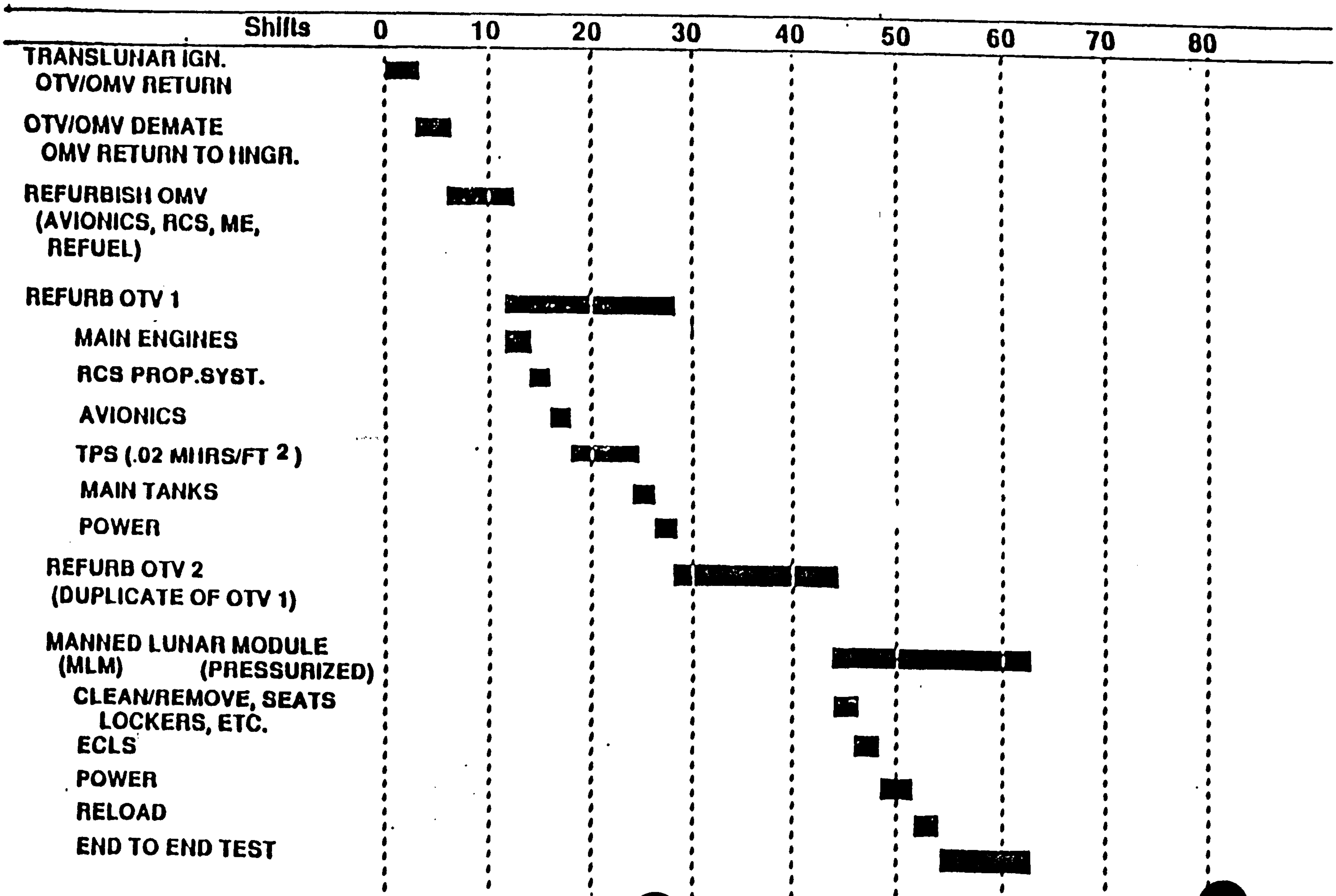




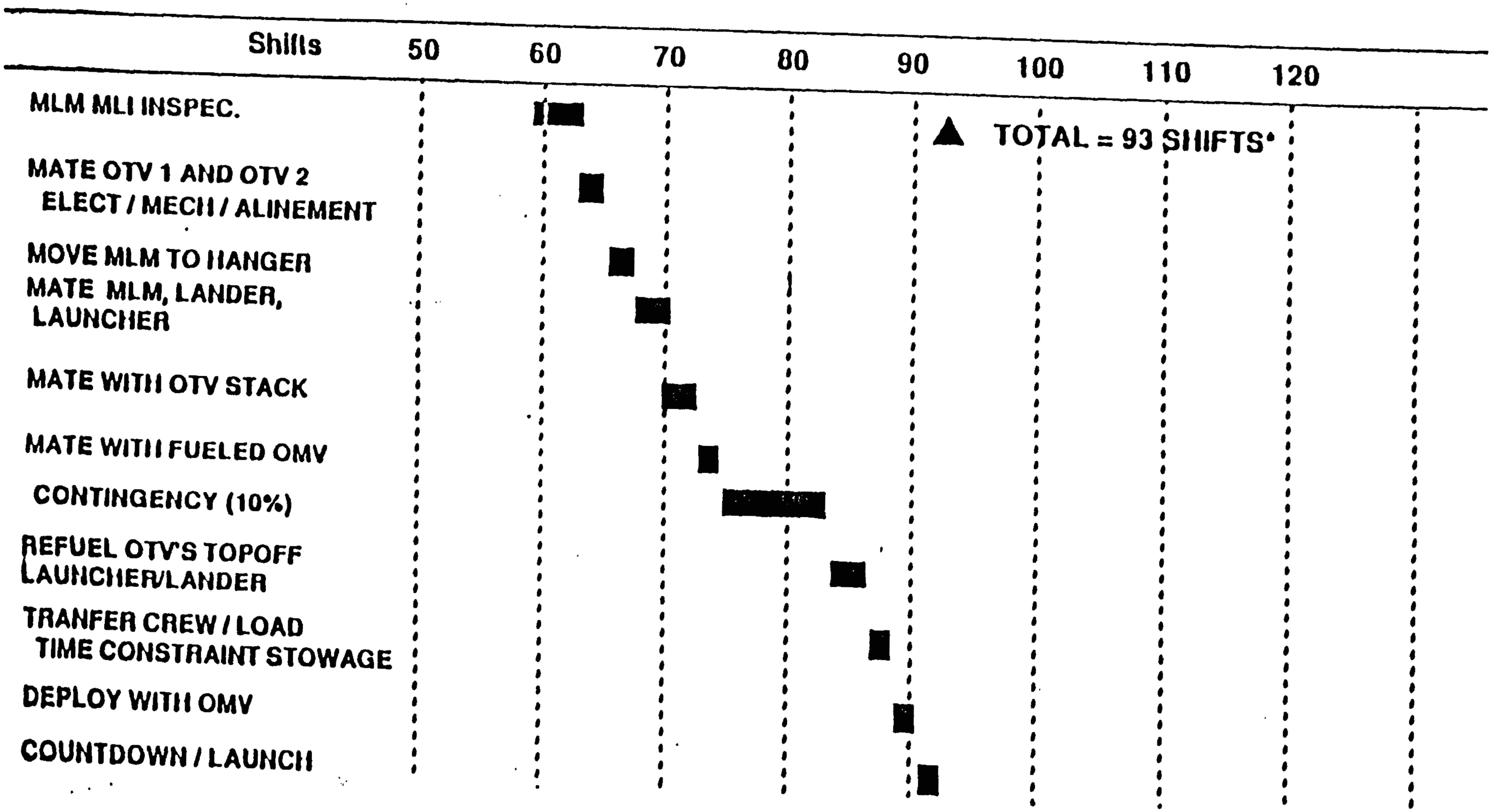
SPACE SHUTTLE		MANNED LUNAR VEHICLE	
LENGTH	122 FT	LENGTH	122FT
WIDTH	78 FT	WIDTH	50 FT
HEIGHT	46 FT	DRY WEIGHT	56 K#'S
DRY WEIGHT	165 K#'S	EARTH DEP. WEIGHT	248 K#'S
ON-ORBIT WEIGHT	230 K#'S	ENGINE SYS'S	
ENGINE SYS'S		SPACE PROP. SYS	LOX/LH
SSME	LOX/LH	RCS	MMH/NTO
OMS	MMH/NTO	E-LANDER	LOX/LH
RCS	MMH/NTO	E-LAUNCHER	MMH/NTO
SUBSYSTEMS		SUBSYSTEMS	
ECLSS		ECLSS	
GN&C		GN&C	
C&T		C&T	
EMP		EPS	
DMS		DMS	
EVA		EVA	

Lunar Base Accommodation Study

SINGLE SITE SERIAL SERVICING



SINGLE SITE SERIAL SERVICING CONTINUED



* SHIFTS EQUALS A WORKING DAY FOR A TEAM OF 3 CREW

MISSION/VEHICLE CHARACTERISTICS

MANNED MARS - SPLIT SPRINT MISSION CLASS

- MULTIPLE VEHICLES, MULTIPLE MISSION
- REQUIRES CONTINUOUS ON-ORBIT SUPPORT
- ON-ORBIT ASSEMBLY/SERVICING/REFURB.
- AEROBRAKING FOR MARS/EARTH CAPTURE
- REQUIRES HLLV
- REQUIRES MANNED OMV-TYPE VEHICLE

MANNED LUNAR BASE MISSION

- MULTIPLE VEHICLES, MULTIPLE MISSIONS (HIGH FLIGHT RATES)
- REQUIRES CONTINUOUS ON-ORBIT SUPPORT
- ON-ORBIT ASSEMBLY/SERVICING/REFURB.
- AEROBRAKING FOR EARTH CAPTURE
- REQUIRES HLLV
- REQUIRES MANNED OMV-TYPE VEHICLE

SIGNIFICANT NEEDS/ISSUES ASSOCIATED WITH MANNED MARS/LUNAR BASE MISSIONS

- BETTER DEFINITION OF THE MARS AND LUNAR VEHICLES
- BETTER DEFINITION OF CREW ACTIVITIES (IVA, EVA) ASSOCIATED WITH ON-ORBIT ASSEMBLY/CONSTRUCTION TASKS
- BETTER DEFINITION OF CREW/ROBOTICS AND CREW/AUTOMATED SYSTEMS ROLES AND INTERFACES
- DEVELOPMENT OF PROCEDURES FOR IN-SPACE PROCESSING OF HAZARDOUS (WET) SYSTEMS
- DEVELOPMENT OF SPACE BASED DIAGNOSTICS/PROGNOSTICS CAPABILITY
  - IN-SPACE SYSTEMS CHECKOUT
  - ON-BOARD/ORBIT DECISION MAKING FOR SAFE SYSTEMS OPERATIONS
  - SYSTEMS HEALTH PREDICTION/STATUS
- A MANNED OMV-TYPE VEHICLE DESIGN CAPABLE OF:
  - HANDLING MASSES IN EXCESS OF 250,000 POUNDS
  - PRESSURIZED TRANSFER OF UP TO SIX CREWMEN BETWEEN THE SPACE STATION AND THE CO-ORBITING FACILITY IN LESS THAN 2 HOURS TRANSIT TIME

Automation/Robotic Challenges Driven by LEO Transportation Requirements for Leadership Initiatives

Harvey Feingold, SAIC

AUTOMATION/ROBOTIC CHALLENGES DRIVEN BY LEO TRANSPORTATION REQUIREMENTS FOR LEADERSHIP INITIATIVES

Presented by

Dr. Harvey Feingold
Science Applications International Corporation
Schaumburg, IL 60173

to

OFFICE OF EXPLORATION'S WORKSHOP

ON

ROBOTIC NEEDS FOR EXPLORATION OF MOON AND MARS

MAY 10, 1988

INTRODUCTION

- **PURPOSE** — TO IDENTIFY EARLY AUTOMATION & ROBOTICS DEMANDS DRIVEN BY THE LEO TRANSPORTATION REQUIREMENTS/ACTIVITIES ASSOCIATED WITH THE ALTERNATIVE LEADERSHIP SCENARIOS, I.E.,
    - PHOBOS EXPEDITION
    - MARS EXPEDITION
    - LUNAR OBSERVATORIES
    - EVOLUTIONARY EXPANSION

- **APPROACH** — DEFINE THE RANGE OF TRANSPORTATION ACTIVITIES REQUIRED WITHIN EACH SCENARIO (E.G., DOCKING, ASSEMBLY, FUELING, CHECKOUT, DEPLOYMENT, RECOVERY, ETC.) AND INFER THE A/R CAPABILITIES WHICH WILL ENABLE/ENHANCE THOSE ACTIVITIES CONSISTENT WITH THE IMPOSED SCENARIO GROUNDRULES AND CONSTRAINTS

- **CAVEAT** — SINCE CONSIDERABLE LATITUDE STILL EXISTS IN HOW THESE SCENARIOS ARE TO BE IMPLEMENTED, THE PRESENTED INFORMATION IS NECESSARILY JUDGEMENTAL AND SUBJECTIVE; IT SHOULD THEREFORE BE TREATED AS A CATALYST FOR DISCUSSIONS IN THE WORKSHOP RATHER THAN AS A DEFINITION STATEMENT OF A/R REQUIREMENTS

LEO TRANSPORTATION REQUIREMENTS

	PHOBOS EXPEDITION	MARS EXPEDITION	LUNAR OBSERVATORIES	EVOLUTIONARY EXPANSION
1. RENDEZVOUS/DOCKING/MATING	●	●	●	●
2. LARGE-SCALE DEPLOYMENT	◒	◒		●
3. LARGE-SCALE VEHICLE ASSEMBLY		○		●
4. CHECKOUT/TEST	●	●	●	●
5. FUELING/REFUELING	●	●	◒	●
6. PROPELLANT FARM				●
7. CREW RECOVERY	●	●	●	●
8. REFURBISHMENT/REPAIR			●	●
9. SPACE-BASED PROPULSION SYSTEMS				●
10. SPACE-BASED PILOTED VEHICLES			●	●
11. SPACE-BASED CARGO VEHICLES				○
12. HIGH MASS THROUGHPUT	○	◒	○	●
13. HIGH TRAFFIC THROUGHPUT		◒	○	●

KEY:
POSSIBLY ○
PROBABLY ◒
DEFINITELY ●

EARLY AUTOMATION/ROBOTICS DEMANDS (LEO TRANSPORTATION)

	PHOBOS EXPEDITION	MARS EXPEDITION	LUNAR OBSERVATORIES	EVOLUTIONARY EXPANSION
1. RENDEZVOUS/DOCKING/MATING	●	◒	◒	◒
2. LARGE-SCALE DEPLOYMENT	●	●		◒
3. LARGE-SCALE VEHICLE ASSEMBLY		◒		◒
4. CHECKOUT/TEST	●	●	●	●
5. FUELING/REFUELING	●	●	●	●
6. PROPELLANT FARM				●
7. CREW RECOVERY	◒	◒	◒	●
8. REFURBISHMENT/REPAIR			○	●
9. SPACE-BASED PROPULSION SYSTEMS				●
10. SPACE-BASED PILOTED VEHICLES			○	◒
11. SPACE-BASED CARGO VEHICLES				○
12. HIGH MASS THROUGHPUT	○	○	○	●
13. HIGH TRAFFIC THROUGHPUT		◒	◒	●

KEY:
LOW ○
MODERATE ◒
HIGH ●

LEO OPERATIONS - HUMAN EXPEDITION TO PHOBOS

OPERATION	A/R APPLICATIONS	REQUIRED IOC	SUPPORTING INTRASTRUCTURE
RENDEZVOUS/DOCKING CARGO VEHICLES AND PROPELLANT TANK	TELEOPERATIONS ALIGNMENT SENSORS ADAPTIVE CONTROL	2001	GROUND CONTROL
SPACECRAFT FUELING	TELEOPERATIONS, GAUGING & MONITORING SYS. ROBOTIC MANIPULATION FLUID TRANSFER TECH.		GROUND CONTROL SPACE STATION, OR STS
AEROBRAKING DEPLOYMENT/ASSEMBLY	FULLY AUTO. DEPLOYMENT TELEOPERATED OR EVA ASSEMBLY		SPACE STATION OR STS
VEHICLE CHECKOUT/TEST	DIAGNOSTIC/TEST EQMT. EXPERT SYSTEM SOFTWARE		SPACE STATION GROUND CONTROL
EARTH-MARS TRANSFER	AUTOMATIC SEQUENCING		GROUND CONTROL
SIMILAR OPERATIONS FOR PILOTED VEHICLE PORTION OF MISSION	SEE ABOVE	2002	GROUND CONTROL CREW ASSIST
OMV RENDEZVOUS/DOCKING WITH RETURNING SPACECRAFT	TELEOPERATIONS SMART SENSORS ADAPTIVE CONTROL	2003	SPACE STATION
TRANSPORT TO SPACE STATION	REMOTE CONTROL		SPACE STATION

LEO OPERATIONS - HUMAN EXPEDITIONS TO MARS

OPERATION	A/R APPLICATIONS	REQUIRED IOC	SUPPORTING INTRASTRUCTURE
RENDEZVOUS, DOCKING, BERTHING OF VEHICLES AND PROPELLANT TANKS AT LEO TRANSPORTATION NODE		2003	LEO NODE
MATE CARGO VEHICLE AND ESCAPE STAGE			
SPACECRAFT FUELING			
SPACECRAFT/AEROBRAKING ASSEMBLY			
VEHICLE CHECKOUT/TEST			
EARTH-MARS TRANSFER			
SIMILAR OPERATIONS FOR PILOTED VEHICLE PORTION OF MISSION		2004-2005	
OMV RENDEZVOUS/DOCKING WITH RETURNING SPACECRAFT		2006	
TRANSPORT TO LEO NODE			

LEO OPERATIONS - LUNAR OBSERVATORIES

OPERATION	A/R APPLICATIONS	REQUIRED IOC	SUPPORTING INTRASTRUCTURE
RENDEZVOUS, DOCKING, BERTHING OF VEHICLES AND PROPELLANT TANKS AT LEO TRANSPORTATION NODE		2000	LEO NODE
MATE UNMANNED/MANNED LUNAR LANDER WITH OTV			
FUEL VEHICLES			
VEHICLE CHECKOUT/TEST			
EARTH-MOON TRANSFER			
OMV RENDEZVOUS/DOCKING WITH RETURNING SPACECRAFT			
TRANSPORT TO LEO NODE			
REFURBISH/REPAIR SPACECRAFT			

LEO OPERATIONS - EVOLUTIONARY EXPANSIONS

OPERATION	A/R APPLICATIONS	REQUIRED IOC	SUPPORTING INTRASTRUCTURE
RENDEZVOUS, DOCKING, BERTHING OF LUNAR AND CARGO VEHICLES AT LEO TRANSPORTATION NODE		2000	LEO NODE AT SPACE STATION
RENDEZVOUS/DOCKING OF STV'S AT PROPELLANT DEPOT			
STV FUELING			
STV TRANSFER AND MATING WITH SPACECRAFT			
VEHICLE CHECKOUT/TEST			
EARTH-MOON TRANSFER			
OMV RENDEZVOUS/DOCKING WITH RETURNING SPACECRAFT			
TRANSPORT TO LEO NODE			
SIMILAR OPERATIONS FOR MARS MISSIONS WITH ADDITIONAL REQUIREMENT FOR ASSEMBLY OF MARS VEHICLE AND AEROBRAK!NG		2009-2010	

KEY TRANSPORTATION RELATED AUTOMATION/ROBOTICS ISSUES?

- DOES EARLY IMPLEMENTATION WITHOUT SIGNIFICANT SPACE STATION SUPPORT (I.E., THE PHOBOS EXPEDITION) ACTUALLY DRIVE ERALY A/R REQUIREMENTS AS A SUBSTITUTE FOR SIGNIFICANT MANNED PRESENCE DURING LEO ASSEMBLY/CHECKOUT?

- IN A NUMBER OF CASES AUTOMATION VERSUS TELROBOTICS POSE ALTERNATIVE SOLUTIONS TO THE SAME REQUIREMENT (E.G., RENDEZVOUS/DOCKING/MATING); WHAT CRITERIA SHOULD BE USED TO CHOOSE THE PREFERRED APPROACH?

- A/R CAN HAVE AN IMPACT BOTH ON COST AND RELIABILITY IN ENABLING/ENHANCING INDIVIDUAL SCENARIOS; WHERE SHOULD THE EMPHASIS IN EARLY A/R TECHNOLOGY DEVELOPMENT BE PLACED, ON COST BENEFITS OR RISK BENEFITS (IF BOTH CAN'T BE DONE SIMULTANEOUSLY)?

- TWO TRANSPORTATION ACTIVITIES WHICH ALMOST CERTAINLY REQUIRE A/R TECHNOLOGY (I.E., ENABLING) ARE ORBIT FUELING AND CHECKOUT/TEST; ARE THERE OTHERS WHICH ARE NOT JUST ENHANCED, BUT ENABLED BY A/R CAPABILITIES?

In-Situ Resource Utilization
John Alred, JSC

OFFICE
OF
EXPLORATION

A&R
WORKSHOP

NASA Ames Research Center

John W. Alred
Integration Agent,
Planetary Surface Systems

May 10, 1988

Proposal For a Remotely Manned Space Station
Marvin Minsky

PROPOSAL FOR A REMOTELY MANNED SPACE STATION

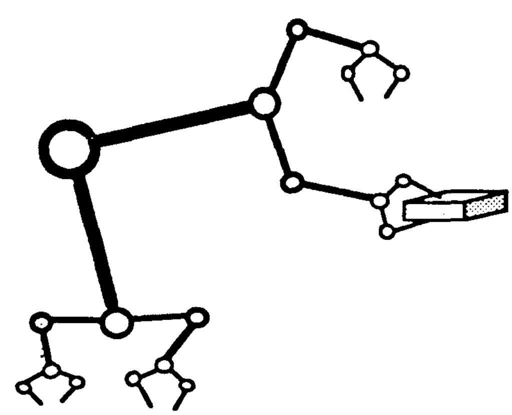
Marvin Minsky, MIT

The United States is in trouble in space. Budgets are up, transportation is down, and our systems have turned into complex, expensive dinosaurs that require so many years of planning in advance that they cannot respond to new opportunities.

What went wrong? The trouble was in thinking that we had only two alternatives: either having people working in space - or using self-contained machinery. Both choices led to spacecraft that were expensive, rigid, and hard to repair - and required so many years of planning in advance that they could not respond to new opportunities. Fortunately, there is another alternative that accomplishes more at far less cost.

Design a space station made of modular parts like those of an Erector set.
Develop mechanical hands that can be remote-controlled from distant locations.
Train earth-based workers to build the station in space - using simulators.
Send hands into orbit with a stockpile of parts.
Use people working on earth to assemble the actual space station .
Finally, assemble life-support systems and living quarters.
Populate them with human scientists and explorers.

The initial cargo will include systems for power, propulsion, and communication that use conventional components of established reliability. The novel aspect is to include three remote-controlled mobile mechanical manipulators - call them "telerobots" - along with a stockpile of modular parts. Because the telerobots themselves are modular, they will be able both to build new structures and to maintain and repair other telerobots .

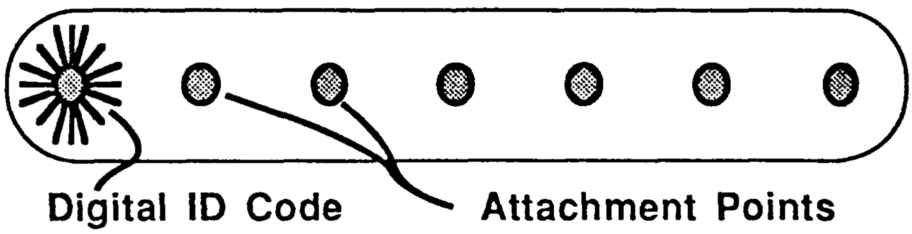


The initial cost of a "remotely manned" space station could be very modest. Safety and life-support demands are minimal until we send up human scientists and explorers - after first using the telerobots to construct and test life-support systems and living quarters. The technology can all be developed in a very few years. In less than a decade, the project would be years ahead of what is being planned today.

INGREDIENTS OF A REMOTELY MANNED SPACE STATION

THE TELEROBOTS. The station is equipped with three or more remote-controlled mechanical hands that can move themselves from place to place. The telerobots are controlled by human operators who have donned "control suits" that translate movements of the operator into corresponding telerobotic acts. Each telerobot, in turn, provides its operator with a sense of "telepresence" by returning feedback in the forms of visual, auditory and tactile sensations. The first human operators will work on Earth but can later be based on the station itself or on the surface of Mars or the Moon. The mechanical hands must be versatile enough to assemble and maintain most space station components and we must also be sure that any telerobot can disassemble and repair another telerobot.

THE MICROMODULES: To insure that the telerobots will be able to manipulate components of the space station, we adopt a policy of using modular components wherever possible. In effect, the entire station should be composed mainly of elements in the style of those of construction toys like ©Erector, ©Meccano, or ©TinkerToy. Every exposed surface should be studded with "Attachment-points" at frequent intervals, and these should be carefully engineered to easily mate with suitable Connector devices so that they can be used, not only for assembling larger structures, but also to enable the telerobots to move from place to place. Each Attachment-point is also labelled with a unique, easily machine-readable identification mark to enable a computer system to keep track of all spaceborne materials.



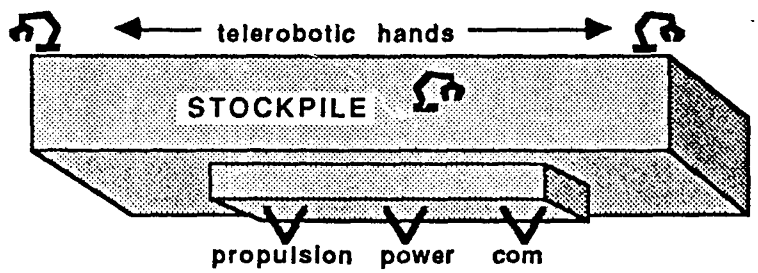
There are many advantages to adopting this "micromodular " policy. It will enable us to re-use the same parts, at different times, for many different purposes. Because it makes it much easier to represent both the structures themselves and the skills involved in assembling them, it will not only reduce the total inventory mass of material and spare parts, but simplify simulation, assembly, and design .

THE SIMULATOR: The micromodular policy will make it easier to develop a computer system for simulating the entire station in great detail, not only under present and actual conditions, but also under hypothetical conditions proposed from any terminal. That Simulator would have access to a data base that normally knows the locations of every component and its Attachment-points. The data base should also know as much as possible about physical states of every part: stresses, velocities, temperatures, currents, etc. This data base could then be used to enhance the telerobots with supervisory control and, eventually, to automate whichever operations we can reliably program. The data base can help in assembling new systems, by determining which materials are available - or can be borrowed from other systems that can tolerate the resulting down-time. Thus, such a system would at first operate under direct human control, but would be open to advances in automation as we develop and upload more advanced automatic systems.

STAGES OF DEVELOPMENT

Phase I: Before initial launch will be a period for developing the ingredients of the project. During this period, many things can be done concurrently. Develop the modular components and connectors, for space structures as well as those for the telerobots. Develop the telepresence communication systems for the sensors and actuators. Build one Telerobot and use it to assemble the others. Start training workers on Earth in the operational and maintenance skills for constructing various structures and assemblies.

Phase II. The initial launch will place an unmanned station in LEO, equipped with conventional packages for power, orbital maneuver propulsion, and satellite communications. The principal payload will consist of three telerobots and a stockpile of parts. All further additions will be assembled by the telerobots, which are controlled by workers on the ground to reconfigure or extend structures whenever necessary. Because it is both desirable and feasible to move very slowly at first, the initial configuration will have very modest power and communication requirements. Less than 100 watts might suffice.



Each hand can repair the others.
Controlled from distant locations.
Tolerate heat, cold, vacuum, and loneliness.

A useful early operation would be to assemble a large experimental antenna, and to practice using a Telerobot to steer it. If successful, this would increase the teleoperation feedback bandwidth for subsequent operations.

Phase III. Ship additional parts and supplies, still using remote control from the earth to assemble and operate instruments for scientific research. The next exercises should include: Practice Telerobot dissassembly and repair. Experiment with battery transfer operations. Launch materials for life-support systems and living quarters. Assemble and test these larger structures. Launch materials for commercial prototypes. Experiment with tethered moderate-gravity habitat and free-flying operations.

Phase IV. Begin larger scale operations, including human habitats and commercial operations. Introduce semi-automatic assembly operations, using supervisory control, CAD, and planning programs to design new structures. Assemble and test a pressurized human habitat with life-support equipment. Proceed to send up human scientists and explorers We can use similar procedures to assemble a lunar base and an interplanetary exploration vessel.

By adopting this strategy for going into space, we can prepare each expedition by using earth-based workers to do what in space would be much more dangerous, expensive, and difficult. The result would be savings in safety, cost, and time.

THE CONCEPT OF MICROMODULARITY

We use here the term "micromodule" for the idea that, wherever possible, every structure should be composed of standardized objects. (In contemporary space station jargon, the term "module" is sometimes applied to any self-contained system - even one as large as an entire space shuttle payload.) The micromodules will usually resemble the components of children's' mechanical construction sets, and the policy would be to assemble larger structures from them wherever there is no critical reason not to. Even a simple box might be composed of many plates and angle beams. Sometimes this policy leads to increase in mass because of needing more fasteners. However, it will also lead to reductions in mass when the parts of systems are reclaimed for other applications. In any case, every micromodule should be convered with conveniently located "Attachment-points", each marked with a unique and precisely located optical identification pattern. These Attachment-points make it possible to assemble larger structures, by using Fasteners.

The Fastener technology is critical, and needs devices that are easily applied and removed, but also lock with a specified strength. We might use reversible welding techniques. The design of the identification marking is critical, too. The policy of adopting uniform surface identification patterns would make it feasible to maintain an international register in which every object launched into space could have a unique and permanent ID that would let us use pattern-recognition software to keeping track of every Attachment-point. Whenever a new structure is needed, a computer-based design system could automatically investigate the availability of materials. This includes other, already assembled systems. These spatial ID markings could also be used to locate passive monitoring devices, such as thermal and stress indications that indicate conditions by changing how they appear to optical scanners. Since such indicators would require no other hardware provisions, they can be supplied in massive quantities at very little cost.

AUTOMATING ROBOTIC ASSEMBLY. What if we wanted a telerobot to grasp a certain object? We have already discussed having humans do such things "by hand" - that is, by using remote control. But we might also want our telerobot to be able to work automatically, without human assistance. In general, the art of robotics is not yet advanced enough to provide us with reliable automatic manipulation programs; in particular, our technology for Machine Vision is still too weak. However, because we could very easily develop robust vision software for dealing with the precise and unambiguous visual ID markings of our micromodular components, there is no need to wait for the maturity of more advanced computer-vision systems. Very simple vision software could reliably locate Attachment-points. Then, using suitable data-bases for the spatial form of every object, it should be easy to develop effective manipulation programs. Such systems could be made extremely foolproof by testing the match, at every step, between the appearance of the actual scene with that predicted by the Simulator. When any discrepancy appears, the system could call for human assistance.

MOBILITY. How do the telerobots move? This would be very difficult in a conventional spaceship, where each change in location would pose a novel mooring problem. However, every surface of the micromodular spaceship is studded with Attachment-points, and the telerobots can exploit these for mobility! At every step, the Simulator would propose attachment-points for the next step, so that the telerobot can move by grasping one attachment-point after another - as soon as each point's actual existence is confirmed, whether by vision or by feel. Simulation could also confirm each attachment-point's suitability for the expected stress or load.

GRAVITY. Our traditional approach has not exploited the virtues of space, but has usually regarded vacuum and lack of gravity as antagonists to be battled at all cost. This has led to wasting of orbiting mass - the most expensive resource of all. The remotely-manned micromodular scheme offers substantial savings in mass by reducing life support requirements, by making materials reusable, and by exploiting the use of ballistic transport methods which eliminate piping, wiring, and other materials.

TELEROBOTS SHOULD BE MODULAR, TOO

How can we supply power and signal to the motors and sensors of our telerobots while they move around so much? These problems have been difficult in designing terrestrial robot arms. However, these problems should actually be simpler in space, where the power requirements are surprisingly small; a typical motor needs less than one watt. Most such problems could be solved by routing a simple bus throughout the entire telerobot tree, superimposing power and control by treating each sensor and motor as a network node.

A more radical method would treat each section of the telerobot as a virtually independent, self-contained module with its own communication system and power source; this would also simplify their assembly and repair. In space, a 1 watt motor with 25% duty cycle could run for 16 hours from a single size D cell - rechargeble by a solar cell. Such arms would be very dependable, if each motor includes a fail-safe mechanism for braking any joint whose power fails. If the multi-jointed "binary-tree arm" in the first illustration were so designed, any failure could be repaired simply by replacing a faulty module. Note that our binary-tree arm has so many degrees of freedom that it always would be usable in spite of several frozen joints. Why do we not see such robots on Earth? Simply because they would not be feasible: to operate in "real time", earthbound manipulators need such massive motors - and hence such massive structural components, that they are always designed with the fewest joints that will possibly suffice.

NASA has frequently considered developing free-flying teleoperators but, again, the jet powered systems that have been proposed have never seemed suitably practical - mainly because they consumed too much nonrecoverable reaction-mass. However, the availability of good telerobots would permit another approach: a telerobot could propel itself outward by throwing and catching reaction-mass objects under computer control. No matter that this idea seems to horrify every engineer exposed to it; eventually this should be feasible since, if an object is projected slowly enough, it should be possible to verify its trajectory before it goes beyond the reach of the throwing arm. Return propulsion would require tethers. This scheme would also relieve the power problem, by allowing telerobots to exchange low batteries for others projected to them on slow trajectories. For larger scale operations we could center the space station itself inside a wire tetrahedral skeleton supported by the momentum of exchanging mass between its vertices.

TELEPRESENCE AND TIME DELAY. Why have telerobots not have been used more in space? I think it is because of a widespread belief that people will not be able to cope with the time delays. But I am convinced that they will. *We forget that human workers, too, are telerobots in a sense.* We also always have to cope with significant internal time-delays - typically, of the order of 0.2 seconds. You simply cannot catch a ball by tracking it till it reaches your hand; your brain must anticipate its trajectory for the penultimate fifth of a second. In a worst case delay situation between an orbiting telerobot and a human on the ground, the signals will be relayed through two geosynchronous satellite links, with a total loop delay of nearly one second. I am confident that with suitable training, workers should be able to learn to deal with phenomena that proceed at "one-fifth real-time" speeds. Eventually, many of those effects can be reduced by using computers to produce "anticipatory feedback" and, even with less powerful automation aids, each set of remotely manned hands will surely be able to do more than any human EVA worker. It is hard to imagine an EVA astronaut working effectively for more than 6 hours a day, or producing results at more than 1/2 real-time speed during that period. I would not be surprised, then, to find that each telerobot could do a human equivalent of work, even when operating at less than 1/5 real-time speed - and at perhaps one-hundredth the cost with an infinite gain in safety. And this means much more than mere savings lives: it means letting us reach for the wonders of space.

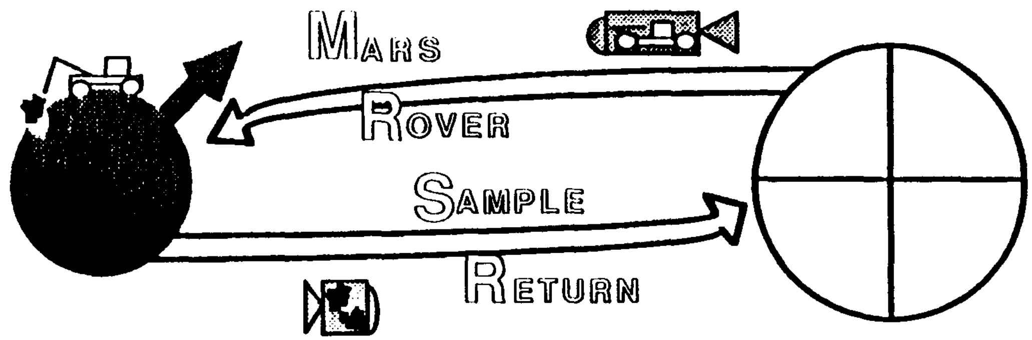
THE US SPACE PROGRAM IS A HOSTAGE TO SAFETY

Our manned space expeditions were wonderful accomplishments, but were risky, costly and limited. And though it is often claimed that the presence of humans on board made emergency repairs possible, I do not find the record impressive. The Apollo 13 crew was unable even to examine the damage. The repair of Skylab's parasol showed that astronauts could barely do what mechanically was an easy task, and they were able only to restore a portion of lost function. Shuttle crews have managed a few satellite repairs but, again, only in very simple situations. Our unmanned missions worked remarkably well. But this was achieved by conservative plan, with virtually every operation planned out years in advance. Since onboard repairs were impossible, these missions depended on the continued operation of critical systems throughout the lifetimes of those expeditions. Dependability became the name of the game, and conservative design, - reliability, rather than resourcefulness and versatility - became the centerpiece of NASA's attitude. Today we are finally paying the price for that habit of planning for years in advance. It served us well when cost was no object and competition seemed inconceivable. But now our obsession with inflexible reliability is confining us to decade-old technologies, paralyzing our launching pads, and institutionalizing a sluggishness that virtually bars us from conquering Space.

There is another aspect to the problem. The past few years have seen an unprecedented expansion of our public concern for human safety. The Challenger episode shows that we are now willing to delay the entire space program, if necessary, for several years - to reduce the chance of an accident. This kind of caution was not characteristic of the explorers of the past, who were careful, indeed, but not to such a degree. This is why the term "hostage". applies: this is no mere concern of NASA alone, but part of a broader phenomenon in which the public has come to demand extraordinary constraints on the reliability of industrial and consumer products. Even in medicine - the technology of maintaining life itself - we are becoming paralyzed by the costs of liability. And on the national policy scale, we have all witnessed how concerns for safety of a few individuals has dominated the perceptions and activities of two powerful American Presidents, one from each of our major political divisions.

For NASA itself, this perception poses a dreadful dilemma: the public is seen as supporting nothing less adventurous than manned exploration - but will not forgive any accident. In this new cultural context, there is simply no way for NASA to obtain the "liability insurance" it needs. And so, like many physicians in recent years, NASA has essentially had to retire from practice, albeit without admitting it. The only solution that I can see (and would have preferred, in any case) is to rebuild the whole program around the third alternative: neither manned or unmanned, but remotely manned. By expoiting telerobotics, we can return to taking the kinds of risks that have always attended expeditions into the unknown. Any malfunctioning systemon a remotely manned station can usually be repaired or replaced - and most of its parts can be used again without new shipments from the ground.

No one can be injured if no one is there. To be sure, many people have supposed that the public would never support unmanned space operations because it simply isn't exciting enough; because there would be so little risk, it could not produce the heros that people need to identify with. But in fact there will be heros indeed, albeit of a different kind. Many individuals would soon be engaged with the new art and technology of teleoperation - and among them, popular "stars" will emerge, as happens in all other areas. Furthermore, thousands of people will soon become personally involved in active exploration roles. Great numbers of other individuals can be invited to share brief in-space experiences - and the development of powerful telepresence simulation facilities will produce new forms of vastly entertaining personal involvement for the public at large. Finally, it should be emphasized that this is not at all, in any case, a step toward abandoning manned exploration - nor even, probably, to postpone it. Instead, it provides a path toward far more ambitious and adventurous programs for expansion into space.



Phobos Expedition Mars Rover

Presentation to Michael H. Sims
NASA/Ames Research Center

by

Donna Pivirotto, Roger Bourke, Bill Dias, Andy Mishkin

July 29, 1988

PHOBOS EXPEDITION MARS ROVER STUDY REPORT

8/4/88

CONTENTS

OVERVIEW/ISSUES

The Task Statement points out that the Phobos Expedition mission calls for the operation of two rovers on the Martian surface by four astronauts in Mars orbit. The Statement requires that the operational feasibility of this mission be examined to see if the mission can accomplish the MRSR science objectives. These objectives, as defined by the MRSR Science Working Group, are to return 100 samples massing 5 kilograms, gathered from a wide variety of geologies, by a rover traveling between 20 and 40 kilometers in 150 to 240 days on the surface of Mars. An MRSR concept has been defined which uses a rover with a "semi-autonomous" level of automation to accomplish these science objectives.

Discussions with Michael Sims of ARC, the Task sponsor, indicated that the Phobos Expidition rovers were seen by some as a hedge against possible technical problems of Earth-based operation of semi-autonomous rovers. The Task Statement reveals a concern as to whether the planned Phobos Expedition can achieve this level of performance by teleoperating the rovers from Mars orbit, due to the operational complexity.

Because of the short duration of this study, we took the approach of comparing the performance of the Phobos Expedition teleoperated rover to the MRSR semi-autonomous rover. Roger Bourke attended a Code Z review and acquired information on the design of the Phobos expedition. He then developed a timeline for the mission, which included a rover operations strategy. Andy Mishkin, the MRSR rover local navigation and sample aquisition cognizant engineer, defined the technology which would be appropriate to carry out this strategy. Bill Dias, who is developing rover scenarios and timelines for MRSR, developed a timeline for the teleoperated rover. The team then evaluated the performance of the teleoperated rover vs. the performance of the semi-autonomous MRSR rover, and drew conclusions.

OVERVIEW/ISSUES

- MAJOR QUESTION: CAN HUMANS ON PHOBOS EXPEDITION PERFORM A SUCCESSFUL MARS ROVER SAMPLE RETURN MISSION?

- MAJOR ISSUE: ROVER OPERATIONS FEASIBILITY

- APPARENT MAJOR ASSUMPTION: ROVER AUTOMATION REQUIRED TO PERFORM MRSR IS INFEASIBLE OR VERY COSTLY, THEREFORE, TELEOPERATION FROM ORBIT IS RELATIVELY ATTRACTIVE

- STUDY PLAN:

    - DEFINE MISSION OPERATIONS TIMELINE

    - DEVELOP ROVER OPERATIONS STRATEGY

    - DEVELOP ROVER OPERATIONS TIMELINE

    - EVALUATE TECHNOLOGY REQUIREMENTS/OPTIONS

ASSUMPTIONS

1) We assumed that the Phobos mission was as defined in the July 20, 1988 Code Z program review attended by Roger Bourke.

2) We assumed that continuous communication capability would exist between the rover and its operators, whether on earth or in Mars orbit. This will require an aerosynchronous relay satellite or a set of several relay satellites at lower altitudes.

3) At the Code Z review we were told to assume that the crew of two remaining in orbit, while the other two went to Phobos, would be available 4 hours per day each to operate the rovers. We also assumed that they would be uninterrupted during these operations periods. Therefore, this scenario is non-conservative to the extent that it ignores emergencies or crew illness.

4) We assumed that the crew would have automated work stations, including expert system geology advisors for sample selection, which would enable them to effectively operate the rovers.

5) We also assumed that the level of risk on landing would be the same as that acceptable to MRSR. That is, either the landing site would be mapped in advance to a real resolution of one meter so that an area free of > 1 meter obstacles would be selected for landing, or, the lander would include active, autonomous hazard avoidance. We did not develop a scenario for detailed mapping by the cargo carrier, therefore, we assumed that the lander would use active hazard avoidance.

ASSUMPTIONS

- PHOBOS EXPEDITION DEFINED AS IN JULY 20, 1988 CODE Z PROGRAM REVIEW

- CONTINUOUS COMMUNICATIONS CAPABILITY BETWEEN ROVER AND OPERATORS ON EARTH OR IN PILOTED SEGMENT

- CREW OF 2 AVAILABLE 4 HOURS PER DAY EACH, WITH NO INTERRUPTIONS

- CREW HAS AUTOMATED WORK STATIONS FOR ROVER TELEOPERATION

- LEVEL OF LANDING RISK = MRSR

The Mars surface operations have been divided into two distinct phases: Search and Mark, followed by Sample Collection. The former takes place from the point that the cargo vehicle arrives until the humans arrive. The latter is conducted while the humans are in orbit about Mars.

Phobos Mars Rover Timeline

- Search and Mark Phase
- Sample Collection Phase
- Features

Search and Mark Phase

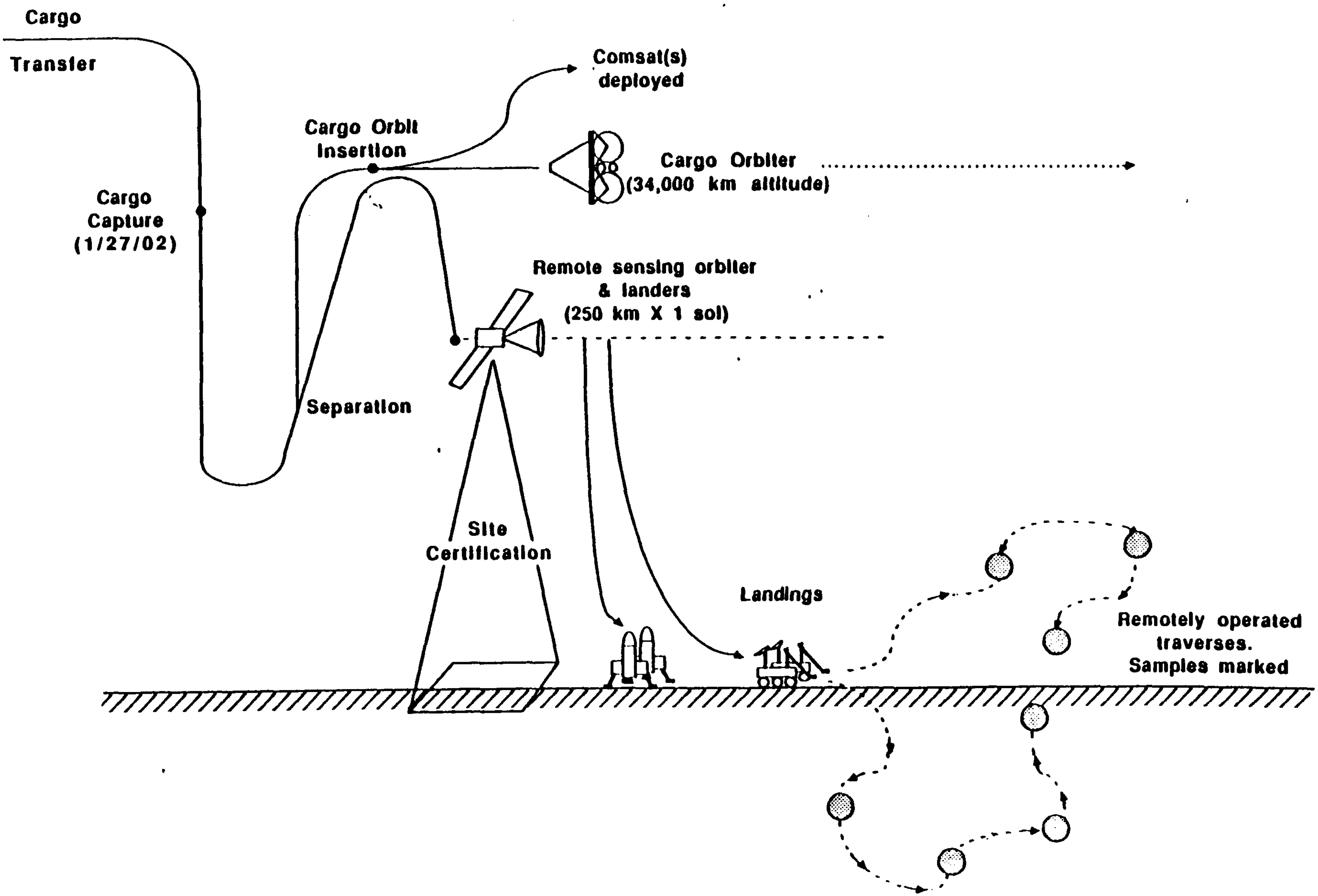
- First 12 hours after cargo vehicle arrival at Mars (1/27/02)
  - Deploy remote sensing orbiter/lander into 250km X 1 sol orbit
  - Deploy com-sat(s)
- First 60 days after Mars orbit insertion
  - Landing site certification
    - Operated from Earth
    - Independent of Phobos cargo vehicle except possibly for com relay
  - Lander separation
    - Autonomous landing similar to current MRSR plan
- Mars orbit insertion + 60 days to human arrival (5/28/03) [14 months]
  - Rover deployed
  - Traverses planned and controlled from earth
  - Samples identified and marked for later retrieval
    - Operations similar to MRSR
  - Return-to-sample-sites plan developed at earth and uplinked to Phobos crew enroute

The Phobos Expedition scenario calls for transfer to a high Mars orbit by initially performing a burn at 250 km periapsis into an ellipse with apopasis at 34,000 km then circularizing at that altitude. Because the remote-sensing orbiter with attached lander needs to be near the planet, it would separate from the cargo vehicle after the initial burn and before circularization. This leaves the orbiter/lander in a 250 km periapsis, 1 sol period orbit.

For the next sixty days, sites on the Mars surface would be surveyed from orbit and certified for landing and roving. Separation of the lander would occur at the end of that period. The lander would perform an automated landing to a pre-designated site avoiding hazards in the terminal phase as necessary; this is identical to the current plan for MRSR.

After reaching the ground, the rover is deployed and begins surface exploration under control from Earth. Samples for later collection are identified and marked over a period of 14 months. Near the end of this period a traverse is planned to return to the sample sites and uplinked to the astronauts enroute to Phobos.

Search and Mark Phase



Sample Collection Phase

- Rendezvous with cargo vehicle
- 15 days between human arrival (5/28/03) and human departure (6/27/03)
  - Rover operated by local crew
    - Returns to sites of marked samples
    - Acquires samples
    - Returns to ascent vehicle
    - Transfers samples
- Human departure - 2 to 5 days (6/22-6/25/03)
  - Ascent vehicle launched from Mars surface
  - Rendezvous with Piloted ship
  - Sample transfered
- Post human departure
  - Rover extended mission

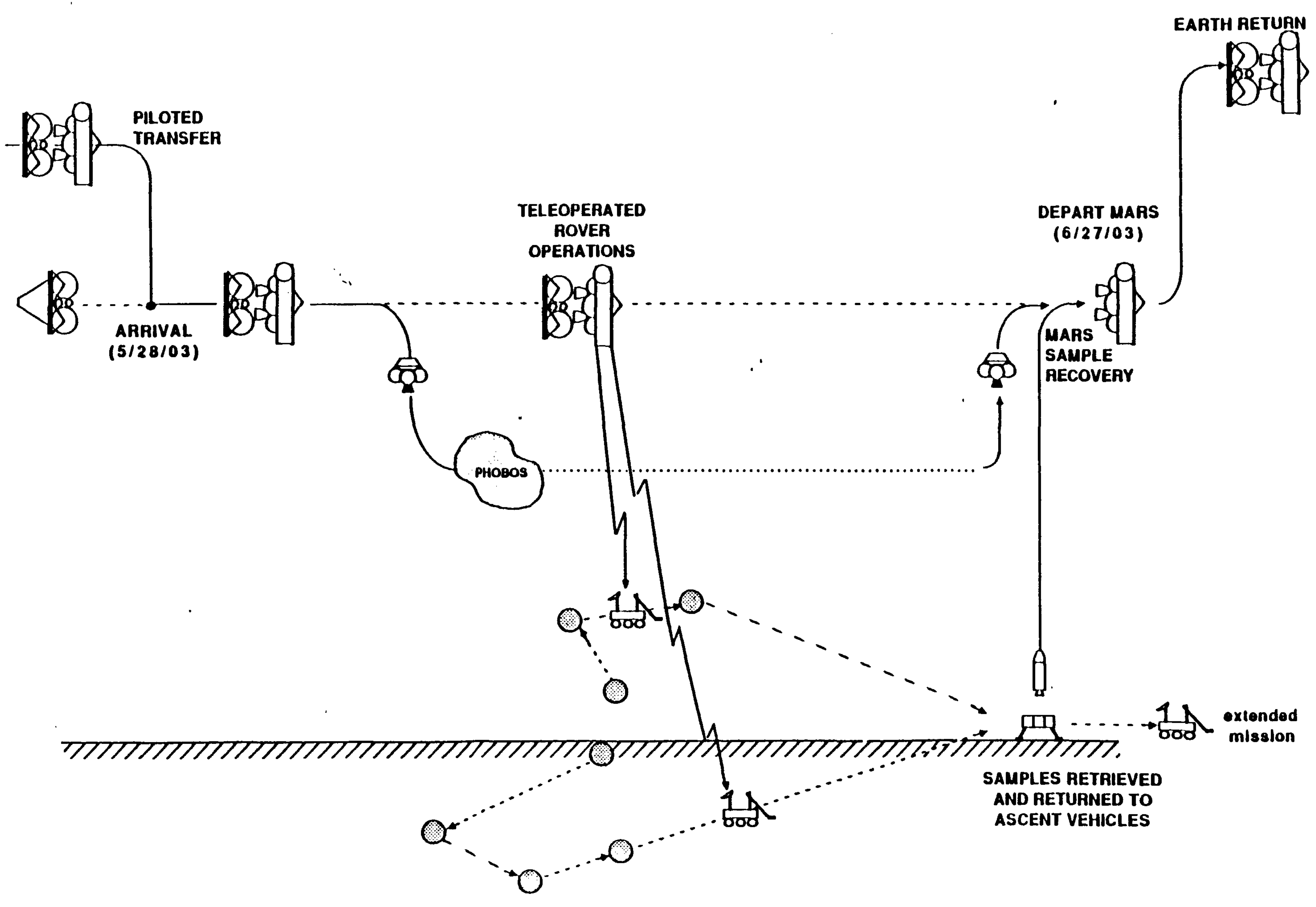
Upon reaching Mars, the piloted vehicle transfers to the orbit of the cargo vehicle, rendezvous and docks. In the next few days the Phobos excursion module leaves the main spacecraft with a crew of two for 20 days. During 15 of those days, the two remaining astronauts are available to operate the Mars rovers.

The rovers are driven back to the marked sample sites and the samples are acquired and packaged. After collecting all the samples, the *rover returns* to the ascent vehicle and transfers the sample cannister thereto.

The ascent vehicle is launched 2-5 days *prior to crew* departure' from Mars, rendezvous with the piloted vehicle and the sample cannister is transfered.

From then on, the rover is available to conduct an extended mission under control from earth similar to the Search and Mark phase.

Sample Collection Phase



Features

- Approximately the same time on Mars surface as MRSR with split opportunity
- High feedback (sampling & transfer) tasks done by nearby crew
- Longer duration tasks (sample selection) done by earth team over months
- Site certification and landing technology and performance requirements same as MRSR
  - Requires automated hazard avoidance

Rover Traverse Approach: Search and Mark Phase

Technique: Computer-Aided Remote Driving (CARD)

Computer-Aided Remote Driving (CARD) is a means of controlling vehicle traversals under conditions of limited bandwidth communications and/or significant communications time delays. CARD is therefore appropriate to control of a Mars rover from Earth, a situation in which both constraints exist (average roundtrip communications time to Mars is 25 minutes). This approach requires only rudimentary autonomy on the remote vehicle.

<u>Sequence of operations for Earth-Based CARD:</u>

1) Stereo cameras on the rover's pan/tilt head provide 90° x 45° panoramas at mradian resolution. Stereo cameras are approximately 0.5 meters apert and 2 meters above the ground. A strobe light provides illumination out to about 30 meters in front of the rover for night imaging.
2) Panoramas (~2Mbytes) are downlinked to Earth for path designation.
3) Human operators on Earth view panoramas in stereo display. Five to thirty meter long paths are designated by a human operator using a joystick and 3D cursor in the stereo display.
4) Meanwhile, the terrain contours are modelled using stereo correlation of the stereo images. Execution of the selected path is simulated to derive a model of sensor readings expected during traversal. The model defines the envelope of acceptable sensor readings appropriate at each point in the traverse (e.g., certain tilt and accelerations would be acceptable during an anticipated crossing of a 1-meter high rock, but the same readings signal a problem if the rover were expected to be rolling over flat level ground).
5) The path definition and sensor expectation model are uplinked.
6) Rover executes path while monitoring execution. Minor deviations from expected sensor readings are ignored or used to make dead reckoning corrections; major deviations result in abort or the traverse or to limited backtracking by the rover.
7) Surface property assessment is performed by human interpretation of stereo images and use of mechanical probes in questionable terrains.
8) Process repeats at end of each traverse.

Possible sensors: inertial reference unit, contact, attitude, acceleration, stereo CCD cameras, over-the-ground distance, odometer, articulation, mechanical probe, traction & slip.

ROVER TRAVERSE APPROACH: SEARCH AND MARK PHASE

- Earth-based Computer-Aided Remote Driving (CARD).
- Rover mounted stereo cameras capture image pairs. Images downlinked to Earth.
- Human operator designates 5-30 meter safe path using stereo display.
- Path traversal is computer-simulated; sensor expectation model generated.
- Path and expectation model are uplinked.
- Rover executes path while monitoring sensors. Rover stops or backs up if actual sensor values are sufficiently anomalous.
- New image pair captured at end of path traversal.

Rover Traverse Approach: Sample Collection Phase

Technique: Direct Teleoperation

In direct teleoperation, the human operator is effectively driving the remote vehicle as if he were driving his own car. Television cameras on the vehicle provide continuous video of the terrain ahead of the rover as traverses the surface. Steering and speed commands from the operator are transmitted directly to the vehicle and implemented. The rover moves continuously for long periods of time (unlike CARD), until the human operator chooses to halt the vehicle.

Features:

- **Mars orbit use.** Direct teleoperation of a Mars rover is only feasible from Mars orbit (or the Martian surface), not from Earth. The human operator must be able to immediately observe and respond as he drives the rover over varying terrain. This would be impossible if the operator were on Earth, given the up to 45 minute roundtrip communications delay.

- **Minimal time delay eliminates CARD requirement.** From the Mars orbit selected for the Phobos mission, the roundtrip light time delay would be only 0.2 seconds. A human in orbit would be able to directly teleoperate the slow moving (< 60 cm/sec) rover, so long as video at several frames per second was available from the rover.

- **Live video required.** In order to receive the appropriate motion cues for driving, as well as have time enough to react to potentially catastrophic situations as they arise, the human operator must be provided with video from the rover at a rate of at least several frames per second while the rover is in motion.

- **Stereo images desired.** While the motion cues provided by live video are generally sufficient to support direct teleoperation, stereo images presented through an appropriate display would provide the operator with direct depth information, thereby improving performance.

- **Operator performs monitoring function.** In CARD, the path to be traversed is simulated, a sensor expectation model is generated and uplinked, and the rover autonomously monitors its traversal. In direct teleoperation, the human operator is continously monitoring and controlling the vehicle traversal, through video and other sensor readings, taking over these functions.

ROVER TRAVERSE APPROACH: SAMPLE COLLECTION PHASE

- Direct teleoperation ("joysticking")
- From Mars orbit only
- Short (0.2 sec) time delay eliminates requirement for CARD
- "Live" video required (several frames per second)
- Stereo imaging desired but not required
- Human operator performs execution monitoring function

Assumption: Loss of Video During Traverse

Alternate Rover Traverse Approach: Sample Collection Phase

Technique: CARD

In direct teleoperation, the human operator continuously specifies rover direction and speed; in addition, he takes over the execution monitoring and reflex response functions that must be automated in CARD. In order to perform these functions, the operator must receive video from the rover at several frames per second, while the rover is in motion. The rover must drive across sometimes rough terrain, while maintaining appropriate antenna pointing in spite of terrain-dependent changes in vehicle attitude. Under these conditions, continuous video transmission from the rover to Mars orbit may be difficult or impossible.

If the video link between the rover and its operator in Mars orbit is broken for any significant fraction of the time that the rover is in motion, then the operator cannot perform the required functions of vehicle control and execution monitoring. The vehicle would effectively be driven blind during these periods, or would be forced to stop. Average vehicle speed would be reduced and rover safety compromised.

For these reasons, Mars-orbit-based Computer-Aided Remote Driving (CARD) becomes the desired alternate traverse approach when a continuous video link cannot be maintained. The human operator receives a set of stereo images from the stationary rover, then plans an approximately 10 meter long path in the stereo display; the path traversal is simulated; the designated path and sensor expectation model are transmitted to the rover, which executes the path while monitoring its progress. At the end of the path, the rover stops, captures a new pair of images, re-establishes contact with the orbiting control station, and transmits the new images.

The use of CARD significantly reduces the bandwidth required for the transmission link from the Martian surface to orbit, compared with direct teleoperation. The number of images transmitted is reduced 2-3 orders of magnitude. If desired, several seconds can be allotted to transmitting a single stereo image pair.

However, CARD with a human operator in Mars orbit requires that traversal simulation to generate the sensor expectation model also be performed in Mars orbit. The time required to perform this simulation will significantly affect average rover speed in the CARD scenario. Sensor expectation modeling and execution monitoring are current research topics; the fidelity (and hence, the computing) required to support safe vehicle traverse is presently uncertain. For Earth-based CARD, effectively infinite computing resources can be applied to allow the simulation to be performed in near real-time, while the communications time delay is so large (between 6 and 45 minutes) that additions of a few minutes to computing time would not significantly affect average rover speed. For Mars orbit-based CARD, the computing resources available will certainly be mass and power limited, while the communications time delay will be minimal (< 1 second); a simulation time of 1-2 minutes could cut the average rover speed in half.

ASSUMPTION: LOSS OF VIDEO DURING TRAVERSE

ALTERNATE ROVER TRAVERSE APPROACH: SAMPLE COLLECTION PHASE

- If human operator cannot monitor traverse, on-board execution monitoring is required.
- CARD from Mars orbit becomes traverse approach of choice.
- Time required for generation of expectation model in Mars orbit is uncertain (cannot assume virtually infinite computing resources as on Earth). Expectation planning may become time driver.
- When rover is stationary at end of path traversal: Capture new image pair, re-establish communications link with control station in Mars orbit, transmit images.

Control of Sample Acquisition: Search and Mark Phase

Technique: Earth-Based Supervisory Control

The long communications time delay from Earth to Mars precludes the use of direct teleoperation to control sample collection and manipulation. Instead, supervisory control is used: high level commands are transmitted to the rover, interpreted and carried out by the sampling subsytem located at the work site. High level commands will specify such parameters as end effector choice, desired manipulator arm positions, forces to be applied by the end effector.

Since the manipulator arm and any other sample gathering eqipment must ineract with the uncontrolled, crudely modelled Martian surface, multi-sensor-based manipulator control must be used. This type of control includes sensor data as parameters in commands, so that the manipulator can respond to its environment as perceived using its force/torque and slip sensors. In this way, the manipulator can halt its forward motion once it makes contact with a rock, and can avoid building up large forces during sampling operations. Without sensor-based control, unexpected force buildup could damage the sampling arm or a potential sample.

Functional Implementation:

1) Stereo panoramas and multispectral scanning are used to identify samples. Earth-based operators designate desired sample(s), and relocate rover to enable sampling (via CARD).
2) Pre-sampling sensor scan is performed, using cameras on manipulators and possibly other sensors. Sensor data is downlinked.
3) Using the downlinked data, the following operations are performed on Earth: manual or computer-aided 3-D modeling of sample objects, manual or computer-aided selection of end effectors and grasp contact points on object, task planning and manipulator trajectory simulation. Supervisory commands and a sensor expectation model are uplinked.
4) On-board expansion of supervisory commands to executable commands; sensor-based manipulation.
5) Commands uplinked during single command cycle may encompass acquisition of multiple samples within reach; if a sample acquisition attempt fails, the system can attempt to retrieve the next designated sample without an additional communications cycle.
6) Retrieved samples are dumped into a sample preparation subsystem, which uses hard automation to slice, crush, or otherwise prepare samples for analysis or storage.

CONTROL OF SAMPLE ACQUISITION: SEARCH AND MARK PHASE

- Earth-based simple supervisory control; multi-sensor-based manipulator control.
- Human operators designate desired samples, relocate rover using CARD.
- Pre-sampling sensor scan performed; data downlinked.
- Earth-based operations on downlinked data: computer-aided 3D modeling of objects, selection of end-effectors and grasp points, task planning, manipulator trajectory simulation.
- Supervisory commands and sensor expectation model uplinked. Uplink may include acquisition of multiple samples within reach.
- Rover-based expansion of supervisory commands to executable commands; sensor-based manipulation.
- Significant variation in sample acquisition times due to uncertainty in environment, frequency of acquisition failures requiring Earth-based replanning.

Control of Sample Acquisition: Sample Collection Phase

Technique: Augmented Master-Slave Teleoperation

In master-slave teleoperation, the human operator uses a "master" arm (perhaps his own, encased in an instrumented harness) to direct the action of the remote "slave" arm at the worksite. Motions of the master are translated into analogous motions by the slave arm. Forces experienced by the slave may be displayed to the operator through the master, informing the operator when the slave has made contact with an object in its environment, and providing a sense of how much force is being exerted by the slave arm. Continuous video and force feedback provide the principal sensory information to the operator to aid him in performing the task. Depending on the dexterity required for the task, the fidelity of the master-slave manipulation system, and the time delay in the master-slave link, a task may take between 3 and 100 times as long to perform as it would if the human were present "in person" performing the task. In the sample collection phase, the tasks include picking up rocks, scooping up pebbles and soil, drilling shallow cores in rock, and making deep cores (1-2 meters) in soil.

Features:

- **Mars orbit use.** Direct teleoperation of a sampling manipulator arm is only feasible from Mars orbit (or the Martian surface), not from Earth. The human operator must be able to move the arm carefully and respond quickly to manage the forces applied when the arm comes into contact with objects in the environment.
- **Time delay requires some on-site control.** The time delay of 0.2 seconds is much more significant in the case of manipulating objects than in the case of rover traversal. Compared to a no delay system, operator performance can be significantly reduced by delays of .05-.3 seconds. The manipulator system can become unstable if human responses are out of phase with manipulator motions. Control algorithms on-board the rover can help manage forces, and can place a limit on the maximum force exerted by the arm.
- **Teleoperation sufficient for unconstrained motions.** For motions of the slave arm that are not intended to (and are unlikely to) make contact with any objects in the environment, pure teleoperation can be used, despite the time delay. The human operator must avoid rapid motions to avoid collisions.
- **Shared control.** Local (rover-based) control will be particularly necessary for delicate operations in which the forces will vary rapidly, such as during drilling or coring. In some cases, rover-based control algorithms will control one or more axes of manipulator motion, while the human operator controls the remaining axes.
- **Variation in sample acquisition time.** In most cases in which teleoperation is used, the environment is well understood, and contains man-made objects that have well known physical properties and are thoroughly modelled. The Martian environment in the immediate vicinity of the manipulator will be only crudely modelled, at best, before samples are collected. It will not always be clear whether a desired sample is a free-standing rock or an outcropping until the operator attempts to pick it up. Objects may slip from the manipulator's grip, or crumble to nothing. Significant variation in the time required to acquire samples must be assumed.

CONTROL OF SAMPLE ACQUISITION: SAMPLE COLLECTION PHASE

- Augmented master-slave teleoperation
- From Mars orbit only
- Time delay of 0.2 seconds requires some degree of on-site control
- Teleoperation sufficient for unconstrained motions
- Shared control needed for object manipulation, drilling
- Significant variation in sample acquisition times due to uncertainty in environment (rock masses, friction coefficients, structural integrity, etc.)

TIME DELAY ISSU

Time delay between when the operator senses what is happening to the rover, and the time when a command to respond to that sensory data reaches the rover, is a key operations issue. The round trip light time for earth commanding averages 20 minutes. This requires the CARD technology, and strongly indicates that on-board execution monitoring and emergency reflex response be included in the rover for the search and mark phase.

During the sample collection phase the scenario assumes that the crew has continuous video from the rover and continuous command capability to the rover, so that they can detect and avoid hazardous situations. The time delay for this phase needs to be on the order of a second or so, short enough to allow emergency commands to take effect before the rover falls over a cliff, for example.

The crew is most valuable for controlling the manipulation tasks involved in sample collection. Sample collection using master-slave teleoperation will have two modes: Free space movement of the manipulator, and when the manipulator is in contact with an object. During free space movement the time delay can be up to several seconds as long as the manipulator is on a trajectory where there is no danger of it contacting anything.

When the manipulator is touching a sample force feedback will be required for the operator to effectively control sample manipulation. As long the contact takes place slowly an operator with force feedback can perform manipulation well, provided the time delay is short enough. Human factors studies have shown that there is a trade between time delay and performance in relatively slow force feedback operations. If the time delay is less than 0.05 seconds the performance will be crisp in terms of the operator's "feel" and in terms of the system's response to operator control. Between 0.05 and 0.3 seconds the performance will be good to fair. Above 0.3 seconds a human is so misled by the delay between when the "feel" occurs and when the control signal is received by the manipulator, that control is no longer possible. For operations with high frequency motions, such as drilling, local automated control is required. "Shared" control, currently under development by OAST's telerobotics project, allows the operator to position a tool, for example, and control force in one direction (say keeping forward pressure on a drill), while an automated control loop controls forces in pitch and yaw.

TIME DELAY ISSUES FOR TELEOPERATION

- TRAVERSE CONTROL
  - REQUIRES CONTINUOUS VIDEO WHILE MOVING TO ALLOW CREW TO PROVIDE EXECUTION MONITORING AND EMERGENCY ACTIONS
  - TIME DELAY HAS TO BE SHORT ENOUGH TO ALLOW EMERGENCY ACTIONS

- SAMPLE COLLECTION - FREE SPACE MOVEMENT OF MASTER/SLAVE CONTROLLED MANIPULATOR
  - TIME DELAY CAN BE SECONDS AS LONG AS MANIPULATOR IS NOT IN DANGER OF CONTACTING ANYTHING

- SAMPLE COLLECTION - MANIPULATOR IN CONTACT
  - FORCE FEEDBACK REQUIRED FOR CONTACT OPERATIONS
  - LOCAL AUTOMATION REQUIRED FOR CONTROL OF HIGH FREQUENCY OPERATIONS SUCH AS DRILLING
  - TRADE BETWEEN TIME DELAY AND PERFORMANCE IN FORCE FEEDBACK OPERATIONS (ROUND TRIP LIGHT TIME) =
    - ◊ <.05 SEC = CRISP PERFORMANCE
    - ◊ .05-.3 SEC = GOOD TO FAIR PERFORMANCE
    - ◊ > .3 SEC = HUMAN CANNOT CONTROL

- "SHARED CONTROL" = TECHNIQUE FOR OVERCOMING TIME DELAY PROBLEMS IN < 1 SECOND RANGE

Description of Possible Phobos Expedition Mars Rover Activity Pattern

7/29/88

INTRODUCTION

In order to facilitate comparison to MRSR, we kept the sample mission goals equivalent to an 8-month (235-sol) MRSR mission we have already analyzed, in which 25 contingency samples are collected by the rover near the ascent vehicle (AV), 16 kilometers are traversed in a search pattern outbound from the AV towards objectives discovered from orbit, and 24 kilometers are spent getting back towards the AV while getting samples from areas just discovered. The main differences will be that in the Phobos Expedition (PE) version, during the 14 months before the astronauts arrive (Search and Mark Phase) we will just get 100 close-ups on the way back to the AV rather than collect the samples. After the astronauts arrive (Sample Gathering Phase) we will go back to the same tracks again and try to collect the 75 samples.

SEARCH AND MARK PHASE

The numbers on the diagram refer to the Search and Mark phase of the mission. The circle at the center of the diagram shows where the rover and ascent vehicle (AV) land together.

In the "contingency traverse", earth scientists and engineers select samples near the AV for collection by the rover. Even though the referenced 8-month MRSR mission has a semi-autonomous rover, the contingency sample is procured with low autonomy for reasons of safety, using the CARD navigation method, so 50 sols are required for this. The decision to acquire a contingency sample is based mainly on imaging from the AV.

Next the rover traverses a total of 40 km, 16 km outbound from AV (thick lines), scanning the general area for about 60 meters on each side. Earth scientists pick 100 areas for close-up while heading back to AV (thin lines). We count no science decision time because there is a long time in between initial pass-by and return for close-up. It would take 448 sols (460 days, 15.3 months) to "duplicate" the 8-month MRSR mission with a search and mark effort, which is a little more than the 416 sols (14 months) alotted, so result is only marginally equivalent.

SAMPLE GATHERING PHASE

60 hours are available in which to try to collect the 75 samples. It would require 115 hours, or about twice as much time, to complete the job. This assumes maintaining a 20 cm / second teleoperated traverse rate along the same 40 km as in the earlier phase. Full advantage is taken of the fact the territory is familiar, and an expert system informing the operator of how to steer and where to stop is assumed. Terrain must not be assumed to be easy all the way, however, since we must get close to samples.

C-14

PHOBOS EXPEDITION (PE) MARS ROVER
POSSIBLE ROVER ACTIVITY PATTERN

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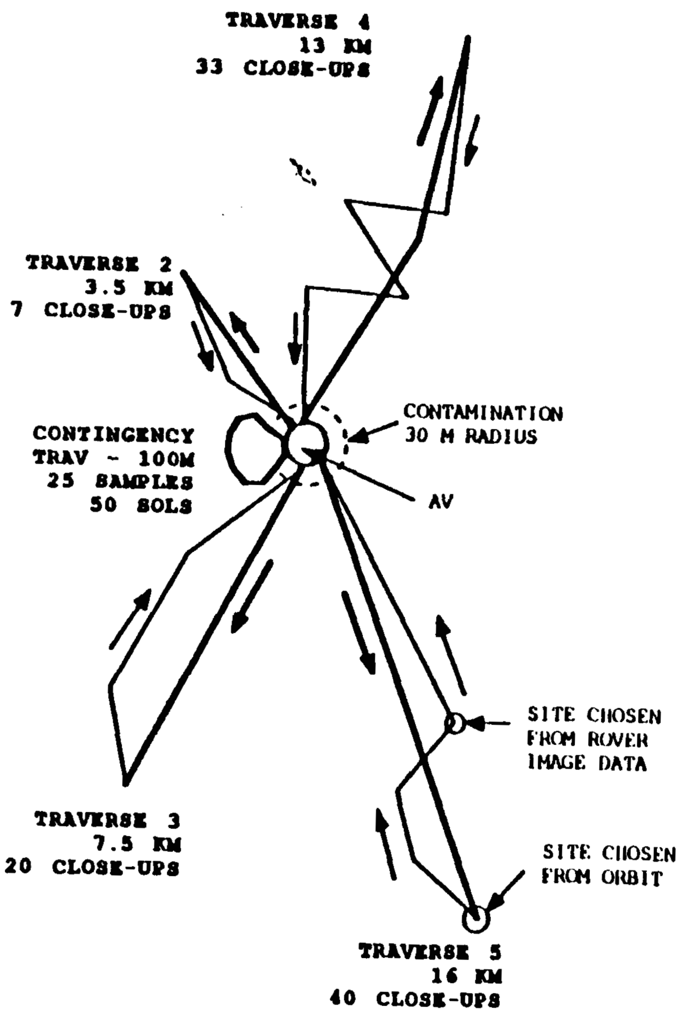

SEARCH & MARK PHASE

14 MONTHS (416 SOLS) AVAIL

DEPLOYMENT, 25 CONTINGENCY SAMPLES 100 CLOSE-UPS PLUS 40 KM ACHIEVED IN 448 SOLS

MRSR 235-SOL RESULTS ACHIEVABLE ONLY MARGINALLY

SAMPLE COLLECTION PHASE

60 HOURS OPS TIME AVAIL

75 SAMPLES + 40 KM REQUIRES 115 HOURS

MRSR 235-SOL RESULTS NOT ACHIEVED BY A FACTOR OF TWO

ORIGINAL PAGE IS OF POOR QUALITY

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PHOBOS EXPEDITION (PE) MARS ROVER SCENARIO ASSUMPTIONS

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STUDY OBJECTIVE

COMPARE SCIENCE OBTAINED FROM A PE MARS ROVER IN 15 MONTHS TO THAT FROM AN 8 MONTH MRSR SCENARIO, WITH EARTH-DIRECTED AUTONOMOUS ROVER

ASSUMED MISSION STRUCTURE

ALLOW FOR MULTIPLE PASSES THROUGH THE SAME AREAS.
FIRST PASS PROVIDES DATA FOR GROSS SCIENCE DECISION MAKING.
SECOND PASS GETS DETAILED INFO PER DIRECTIONS FROM FIRST PASS.
SAVES TIME BY GETTING DECISION MAKING OUT OF SERIAL MODE.

SEARCH AND MARK PHASE - 14 MONTHS (3/27/02 - 5/28/03) 426 DAYS = 416 SOLS

COLLECT AND DELIVER 25 NEARBY CONTINGENCY SAMPLES TO ASCENT VEHICLE (AV) - 50 SOLS

SURVEY WHILE TRAVERSING 16 KM (LIKE MRSR). EARTH SCIENCE SELECT 75 SAMPLES FROM 100 CLOSE-UPS BEFORE ASTRONAUTS ARRIVE. PROGRAM EXPEDITION EXPERT SYSTEM FOR SAMPLE SELECTION.

SAMPLE COLLECTION PHASE - 30 DAYS (5/28/03 - 6/27/03)

FOR 15 OUT OF THE 30 DAYS, ASTRONAUTS SPEND 4 HOURS PER DAY PER ROVER COLLECTING SAMPLES DESIGNATED IN FIRST PHASE

RETURN THESE TO ASCENT VEHICLE (AV) IN SEVERAL TRIPS

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PHOBOS EXPEDITION (PE) MARS ROVER SCENARIO ELEMENTS

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SEARCH AND MARK PHASE

COMPUTER-AIDED REMOTE DRIVING (CARD) NAVIGATION

ROUND-THE-CLOCK EARTH OPS, MARTIAN NIGHT OPS

ONE INTERACTION IN FOUR HOURS

SIX 20m INTERACTIONS PER SOL = 120 METERS PER SOL

16 HOURS PER SAMPLE CLOSE-UP (IE, 4 INTERACTIONS)

25 CONTINGENCY SAMPLES ACQUIRED NEAR AV (35 HOURS EA)

SAMPLE COLLECTION PHASE

SWITCH TO TELEOPERATION, CONTINUOUS VIDEO

20 CM / SEC (0.7 KM / HOUR) AVERAGE SPEED

PEBBLE LOCATE, ACQUIRE + INSPECT = 0.5 HOURS

SOIL SCOOP LOCATE, ACQUIRE + INSPECT = 0.5 HOURS

2 CM ROCK CORE LOC, ACQ + INSPECT = 1.0 HOURS

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PHOBOS EXPEDITION (PE) MARS ROVER SCENARIO CONCLUSIONS

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OVER 14 MONTHS, A ROVER INTERACTIVELY GUIDED FROM EARTH COULD ONLY MARGINALLY SURVEY AN AREA SIMILAR TO THAT OVER WHICH MRSR'S SEMI-AUTONOMOUS ROVER COULD BOTH SURVEY AND COLLECT 75 SAMPLES IN 8 MONTHS

A ROVER INTERACTIVELY DIRECTED FROM MARS ORBIT COULD NOT REACH AND COLLECT 75 SAMPLES DESIGNATED IN THE 14 MONTH SURVEY, ASSUMING 60 HOURS TO WORK

THE ACQUISITION PHASE WOULD ALLOW NO TIME FOR SAMPLE ANALYSIS BEYOND IMAGING OF THE ACQUIRED SAMPLE'S SURFACE

FINAL SELECTION DECISION WOULD BE MADE VERY RAPIDLY BY ONE PERSON, GREATLY DEPENDENT ON AN EXPERT SYSTEM PROGRAMMED WHILE ASTRONAUTS EN ROUTE TO MARS ORBIT

NO TIME OR PROVISION FOR SCIENCE FEEDBACK TO FINAL SELECTION

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PHOBOS EXPEDITION (PE) MARS ROVER
SCENARIO BACKGROUND DETAIL
SEARCH AND MARK PHASE SCENARIO ELEMENTS

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COMMAND CYCLE:	EARTH-BASED ENGINEERING, SEQUENCING	80 MIN
	COMM (W / 20 MIN DELAY)	25
	ROVER MOVE	5
	ROVER DO SCIENCE, NAVIGATION IMAGING	70
	COMM (W / 20 MIN DELAY)	40
	TOTAL ABOUT 4 HOURS / CMD CYCLE	220 MIN

TRAVERSE: 1 COMMAND CYCLE - 4 HOURS - 6 CYCLES / DAY
IMAGE CLOSE-UP: 4 COMMAND CYCLES - 16 HOURS

CONTINGENCY SAMPLE ACQUISITION: BASED ON INTERACTIVE ROVER ANALYSIS FOR EARLIER VERSION OF MRSR 5/2/88	SCIENCE SAMPLE SELECTION	8 HOURS
	ORIENT ROVER	4
	RE-ORIENT ROVER	4
	ACQUIRE	4
	SCIENCE VALIDATE SAMPLE	8
	STORE SAMPLE	4
	VALIDATE STORAGE	3
	TOTAL ACQUISITION	35 HOURS

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PHOBOS EXPEDITION (PE) MARS ROVER SCENARIO BACKGROUND DETAIL

SAMPLE COLLECTION PHASE DETAILED ASSUMPTIONS

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CONTINUOUS VIDEO TELEOPERATIONS, OR REDISPLAY EACH SECOND

APPROX 0.25 SEC TWO-WAY DELAY

PLENTY OF POWER, ENERGY

TRAVERSE SPEED - AVERAGE 20 CM / SEC (0.7 KM / HOUR)

SAMPLING-RELATED

(TIMES INCLUDE 20 SEC FOR OPERATOR VIEW BETWEEN MAJOR STEPS)

ARM MOVEMENT: 50 SEC

ATTATCH / DETATCH TOOL, DEVICE, CAMERA TO ARM: 40 SEC

TOOL INTERACT WITH SAMPLE: 80 SEC

ACQUIRE ROCK CORE (JUST THE DRILL PART): 30 MIN

THREE SCIENCE CLOSE-UPS OF PROPOSED SAMPLE AREA: 100 SEC

THREE SCIENCE CLOSE-UPS OF ACQUIRED SAMPLE: 110 SEC

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PE MARS ROVER SCENARIO BACKGROUND DETAIL

SAMPLE COLLECTION PHASE SAMPLING SCENARIO ELEMENTS

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INVESTIGATE PROPOSED SAMPLE - DECIDE WHETHER TO ACQUIRE W/ EXPERT SYSTEM ASSISTANCE			ACQUIRE + SCIENCE IMAGE CHOSEN SAMPLE		
ARM MOVE	50	SEC	ARM MOVE	50	SEC
ATTATCH CAMERA	40		ATTATCH GRIPPER	40	
ARM MOVE	50		ARM MOVE	50	
CLOSE-UP	100		CONTACT	80	
ARM MOVE	50		GRASP	80	
CLOSE-UP	100		LIFT	80	
ARM MOVE	50		ARM MOVE	50	
CLOSE-UP	100		CLOSE-UP	110	
ARM MOVE	50		ARM MOVE	50	
STOW CAMERA	40		CLOSE-UP	110	
EXPRT SYS	60	(ACQUIRE?)	ARM MOVE	50	
			CLOSE-UP	110	
TOTAL 11 MIN	690	SEC	EXPERT SYS	60	(KEEP?)
			ARM MOVE	50	
			STORE SAMPLE	80	
			ARM MOVE	50	
			STOW GRIPPER	40	
			STOW ARM	50	
			TOTAL 20 MIN	1190	SEC
			ACQUIRE ROCK CORE 50 MIN		

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PHOBOS EXPEDITION (PE) MARS ROVER
SCENARIO BACKGROUND DETAIL
SAMPLE COLLECTION PHASE SCENARIO ELEMENTS

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SAMPLE COLLECTION PHASE IMAGE CLOSE-UP SCENARIOS

PROPOSED SAMPLE:

VIEW SCENE	20 SEC
CONFIGURE CAMERA	10
POSITION CAMERA, IMAGE	10
VIEW, QUERY EXPRT SYS, DECIDE	60
TOTAL	100 SEC

ACQUIRED SAMPLE:

VIEW SCENE	20 SEC
CONFIGURE CAMERA	10
POSITION SAMPLE, IMAGE	20
VIEW, QUERY EXPRT SYS, DECIDE	60
TOTAL	110 SEC

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Risks Associated with Rover Traversals

Hazards of the Martian Surface:

- Geometric Hazards. Rocks, steep slopes, chasms, cliffs, plateaus, etc. are considered hazards if they are not traversable, require unacceptable drains on rover power to become traversable, or may result in damage to the vehicle. The rover may be unable to climb over a large rock, may not have the power to drive up a steep slope, could get stuck in a chasm or on a plateau, and might destroy itself if it drove over the edge of a cliff. Geometric hazards can generally be detected visually, and therefore can usually be avoided.
- Non-geometric Hazards. This type of hazard can be dangerous because it may not be easily detectable at a distance as are geometric hazards. The rover is more likely to find itself in the middle of a non-geometric hazard, and possibly require a time-critical response to protect itself. One class of hazards includes surfaces with load-bearing capacities too low to support the rover. The rover could drive into a pool of dust or onto the leeward side of a sand dune, and sink into the dust or sand, becoming trapped and overheated. Some lava flows form a shelly surface over a void; a vehicle could drive onto such a surface, then break through and become caught. Another class of hazards during from materials with low fictional coefficients. Slopes that appear shallow enough to be crossed may become impossible due to lack of vehicle traction. The rover might easily drive down into a riverbed, and then be unable to get out.

CARD Execution Monitoring Issues:

- Requirement for Execution Monitoring. In view of the hazards that may be encountered while traversing the Martian surface, execution monitoring must be performed to reduce vehicle risk to an acceptable level. Without execution monitoring, the vehicle would blindly attempt to continue its traverse after a problem was encountered. With execution monitoring, the rover would recognize that the front wheels were slipping and sinking, and bring the rover to a halt (or even backtrack some distance); without it, the vehicle might dig itself deeper into dust, until there was no means to correct the situation.

Teleoperation:

- Human factors issues. The human crew will have spent several month in a state of relative inactivity during the transit to Mars. Rover control will require periods of several hours of intense concentration. Depending on the specific implementation of teleoperation (choices of video update rate, stereo imaging, field of view), teleoperation may be extremely stressful. The teams controlling Lunokhod were exhausted at the end of 4 or 5 hour shifts. Since hazards may be widely separated, rover control could consist of long periods of uneventful operations combined with the need to recognize vehicle endangering hazards quickly, before they become full-fledged catastrophes; human performance under such conditions may be questionable.
- Sensitivity of mission success to minor problems. In the Phobos mission scenario, all rover operations take place over a 15 day period. Given the complexity of the mission (both rover operations and a human excursion to Phobos taking place simultaneously), unplanned crew activities seem likely. If the rover operators must support the Phobos crew for an extra day or two, a significant fraction of rover operations time will have been lost. The effect of any minor hardware downtime will also be magnified by the tight schedule required for rover operations.

RISKS ASSOCIATED WITH ROVER TRAVERSALS

- **Hazards of Martian Surface:**

  Geometric Hazards

  Rocks, steep slopes, chasms, cliffs, plateaus, etc.

  Non-Geometric Hazards

  Insufficient load-bearing capacity (e.g., dust, shelly lava, sand dunes)

  Low frictional coefficients (e.g., dust, sand dunes)

- **CARD Issues**

  Earth-based CARD: execution monitoring and reflex response required; otherwise rover would have no chance to react in time-critical situations.

  Expectation generation/execution monitoring implies significant decrease in probability of loss of rover.

- **Teleoperation**

  Human operator performs monitoring function, permitting relatively rapid response to anomalies.

  Human factors related risks: Vigilance, stress concerns; exceptional performance required in short bursts folllowing months of relative inactivity.

  Even minor hardware or scheduling problems may be potentially catastrophic due to limited time (15 days) available for sample acquisition phase.

QUESTIONS/ISSUES

This section explicitly addresses the questions raised in the task statement.

1) Does this mission design significantly ease the design of an MRSR rover?

This mission can be conducted with a combination of computer aided remote driving (CARD) controlled from earth, and "teleoperation" controlled from Mars orbit for sample collection activities. This is a lower level of technology than the semi-autonomous MRSR rover.

However, the Phobos Expedition cannot meet the MRSR requirements. Furthermore, CARD operation is relatively risky unless some form of automated monitoring and control to avoid hazardous situations is included. Adding this capability means that the automation would approach that of MRSR.

In addition, the difficulty and expense of developing semi-autonomous technology is minor compared to the difficulty and expense of developing the technologies to send humans to Mars.

2) What level of rover autonomy is required for the mission to be operationally feasible and scientifically productive?

See the above discussion re operational feasibility. Since the Phobos Expedition cannot achieve the MRSR science return with a lower level of technology, it appears that a semi-autonomous rover would be required whether humans are present or not to make the mission scientifically productive.

3) How reasonable is it to assume the technological availability of this autonomy in the mission time frames?

CARD and teleoperation technologies, including automated crew stations, are readily achievable with small investments. Semi-autonomous technology at the MRSR level appears achievable if the plans developed for the OAST Pathfinder Planetary Rover and Sample Aquisition, Analysis and Preservation line items are carried out.

QUESTIONS/ISSUES

- DOES MISSION EASE DESIGN OF MRSR ROVER?
  - CAN BE DONE WITH CARD AND TELEOPERATION, SEMI AUTONOMY NOT REQUIRED
  - HOWEVER, HIGHER LEVEL OF AUTOMATION REDUCES RISK, ENHANCES MISSION RETURN
  - DEVELOPMENT COST OF AUTOMATION IS MINOR PART OF PHOBOS EXPEDITION COST

- LEVEL OF AUTONOMY REQUIRED?
  - SEE ABOVE
  - SHOULD NOT BE LOWER THAN LANDING/HAZARD AVOIDANCE

- TIME FRAME FOR LEVEL OF AUTONOMY?
  - CARD/TELEOPERATION EASY
  - SEMI-AUTONOMY RELATIVELY EASY, COMPARED TO OTHER TECHNOLOGY NEEDED FOR HUMAN EXPEDITION
  - CREW STATION AUTOMATION FEASIBLE

QUESTIONS/ISSUES (CONTINUED)

4) If there are no major advancements in the technology for automating the rovers can 4 crew members with operations help from Earth keep 2 rovers scientifically productive?

The answer is no. The MRSR objectives cannot be achieved with only 60 hours of crew time available for operation, even assuming the modest automation developments of shared control for sample aquisition, and CARD for traverse.

5) Is there really a significant advantage of having the astronauts in Mars orbit?

The answer is no, if the stay time is so short. More time on the surface with even a CARD level rover operated from earth would allow the MRSR mission return to be achieved. A semi-autonomous rover is required to achieve the mission objectives with a 240 day time limitation imposed by the current MRSR concept. The Phobos Expedition cannot match the performance of either mode.

6) What sort of scientific background for the crew would be useful for the mission to be scientifically productive?

The crew must be well trained in remote teleoperation to be effective at all. A geology background would help in selecting specific samples from the sites designated by the science group on earth, however, since there may be hundreds of scientists deciding on which sample to return an expert system may be needed to assist even a geology trained astronaut.

7) How will the availability of communication between Earth, crew and rovers affect the operations?

Full time communications is required between the rover and its operators. There is no time for the crew to consult the earth during the sample collection phase and the mission is designed to have the samples pre-designated by the earth while the crew is in transit.

QUESTIONS/ISSUES (CONTINUED)

- CAN CREW OPERATE PRODUCTIVELY WITH NO ROVER AUTOMATION IMPROVEMENTS?
  - ANSWER = NO. MRSR OBJECTIVES NOT ACHIEVED
  - SHARED CONTROL = PROBABLE REQUIREMENT FOR SAMPLING- WILL BE DEVELOPED BY TELEROBOTIC PROGRAM
  - CARD/TRAVERSE TELEOPERATION, SEE ABOVE

- ADVANTAGE OF ASTRONAUTS?
  - MISSION RETURN ABOUT HALF OF 235 SOL MRSR

- CREW SCIENCE BACKGROUND?
  - GEOLOGISTS TRAINED AS TELEOPERATORS

- COMMUNICATIONS REQUIRED BETWEEN ROVER AND OPERATORS?
  - FULL TIME COMMUNICATIONS WITH EARTH DURING SEARCH AND MARK PHASE
  - FULL TIME COMMUNICATIONS WITH CREW DURING SAMPLE COLLECTION PHASE (WHILE CREW IS OPERATING)

PILOTED MISSION MOBILE VEHICLE SCHEDULE

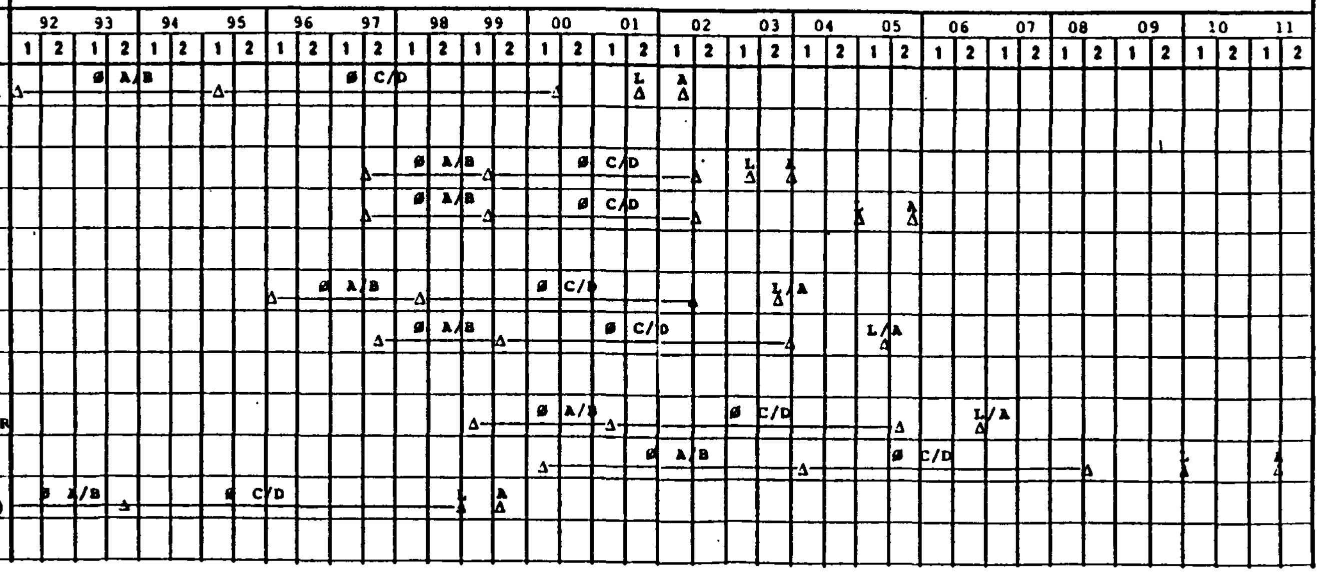
This schedule was derived from the Code Z July 20, 1988 review. It lists the timetables for the rovers/mobile vehicles connected with the four human exploration initiative scenarios. Note that rovers are associated with all four scenarios and that automated rovers (including teleoperated) are associated with three. In fact, automated rovers will almost certainly be a part of all four initiatives, since exploration will be a focus even of the outpost missions. Automated rovers will be needed to extend exploration beyond distances safe for astronauts to drive rovers. Automated mobile vehicles will also be used for site preparation such as digging out sites for habitats before humans arrive, or between their visits.

The Phobos Expedition Mars Rover study has shown that a fair degree of automation is necessary even if a crew is teleoperating a vehicle. Therefore, it appears that automation of mobile vehicles is a generic and fruitful technology area for the Code Z program to pursue.

ORIGINAL PAGE IS
OF POOR QUALITY

PILOTED MISSION MOBILE VEHICLES

	MILESTONES
1	PHOBOS TELOP MARS SURFACE ROVER
2	HUMAN EXPEDITIONS TO MARS
3	MARS TELEOP SURFACE ROVER
4	UNPRESSURIZED DRIVEN ROVER
5	LUNAR OBSERVATORIES
6	UNPRESSURISED DRIVEN ROVER
7	LOCAL/REGIONAL AUTO ROVER
8	LUNAR OUTPOST TO MARS OUTPOST
9	LUNAR UNPRESS'ED DRIVEN ROVER
10	MARS DRIVEN ROVERS
11	MRSR (PRECURSOR FOR HUMAN MARS)
12	



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PHOBOS EXPEDITION MARS ROVER

CONCLUSIONS

- CREW IN ORBIT IS EFFECTIVE FOR TELEOPERATING TRAVERSE AND SAMPLE AQUISITION, BUT 60 HOURS AVAILABLE AT MARS ARE INSUFFICIENT TO TAKE ADVANTAGE OF THAT FACT

- ORBITAL TELEOPERATION FOR 15 DAYS CANNOT SUBSTITUTE FOR MANIPULATION CAPABILITY OF CREW ON SURFACE FOR 15 DAYS FOR SAMPLE COLLECTION (A LA APOLLO)

- OPERATING ROVER IN CARD MODE FROM EARTH FOR SEARCH AND MARK BEFORE CREW ARRIVAL MAXIMIZES SAMPLE RETURN

- HOWEVER, PHOBOS EXPEDITION MARS ROVER DOES NOT MEET MRSR OBJECTIVES, BY A FACTOR OF TWO OR THREE IN NUMBER AND QUALITY OF SAMPLES RETURNED

- SEMI-AUTONOMOUS ROVER CAPABILITY IS A GOOD INVESTMENT, EVEN FOR A PILOTED MISSION

    - SOME DEGREE OF AUTOMATION WILL BE REQUIRED FOR ROVER SAFETY DURING THE SEARCH AND MARK PHASE BEFORE CREW ARRIVAL

    - LANDING HAZARD AVOIDANCE FOR THE ROVER/ASCENT VEHICLE LANDER WILL REQUIRE A HIGH DEGREE OF AUTOMATION

    - THEREFORE, THE ROVER MIGHT AS WELL BE SEMI-AUTONOMOUS

CONCLUSIONS (CONTINUED)

• SEMI-AUTONOMOUS ROVER CAPABILITY IS TECHNICALLY CHALLENGING BUT READILY ACHIEVABLE

- AT LEAST THREE TECHNIQUES FOR OFF-ROAD SEMI-AUTONOMOUS TRAVERSAL HAVE ALREADY BEEN IDENTIFIED AND DEMONSTRATED TO BE FEASIBLE BY JPL, FMC AND CMU

- THE OAST CSTI A&R PROGRAM IS DEVELOPING MUCH OF THE TECHNOLOGY BASE FOR SEMI-AUTONOMOUS ROVER CONTROL AND SAMPLE AQUISITION

- IF OAST'S PATHFINDER PLANETARY ROVER AND SAMPLE AQUISITION, ANALYSIS AND PRESERVATION FUNDING IS APPLIED APPROPRIATELY, SEMI-AUTONOMOUS ROVER CAPABILITY WILL BE DEMONSTRATED BY THE 1992 TIME FRAME

INTERACTIVE ROVER OPERATIONS SCENARIO FOR THE PHOBOS EXPEDITION MARS ROVER

8/8/88

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INTRODUCTION

This report discusses a possible rover operations scenario for a proposed Phobos Expedition (PE) Mars Rover mission. Two rovers and ascent vehicles would be sent to Mars as part of the cargo delivery, landing on 3/27/2002. For the first 14 months, in the Search and Mark Phase, these would be operated interactively from Earth, first taking contingency samples and then sending survey data back to Earth scientists and engineers. In the next phase, a 30-day period called the Sample Collection Phase, astronauts would arrive in Mars orbit and teleoperate the rovers (i.e., operate them under very close manual control). Mission objectives are to cover ground and return samples of scientific interest, in a way comparable to the Mars Rover Sample Return (MRSR) mission proposed for a 1998 launch.

We have taken the tack of coming up with an interactive scenario and comparing the results to the surface portion of a proposed 8-month long MRSR mission with a semi-autonomous rover (ref 2). Essentially our finding is that a PE mission as described could not accomplish the same results as an MRSR in 8 months with a semi-autonomous rover. The 14-month survey period could accomplish comparable objectives, but there is not enough time to gather the corresponding samples in the teleoperations phase.

We have found a useful PE scenario can best be constructed of four elements. (1) The Search and Mark Phase traverse, contingency sample, and image close-up scenarios will be based on the Low-technology, Phase 0 MRSR rover (refs 1,3), because of their similarity in technology level, compatibility of objectives, and long light time delay. (2) Sample Collection phase traverse will be teleoperated. (3) The NASA JPL Telerobotic Test Bed will be used as a basis for a sample acquisition scenario and technology in the Sample Collection Phase, because it is being designed partly to manipulate objects under close human control with short light time delays. (4) Surface mission activities will be as close to the MRSR mission being compared as reasonable (ref 2).

MISSION LAYOUT

Mission layout is depicted in Figure 1. The diagram on the figure depicts the Search and Mark phase. Contingency traverse and

PHOBOS EXPEDITION (PE) MARS ROVER
POSSIBLE ROVER ACTIVITY PATTERN

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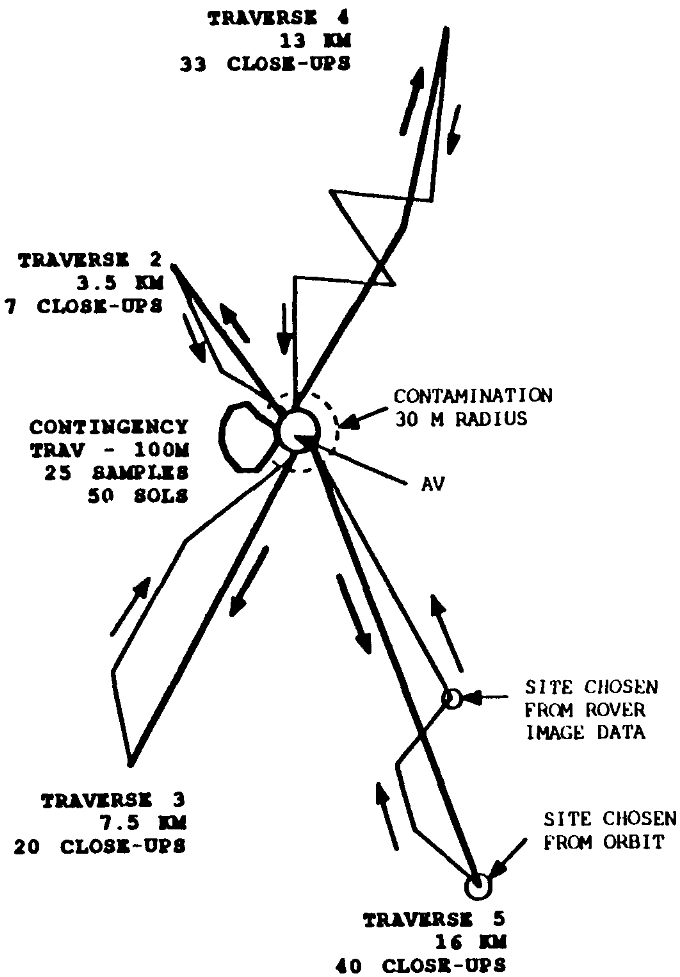

SEARCH & MARK PHASE

14 MONTHS (416 SOLS) AVAIL

DEPLOYMENT, 25 CONTINGENCY SAMPLES 100 CLOSE-UPS PLUS 40 KM ACHIEVED IN 448 SOLS

MRSR 235-SOL RESULTS ACHIEVABLE ONLY MARGINALLY

SAMPLE COLLECTION PHASE

60 HOURS OPS TIME AVAIL

75 SAMPLES + 40 KM REQUIRES 115 HOURS

MRSR 235-SOL RESULTS NOT ACHIEVED BY A FACTOR OF TWO

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FIGURE 1. POSSIBLE ROVER ACTIVITY PATTERN.

outbound traverses are shown with thick lines, inbound traverses with thin lines. To get similarity with MRSR, we attempt to traverse 40 kilometers, take 100 close-ups, collect 25 contingency samples and 75 other samples.

On the 'outbound leg' (ie, going away from ascent vehicle) the rover is commanded to traverse towards known scientically interesting areas, within a few km of the ascent vehicle (AV). It performs science imaging along the way but does not stop for science activities except at those places previously designated from orbit. Science elements analyze these images offline. The science team, concurrently with this analysis, assists in the performing of close-up imaging at the sites predesignated from orbit.

On the 'inbound leg' back towards the AV, science will specify additional areas for close-up which were detected from these scans. Close-up locations average 160 meters apart, and 100 close-ups total will be acquired during the phase.

The need to decide where close-ups need to be taken brings us to the advantage of the double-coverage search pattern. We spend less time with the rover in idle mode, waiting for a science decision on where to take close-ups, if we know that our normal search pattern will take us back through an area where there might be good samples. That decision can be made while the rover is moving or doing other things. The rover will be passing near that area again, so we don't need to do so much backtracking for close-ups as we would if we were trying to do it in one pass.

SEARCH AND MARK PHASE SCENARIO ELEMENTS

Search and Mark Phase Contingency Sample

Per figure 1 and refs 1 and 2, a 100-meter contingency sample traverse requires 50 sols. This includes rover deployment time and science decision time. Rover traverse is done using the same technique outlined below for the main body of the traverse, except more conservative because a contingency sample has not yet been stored. Samples require 7 interactions or 35 hours apiece (Appendix 1, ref 1) and traverse rate is 10 meters per day.

Search and Mark Phase Traverse

Search and Mark traverse is modeled in Appendix 1. This timing is based more or less directly on interactive MRSR scenarios using for the most part 1988 technology (ref 1). Essentially it requires 4 hours for a single round-trip command session, with no science decision making in the loop. From the rover standpoint, this looks like a repeating sequence of: "image-send-wait-receive-move-stop". From the operator perspective, this sequence is perceived as: "receive-decide-command-send-wait."

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This activity mode is what is used exclusively in the outbound traverses from the AV, and while not taking close-ups on the inbound leg. See figure 1.

We will assume that traverse leg lengths average 20 meters. Lunokhod traverse leg length was probably less than 5 meters. We feel that leg lengths of up to 30 meters are feasible in benign terrain with the incorporation of proven automated stereo image technology.[1] Distances over 30 meters will be too hard to manage with quick decision turnaround times, because of the problems of decreasing resolution at distance, where there is any uncertainty about the terrain. Also, some legs will be much shorter than 20 meters due to terrain characteristics.

Daily distance coverage is assumed to be 120 meters, for 6 round-trip command sessions per day.

Search and Mark Phase Close-up Imaging

This activity mode is used while returning to the AV from each Search and Mark traverse.

Broad-brush science data was obtained while traversing outbound. Now it returns to AV while stopping to obtain close-up data of some areas. This is done per Appendix 1. Essentially 4 CARD-type commanding cycles are used per close-up, for a total of 16 hours each.

SAMPLE COLLECTION PHASE SCENARIO ELEMENTS

Sample Collection Phase Traverse

The Sample Collection Phase traverse scenario is modelled as a continuous 20 cm / second teleoperation with continuous video. This converts to 5 seconds per meter, 0.7 kilomenters per hour, or 1.3 hours per kilometer. Traverse is much quicker than in the Search and Mark phase because the terrain is known, there is reduced decision making time and negligible light time delay.

Part of the reason we can move so quickly is we assume that a centimeter-level map is available on-board in binary form. It is programmed in flight from the results of the Search and Mark phase. All routes and sample locations are programmed into it. Only "light-duty" navigation decision have to be made. In a sense the operator is only there to make up for the fact that dead reckoning or automatic navigation techniques are imperfect.

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[1]JPL's Computer-Aided-Remote-Driving (CARD) system.

Some may feel we could move faster than 20 cm / second. In fact the Lunokhod moved at up 60 cm / second. No doubt we will go faster part of the time. We have to allow for more obstacles because the surface of Mars is not so well known and because we will have to take necessary risks to get near samples.

We feel it's reasonable that one person could command a traversing rover for moderate time periods like four hours. We feel that all but one of the 5-person Lunokhod crew could be replaced by automation which is in hand or should be in hand by 1992 - navigator, antenna operator are completely automatable whereas captain, driver and engineer are partially automatable and the rest of the functions could be done by one.[1] We feel that operator stress will be reduced by the additional automation such as graphics.

<u>Sample Collection Phase Sample Acquisition</u>

Specific interactive mode Telerobot Test Bed repair scenarios have not been defined, but would not apply <u>in toto</u> to a natural environment anyway. Enhancements may be necessary for an adaption of the basic teleoperations technology for planetary surface operations. Reflexes would have to be robust and the menu of autonomous routines would need to be particularized to what we expect to encounter and acquire: (1) rocks of various sizes, shapes, masses and orientations with respect to the arms; (2) scoops of soil of various densities, varying forces required to displace varying size particles / pebbles; (3) 1-2 cm rock cores from immovable boulders of various hardness characteristics; (4) some autonomy in drilling will be needed because even with zero light time delay, transmittable force sensing fidelity is insufficient to effect remote control of a drill.

To agree with the reference MRSR mission, we try to acquire 45 rock cores, 15 soil scoops, and 15 pebbles with each rover.

Science does not interact with the acquisition activity because it has already designated the samples, with backups in case of failures, and stored these decisions in an on-board expert system.

Our estimates of interactive sample acquisition times are shown in detail in Appendix 2. Basically, once the rover is in position, it takes 10 minutes to verify that the sample is still desirable using imaging and the expert system, an additional 20 minutes to acquire a soil scoop or pebble with teleoperation, or

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[1]We do not wish to minimize the navigation technology problem. The technology to do it automatically, quickly, reliably and in an operator-friendly manner well integrated with the traverse control terminal needs research even with a low-autonomy rover assumption, if over a few hundred meters is to be covered.

50 minutes to obtain a 1 x 2 cm rock core. We have included sample desirability checks (imaging only) after acquisition in the detailed scenarios, because we think that is realistic, however we have modeled no sample rejection. We assume all samples turn out to be desirable in initial checkout and after acquistion.

Other uses for the expert system interaction include: (1) some sample acquisitions are conditional based on knowledge obtained immediately before or just after acquisition, or from some other sample; (2) complex contingency reprioritization may be needed in case some sample turns out to hard to acquire; (3) it could operate to optimize the returned sample set in the case of foreseeable trade-offs.

OTHER PE ROVER ASSUMPTIONS

Light time delay is assumed to be 20 minutes one-way in the Search and Mark phase and 0.20 seconds two-way throughout the Sample Collection phase.

The crew is a total of four. We assume that there are other duties besides guiding rovers, and therefore only two people are available to run the rovers, each 4 hours per day.

The crew is available to run rovers a total of 15 days. This is a total of 60 hours per rover for sample collection.

The operator may or may not be a full-fledged scientist. He could not fully represent science interests in any case, though added science knowledge is clearly helpful. He is aided in science decision-making by an on-board expert system programmed with both science and engineering information. That system is programmed in-flight based on information discovered in teh Search and Mark phase.

If the crew is not in direct view of the rovers, we assume comm satellites would provide megabit data rates between the rover and operator as needed, preserving less than 200 msec round trip delay.

We assume samples are packaged on-board the rover in a sample canister assembly (SCA). The sample is conveyed from the acquiring arm to the entrance to the SCA by teleoperation, but is thereafter guided through the storage process by hard automation keyed by a single operator command. General purpose robotics (ie, arms with constant adaption to a changing environment) is not required for final storage.

We assume all samples are acquired once time has been spent approaching them, and that acquired samples are always kept after closer inspection. These are undoubtedly not fully realistic assumptions, but they are timesavers, they will facilitate comparison with the 8-month MRSR reference mission since those were the assumptions used there, and throwaway ratios would be guesswork anyway.

The scenario must be responsive to risk. One should not collect samples continually without delivering them back to the AV, since loss of the rovers in the meantime would be close to mission failure. We assume 4 trips back to the AV per rover.

Another risk-handling move is to take contingency samples from beyond the lander contamination area early in the Search and Mark phase. In our scenario, each rover gets 25 contingency samples first. This reduces the sample gathering load on the next phase as well.

We assume the operator never interrupts his shift at the console to call Earth for advice, whether for science or engineering. There simply isn't time to do it any other way -- Earth co-ordination would add many hours of comm and decision time. He may call Earth later to figure out whether it's smart to go back and get that special rock on the edge of the cliff (if time allows), throw out some dubious sample, or run more tests on an acquired sample. Sample acquisition policies could concievably be changed during these offline discussions, however the 15 days during which sample information is rolling in does not provide much time for a carefully considered, multi-disciplinary science review.[1]

A distinct advantage of having human commanding of the rover within a light second or so is that detailed questions can be asked of the rover systems, and answers quickly received. This advantage cannot be fully duplicated by any degree of Earth-directed autonomy unless the MRSR mission were much longer. Many more bits of data can pass between the rover and the operator in a given time period than could be done at Earth-to-Mars distances. Thus the nearby operator is expected to ask for several special images from ad hoc angles, distances, with camera characteristics and in spectral bands he manually selects. The operator can interact directly with his local expert systems, asking questions interactively to reduce doubts, and queries to the rover could be generated either by the expert systems in

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[1]There is an obvious comparison to the familiar Apollo sample acquisition strategy, where samples were gathered at high speed with the minimum Earth co-ordination. We think that even less co-ordination than Apollo would be the best way for a short, piloted PE mission, because of the light time delay. The difficulty with the comparison is in the science area: Mars is more complex than Luna, we know less about it in proportion to what there is to know, therefore more science disciplines ought to be represented in the decision process (e.g., exobiology, meteorology, sedimentology). All this means that even if we could get as many samples and cover as much distance as an Earth-directed mission spanning several months, the science results are going to be less.

response to the interaction or by the operator guided by the expert system. Depending on the operator's scientific expertise and the depth of reasoning embodied in the expert system, this factor operates to increase value of the science returned over an MRSR. Earth scientists asking the same number of questions adaptively would be much more costly.

RESULTS

Figure 1 shows a summary of the results in the form of a mission layout diagram. Essentially, the conclusion is the 14-month Search and Mark phase can come close to duplicating what MRSR could do in less time, but there is not enough time in the 30-day Sample Collection phase.

Search and Mark Phase Results

Phase totals for one rover:

Initial deployment, etc.	10 sols	
Contingency traverse	40	
16 km outbound traverse	133	
24 km inbound traverse	200	
100 sample close-ups	65	
TOTAL	448 sols	
AVAILABLE	416 sols	
OVERRUN	32 sols	(8%)

Sample Collection Phase Results

For one rover:

Traverse 40 km (@ 20 cm / sec)	55 hours	
Gather + check 45 rock cores	45	
Gather + check 15 soil scoops	7.5	
Gather + check 15 pebbles	7.5	
TOTAL REQ'D	115 hours	
AVAIL	60 hours	
OVERRUN	55 hours	(92%)

CONCLUSIONS

A rover interactively directed from Earth for a period of 14 months could concievably perform surveying over a similar area as could a semi-autonomous MRSR rover over 8 months. This includes 40 km traverse distance, with 100 close-ups obtained.

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The Rover would be hard put to interactively acquire the 75 samples designated for acquisition in the total 60 hours available for them to be operated, assuming a random distribution of the samples in the surveyed area, and a reasonable number of trips back to the ascent vehicles for unloading. They miss the goal by a factor of 2.

For the highest speed possible teleoperated navigation, the continuous video must be integrated with a binary terrain map with deirections on where to go, when to stop, what speed to use, all preprogrammed. These decisions can be made in advance and preprogrammed from the data in the Search and Mark phase. This is a technology advance.

Unlike an MRSR, there would be no time for sample analysis beyond spectral imaging of the surface of the samples with final decision by an expert system or the astronaut as to science desirability.

There is no time for science feedback of lessons learned into the ongoing acquisition process, ie, no science adaption in any decisions to acquire.

An advantage over MRSR is having the operators nearby for quick turnaround ad hoc question-answer sessions with the expert systems and to ask for more data from the rover. Since the questions which can be fruitfully asked are limited by the imaging capability of the rover, we feel this advantage does not outweigh the disadvantage of not having time for chemical and mineralogical sample analysis or scientists to interpret it.

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2. Dias, Version 1.01 Rover Surface Operations Scenarios, 6/29/88
3. Muirhead, ed., MRSR Rover Phase 0 Design Data Book, 3/31/88

ACKNOWLEDGEMENTS

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APPENDIX 1
PHASE A TRAVERSAL AND SAMPLE CLOSE-UP SURVEY SCENARIOS

These traversal and sample close-up scenarios apply to Phase A only. They assume a long light time delay (20 minutes one way) and are based mainly on the JPL MRSR Phase 0 interactive rover scenario with Computer-Aided Remote Driving (CARD) traversal technology (ref 1).

TRAVERSE CYCLE

<u>Scenario for a single command cycle to traverse 20 meters on average. Science is acquired but not processed during the cycle.</u>

Earth engineering team decision + sequencing	80 minutes
Comm (w/20 min delay)	25
Rover move	5
Rover perform science and navigation imaging	70
Comm (w/20 min delay)	40
TOTAL about 4 hours	220 minutes

SAMPLE CLOSE-UPS

<u>Each close-up sequence is assumed to require 16 hours. The amount of work and number of decisions is approximately 4 times that of an individual CARD cycle above. The scanner covers out to 60 meters around the rover. We assume that an average sample is one CARD cycle (20 meters) off the prearranged track. It takes one cycle to approach the sample, one to re-orient when close in, one to perform the close-up imaging, and one to return to the path.</u>

Leave path towards sample (20 m)	4 hours
Re-orient rover in sample vicinity	4
Close-up imaging & transmit	4
Return to path (20 m)	4
TOTAL	16 hours

CONTINGENCY SAMPLES

Science sample selection	8 hours
Orient rover	4
Re-orient rover	4
Acquire	4
Science validate sample	8
Store sample	4
Validate storage	3
TOTAL	35 hours

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APPENDIX 2
SAMPLE COLLECTION PHASE SAMPLE ACQUISTION SCENARIOS

These scenarios are constructed using the scenario elements in Appendix 3 as building blocks. They apply only to the teleoperation of the rovers by the orbiting astronauts, not control from Earth.

CLOSE-UP INVESTIGATION OF PROPOSED SAMPLE

Teleoperations scenario for a not-yet-acquired sample close-up investigation. Used to verify the sample is indeed desirable and matches previously determined science description of what was desired from the location.

Major arm move	50 seconds
Grasp close-up camera	40
Major arm move	50
Close-up image	100
Major arm move	50
Close-up image	100
Major arm move	50
Close-up image	100
Major arm move	50
Stow close-up camera	40
Expert sys interact	60 (i.e., decide whether to acquire)

TOTAL 11 min	690 seconds

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ACQUIRE + CERTIFY PEBBLE / SOIL SCOOP

<u>Teleoperations scenario for a pebble sample acquistion (includes close-up science certification imaging of the acquired sample.)</u>

Major arm move	50	seconds
Attatch gripper to arm	40	
Major arm move	50	
Contact sample with tool	80	
Grasp (close gripper)	80	
Lift pebble	80	
Major arm move	50	(swing sample near close-up camera)
Image sample close-up	110	
Major arm move	50	
Image sample close-up	110	
Major arm move	50	
Image sample close-up	110	
Expert sys interact	60	(decide whether to keep)
Major arm move	50	
Xfer sample to SCA	80	
Major arm move	50	
Detatch/Stow gripper	40	
Major arm move	50	(stow arm for rover movement)
TOTAL 20 min	1190	seconds

ACQUIRE + CERTIFY 1 CM X 2 CM ROCK CORE

<u>Teleoperations scenario for a rock core acquistion (includes close-up science certification imaging of the acquired sample.)</u>

Major arm move	50	seconds
Attatch gripper to arm	40	
Major arm move	50	
Contact sample with tool	80	
Drill	1820	
Lift core	80	
Major arm move	50	(swing sample near close-up camera)
Image sample close-up	110	
Major arm move	50	
Image sample close-up	110	
Major arm move	50	
Image sample close-up	110	
Expert sys interact	60	(decide whether to keep)
Major arm move	50	
Xfer sample to SCA	80	
Major arm move	50	
Detatch/Stow gripper	40	
Major arm move	50	(stow arm for rover movement)
TOTAL 50 min	2930	seconds

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APPENDIX 3
SAMPLE ACQUISTION SCENARIO ELEMENTS

Scenario elements are the building blocks from which scenarios in Appendix 2 are constructed. A teleoperation with a borderline significant light time of 200 msec is assumed.

MAJOR ARM MOVE

Teleoperations scenario element for a major arm move. Arm is being moved through an arc by the operator with the aid of a predictive display simulation of projected arm locations before, after, and during. Assumes there is a path well clear of any potential obstacles.

Operator view wide-angle image of overall situation	20 sec
Command + Rover perform move	30
Operator compare simulation with actual	1
TOTAL 0.8 minutes	51 sec

ATTATCH / DETATCH TOOL

Teleop scenario element for the attatchment of any tool, camera or device onto the general purpose arm, or detatchment of same. Assumes automatic macro performs most of the activity but is verified by visual inspection of feedback. Arm is positioned near the tool storage location before and after.

Operator view wide-angle image of overall situation	20 sec
Command with autonomous macro + rover perform	20
Operator compare simulation with actual	1
TOTAL 0.7 minutes	41 sec

SIMPLE TOOL INTERACT WITH SAMPLE

Teleop scenario element for a single end effector autonomous macro command to interact with sample (ie. contact sample with tool, grasp sample with gripper, stow sample in SCA, lift sample). Assumes autonomous reflexes, force sensing on rover, not enough time for operator to react to force changes interactively.

Operator view wide-angle image of overall situation	20 sec
Command with autonomous macro + Rover perform	60
Operator compare simulation with actual	1
TOTAL 1.2 minutes	81 sec

INTERACTIVE ROVER OPERATIONS SCENARIO FOR THE PHOBOS EXPEDITION MARS ROVER

8/8/88

ACQUIRE ROCK CORE SAMPLE

Teleop scenario element for a monitored operation autonomous command to obtain 2 cm core drill sample. Assumes autonomous reflexes, force sensing on rover, not enough time for operator to react to short term force changes interactively. At start, tool is already in contact with rock, and finishes with core detatched and in tool.

Operator view wide-angle image of overall situation	20 sec
Command drill + Rover perform	1800
Operator compare simulation with actual	1
TOTAL 30.3 minutes	1821 sec

CLOSE-UP OF PROPOSED SAMPLE AREA

Teleoperations scenario element for a not-yet-acquired sample close-up investigation (close-up camera is already being held by the general purpose arm, near the proposed sample, but not yet oriented exactly to the sample)

Operator view wide-angle image of overall situation	20 sec
Command camera configuration with autonomous macro	10
Command + rover perform camera positioning + image	10
Operator view image, expert sys interact, decide next	60
TOTAL 1.7 minutes	100 sec

CLOSE-UP IMAGE / CERTIFY ACQUIRED SAMPLE

Teleoperations scenario element for an acquired sample close-up investigation (sample is being held by the general purpose arm, near a body-mounted close-up camera, but the sample still needs orientation to the camera)

Operator view wide-angle image of overall situation	20 sec
Command camera configuration with autonomous macro	10
Command + rover perform sample positioning + image	20
Operator view image, expert sys interact, decide next	60
TOTAL 1.8 minutes	110 sec

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OCT 88

SPACE EXPLORATION ANNUAL REPORT

VOLUME III
COST UNDERSTANDING

KELLEY CYR
JOHNSON SPACE CENTER

OCTOBER 1988

LUNAR
MARS
EXPLORATION

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1. EXECUTIVE SUMMARY

This report describes an attempt to develop a better understanding of the cost of civilian space initiatives, especially those involving manned exploration of the solar system. The objective is to update the assumptions and art of costing major initiatives that have a characteristic of being at the concept stage, with implementation far into the future where experience and techniques are very different. The report will discuss the questions of problem identification, knowledge acquisition, conceptualization, formalization and implementation.

1.1. PROBLEM IDENTIFICATION

The majority of the space activities in the United States are funded by the Federal Government, which means that growth in the space program is constrained by the Federal Budget. The cost of new space initiatives, especially those involving manned exploration of the solar system, may be very high. Compounding this problem is the fact that most NASA spacecraft are now designed to have a very long operational life. Therefore, the cost of any new developments must be added on to the ongoing cost of operating and maintaining existing systems. In some worst case scenarios, it is expected that the NASA budget would have to triple by the year 2000 in order to accommodate all of the projected missions. Since the general public does not perceive a great deal of benefit from the space program, it is unlikely that there will be strong public support for a large, sustained increase in public expenditures on civilian space initiatives.

A number of trends in the development of space hardware have been identified. Projects tend to be bigger, more expensive, and take longer to develop. On the other hand, more cost effective approaches to development have been implemented. The industry is more efficient due to advances in worker productivity. Projects are developed with fewer prototypes and less testing. Traditional cost estimating methods that do not take these changes into account will produce inaccurate cost estimates. Furthermore, the factors that really drive cost, such as institutional and management factors, are not usually included in the variables that drive traditional cost models.

Most of the cost models used by NASA have been developed on very small subsets of data (e.g., manned spacecraft). Since there are only six manned spacecraft developed by NASA, the cost analyst is limited to only one or two explanatory variables. An approach has been developed that allows many different types of spacecraft to be included in the database. This allows many explanatory variables to be utilized. It also enables direct analysis of the cost differences between different development environments. This approach to analyzing the factors that cause cost will lead to better estimates in the future and better approaches to managing the cost of programs.

The new initiatives being considered for the human exploration of space pose some special problems for cost estimating. The estimates given for these missions by traditional models tend to produce cost estimates that are considered too high to make the projects politically viable. Since the new initiatives are not expected to start until late in this century, the cost models must predict much farther into the future than is normal. Some of the initiatives may also require technology that has not been developed yet; hence, there are no historical analogies on which to base estimates. Finally, the programmatic factors, rather than hardware characteristics will probably be the major cost drivers. The cost estimating techniques being developed will be able to deal with all of these considerations.

1.2. KNOWLEDGE ACQUISITION

The knowledge acquisition process consists of two major tasks; acquisition of general problem solving knowledge and acquisition of information specifically related to the problem of cost analysis for new space initiatives. The general problem solving knowledge includes: statistical analysis tools; decision making theory; cost estimating; schedule estimating; cost spreading; project management; systems engineering; forecasting; chart-making; and, other general skills. Fortunately, most of these functions are readily available in various micro-computer products. The major function of knowledge acquisition is the collection of domain specific knowledge. This includes data pertaining to the domain, algorithms for solving specific problems, and heuristic methods.

Collection of cost data is a continuous process. So far, data has been collected on almost every NASA and unclassified DOD spacecraft. In addition, data has been collected on many other types of large scale research and development projects such as aircraft, missiles, ships, and tanks. The raw data is loaded in a computer database and then standardized to a common format. The knowledge of algorithms and heuristics is being acquired by the following: the study technical manager and study leader are cost estimating experts; frequent reviews are held with other cost experts around NASA; members of the study team participate in professional societies, review literature and attend conferences on related subjects.

1.3. CONCEPTUALIZATION

In order to accurately make any forecast using mathematical or statistical modeling, several conditions must be meet. First, the structure of the model; i.e., the nature of the relationships must be identified. Second, the parameters of the equation that are expected to vary, as input or outputs, need to be specified. Third, those factors that remain constant must be identified and estimated. Finally, the conditions under which the structural equations and parameters remain stable must be specified and tested. Only when thorough testing has indicated stability and accuracy over the expected range of forecasting requirements can a model be put to operational use.

The statistical model identified in this paper is a fairly good predictor of general hardware development cost. As such, it proves that using many different programs as a data base for estimating a cost model is a viable concept. The use of many data points from different technology domains has several advantages. First, it increases the number of degrees of freedom in the statistical analysis which allows more explanatory variables to be used.

Second, the wider range of data provides a deeper insight into the nature of the relationship between cost and various program factors. For example, a limited analysis of spacecraft data may lead to the conclusion that quantity elasticities are always greater than unity. In fact, production economies of scale should be achieved once the initial prototype stage is passed.

Third, a model based on a wide range of technologies should be more suitable for estimating the cost of new designs that may have no historical analogies. Finally, validating the model over different time periods may improve the confidence in estimates made far into the future. The model described here demonstrates that such a model can be constructed and will predict cost within fairly reasonable bounds.

In addition, several economic conclusions can be drawn from the statistical model. The analysis shows that significant economies of scale with respect to weight exist for nearly all types of development hardware. The more complex the hardware, the greater the economies of scale. Also, the lower the weight of a subcategory, the greater the economies of scale are for that subcategory. Some classifications, such as ships, even have diseconomies of scale with respect to weight. The estimated elasticity of cost with respect to weight ranges from 0.43 to 1.30 with an average value of approximately 0.65.

Economies of scale with respect to unit quantities also are evident. The range of estimated elasticities is very wide, from 0.3 to 1.17 with the average around 0.58. Some types of systems have diseconomies of scale. These are mostly very low production quantity systems such as spacecraft. The conclusion is that a modified learning curve may be appropriate.

The use of a Culture variable is proven effective for combining different technologies in the same data base. A methodology for deriving a quantitative measure of culture is presented and shown to produce good results. For future space developments, Culture may be the most significant variable the cost analyst has to select. Weight and quantities will usually be given, but the particular hardware may not fall into any of the historical subcategories. It may be possible to estimate Culture for future programs using deterministic methods, such as a function of the ratio between weight and quantity. Another possible method of estimating a new culture would be interpolation or extrapolation of existing cultures.

The inclusion of a time based variable causes the effects of time to be removed from the other variables in the model. Hence, the model can be used for long-range planning if the future effect of time can be predicted. It was found that the cost of programs is increasing with time, even after the effects of inflation are excised. The time related cost growth is not at a constant rate, however. The magnitude of cost growth appears to be from 0.0 to 3.0 percent per year. The exact nature of this time related phenomenon is not yet understood although it is believed to be a combination of increasing performance, complexity and technology offset by improving productivity and development methods.

Finally, the benefits of design inheritance are clearly demonstrated. Substantial reductions in cost from using existing designs rather than starting from scratch are evident. Cost savings of about 22 percent for each subsequent generation are predicted by the model. This fact has been used to great advantage on military acquisition programs and should be incorporated whenever possible in the space program.

The statistical model does have some deficiencies. Most of the problems result from the wide range of estimated coefficients for subcategory models. The model of all data must effectively average these coefficients which results in erroneous estimating at the subcategory level. In addition, it was found that the modeling of time related behavior (e.g., inflation, productivity, technology, et cetera.) is inaccurate. The current model assumes that the rate of change is constant but; in reality, it varies.

The combination of these two deficiencies makes the current statistical model unsuitable for long-range cost estimates of advanced space programs. Although the basic technique demonstrated here is sound, it must be refined even further to produce acceptable cost estimates. The specific weaknesses of the statistical model have been identified and potential solutions will be implemented in the future.

1.4. FORMALIZATION

Formalization of the Advanced Missions Cost Model will consist of encoding the cost and schedule domain knowledge in an expert system containing the appropriate inference engines. The resulting cost model will be made available to the program agents for cost estimating use as they see fit. The cost model will also be linked into the new initiatives scenario integration process to enable the top level parametric estimating of new initiative cost and schedule. Requirements have been formulated, based on conceptual models, for the formalization of the cost model. This includes the requirements for input information, general output requirements, a conceptual description of the model functions, and a software baseline. As currently envisioned, the cost model will run an on IBM-XT compatible microcomputer using the Mainstay database management system.

1.5. IMPLEMENTATION

The Office of Exploration is making a serious effort to minimize the cost of its programs while retaining the high level of mission success that is expected of NASA. A blue ribbon panel was recently commissioned by the President to study Department of Defense acquisition programs and there were eight characteristics the "Packard Commission" found existed in successful system acquisition programs. In addition to the "Packard Commission" recommendations, five other recommendations are being considered. These thirteen characteristics will be used as groundrules for the implementation of new civilian space initiatives. They are as follows:

1. **Acquisition cost realism along with unit production cost as a significant design requirement.**

In order to achieve program stability, it is essential that a program realistically budget for development and production costs at the start of full scale development. The development program manager must address the risk associated with development of the system. In this way, it is possible to determine if the program is affordable. The affordable cost level can then be used to establish a unit production cost design objective for the system which essentially becomes the "design-to-cost" objective.

2. **Developing prototypes (for both cost and performance) and early, extensive testing.**

Prototyping has historically been done; but, its primary objective was proving that something is possible. Traditionally, the system was then almost totally redesigned during the development phase; such that the unit production cost of the system rose dramatically. The prototyping should be done with a system that is close enough to the ultimate production design such that one is able to make a good estimate of the cost and the performance of the ultimate system.

After moving out of the prototype phase, history has shown that one of the ways programs have saved money is to reduce the number of test units and the amount of test time. This is another example of short-sighted attempts to save development dollars at the expense of what ultimately becomes a stretched out and overrun program. Clearly, if you can't afford to do adequate testing early on, then the program is doomed to problems later.

3. **Planned product improvements and maximum use of proven components and subsystems (especially commercial items).**

It has been found that when existing systems are modified, rather than new systems started from scratch, the time and cost for development are both dramatically reduced. The concept here is not to start off a new program assuming that it requires a new set of avionics, a new engine, et cetera; but, to independently develop each of these subsystems with standardized interface specifications so they can be plugged in when they are proven and to insert these upgrades at an appropriate block point in the production cycle.

Consistent with the idea of using proven systems and subsystems is the concept of systems making maximum use of commercial subsystems and components. Both the Packard Commission and a recent Defense Science Board Task Force emphasized the dramatic benefits, particularly with electronics, that could be achieved through greater use of commercial components. The Defense Science Board study found that systems built from commercial components would have costs that were between two and eight times cheaper overall, with comparable or better reliability; and, that these systems could be acquired between two and five times more rapidly as a result of using off-the-shelf, proven, commercial components.

4. Presence of a continuous alternative.

What makes a market economy operate effectively is the continuous presence of an alternative for the buyer; such that, if a supplier reduces his quality or raises his price, the buyer can go elsewhere. However, the typical program has no such alternative present. Rather, there is usually a fierce competition for the initiation of a development program and this is followed by sole-source environment throughout the many years of the development and production phases of the program.

Occasionally, the presence of continuous competition in the development and/or the production phase has been tried and the results have been very impressive. For those programs that had "dual sourcing" during the development phase of a set of Army programs, the R&D costs were better controlled; however - most importantly - the production costs were dramatically reduced as a result of the competitive development phase; thus, far more than justifying the increased development cost for the second source. Equally significant, it was found that, on the average, the performance was much higher on those programs that had been dual sourced.

5. Short and stable schedules for development and production.

All of the successful programs studied began by using previously demonstrated technology and by realistically estimating their program costs. They then fully funded the necessary dollars and maintained the program's initial requirements throughout the program's development. This combination - of demonstrated technology, cost realism, and stable budgets and requirements - allowed them to achieve extremely short development and production schedules. Thus, they realized maximum economic efficiency and got the new systems fully deployed in the fastest possible time.

6. **Experienced, small staffs, with clear command channels and limited reporting.**

In a typical US Government program, the senior managers frequently are quite inexperienced and often rotate a number of times during the development phase. Often, this inexperience is compensated for by having a relatively large staff of people, all responsible for small pieces of the overall activity. This problem is further compounded by having a very large number of layers above the program office, through whom all decisions must be passed. One estimate was that on a seven year development program, over three and a half years of the time was taken up with decision making and the rest with actual development. Finally, it is estimated that something like 20 percent of a typical DOD development program's cost is devoted to reporting on the program. By contrast, in those programs that were successfully run, the primary reports required were deviation reports in which thresholds were established for cost, schedule, and performance and, as long as the program stayed within these limits, very little reporting was required.

7. **Effective communication with users for cost/performance tradeoffs.**

There is a myth that exists within the acquisition world that there is an initial "requirement" established for a system, and then this is turned over to the development community to pursue. By contrast, on a successful program, it is recognized that there must be a continuous trade-off made between the user and the developer in terms of the impact of varying requirements on development and production costs and schedules.

8. **Early development phase funding for production and support considerations.**

Traditionally, the development phase of a new system focuses almost exclusively on that phase. Then, later, we find out how much it will cost to produce and maintain it. However, this is inefficient, in both time and dollars, particularly with current trends towards new, computer integrated manufacturing technologies. If funds are available up-front to include production considerations as part of the original design job, then one can make the transition from computer-aided design through computer aided manufacturing and into computer-aided logistics in a smooth and continuous process. This requires an engineering/production/support team in the early design phases that is more than simply the lip service that has traditionally been given to this area. It requires that the design be continuously modified in order to take producibility and maintainability directly into account.

9. **Mass production techniques should be used as much as possible at every level of hardware (and software).**

The use of mass production techniques to reduce cost is so widely known that it need not be elaborated on here. This lesson has apparently been lost on NASA, however, since the agency still insists on building virtually every spacecraft from scratch. Even in relatively small quantities, mass production can save money by spreading design, prototype, test, and tooling cost. Further economies can be gained by the production learning that occurs with repetitive tasks. Although some small loss in performance may be the price of commonalty, this could be offset by the increased confidence in performance from a proven design.

10. Technology should be pushed forward at realistic rates.

Part of the NASA mission is to develop new technology; but, when a program depends on the successful development of a technology that cannot realistically be accomplished, the result is cost overruns and schedule delays. In order to avoid this, programs should depend on technology that is already demonstrated, or that can be proven well within the required schedule.

2. INTRODUCTION

The analysis and understanding of cost for future manned space exploration missions is a difficult and complex subject. Beginning in 1986, the need was recognized for development of a new cost estimating method specifically design for estimating programs in the conceptual design stages.

On July 21, 1986 a request for proposals was distributed to industry for the development of an Advanced Missions Cost Model (AMCM). The contract would be for a period of four years with a total budget of no more than $850,000. The study effort would be under the technical guidance of the Cost Analysis Office of the Johnson Space Center and was funded by the Advanced Program Office of NASA Headquarters.

After reviewing proposals by seven firms, ECON Incorporated was selected for negotiations and awarded the contract on March 16, 1987. In addition to the contracted effort, a small in-house effort was also begun. During the winter of 1987, NASA created the Office of Exploration (Code Z) at NASA Headquarters with responsibility for the human exploration of space and sponsorship of the Advanced Mission Cost Model was transferred to Code Z.

On December 22, 1987, Code Z issued a mini-budget call which described a work breakdown structure (WBS), study requirements, and an organization for future work. The complete WBS for Code Z is given in Figure 2.1 on page 2-2. The cost understanding work was designated a Special Assessment Study under WBS 3.4. The mini-budget call described the following requirements for the cost understanding task.

2.1. COST UNDERSTANDING REQUIREMENTS

REQUIREMENT: "In-depth assessment of the costing methodology, with the objective of updating the assumptions and art of costing major initiatives that have a characteristic of being at the concept stage, with implementation far into the future where experience and techniques are very different."

REQUIREMENT: "Task involves collecting experience from agency-wide and industry-wide sources, fitting that experience to the environment of the initiatives."

REQUIREMENT: "Key product, in addition to the costing techniques analysis, is a tailored method ('cookbook') that can be used by the human initiatives program agents for costing that would include the programmatic and specific agency assumptions on environment in which the initiatives will be developed."

2.1 WORK BREAKDOWN STRUCTURE

1.0 Office Of Exploration Management
- 1.1 Mission Studies Management
- 1.2 Technology Integration & Coordination
- 1.3 Science Studies Management
- 1.4 Program Support & Special Projects

2.0 Integration
- 2.1 Mission Analysis & Synthesis Support
- 2.2 Space Transfer Vehicle Integration
- 2.3 Orbital Node Analysis & Integration
- 2.4 Planetary Surface Systems Definition & Integration

3.0 Special Assessment Studies
- 3.1 Power Systems
- 3.2 Propulsion Systems
- 3.3 Major In-space Operations Feasibility
- 3.4 "Initiative" Cost Understanding And Methodology Development
- 3.5 A&R/Expert System/AI
- 3.6 Advanced Life Support Concepts
- 3.7 Communications Data Capacity/Handling

4.0 Special Emphasis Studies

5.0 Category 1 Activities
- 5.1 ETO Transportation Analysis
- 5.2 Space Station Evolution Analysis
- 5.3 Mars & Lunar Science & Engineering Data Precursor Mission Analysis
- 5.4 Life Science Precursor Missions & Analysis
- 5.5 Technology Prerequisite Programs & Analysis
- 5.6 Operations Capability Extensions & Analysis

6.0 Science Program

REQUIREMENT: "Methodology provided to each program 'agent' for a distributed cost estimate of the program/scenario pieces."

During 1988, a number of progress reviews were held with the Code Z staff and it was determined that the following requirements should be added.

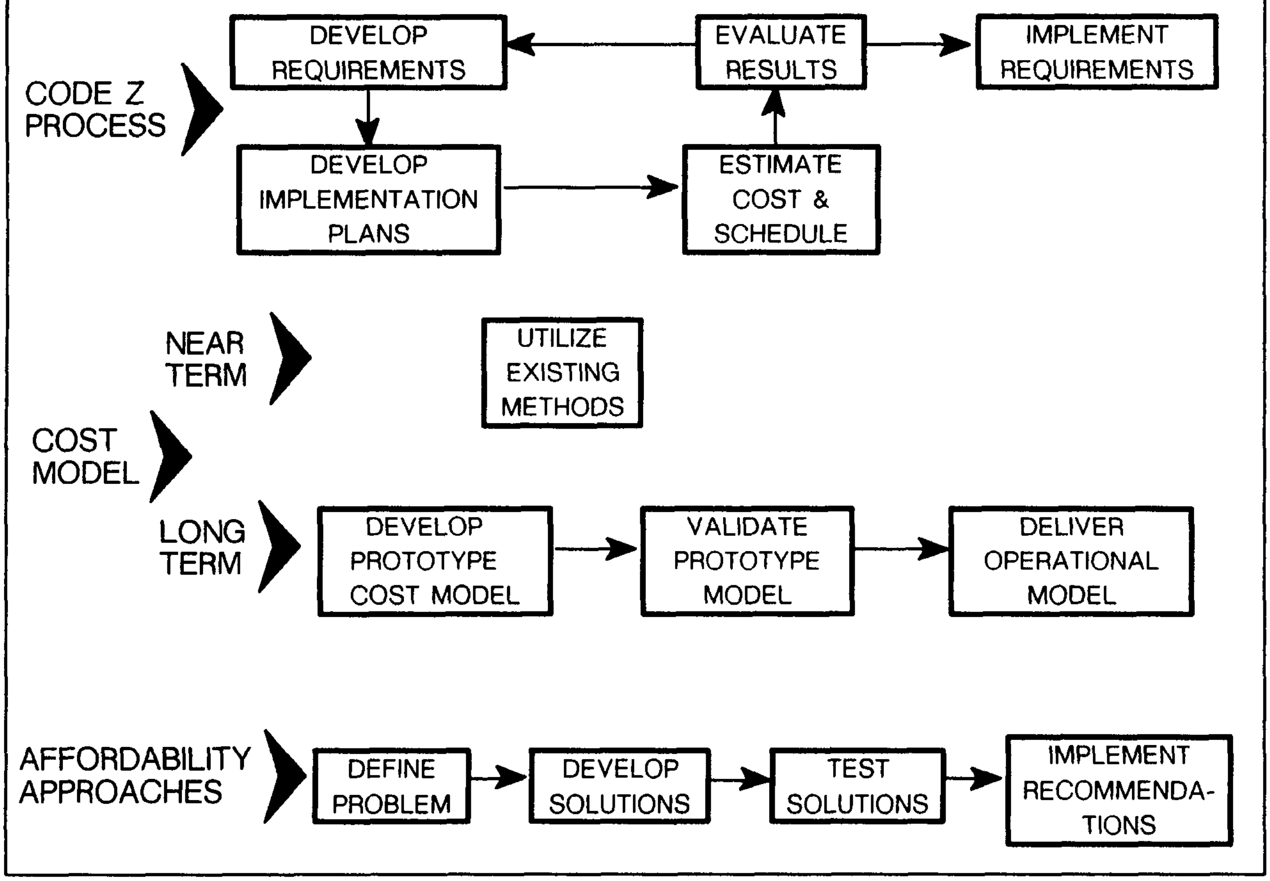
REQUIREMENT: Develop a plan for changing the environment in which exploration programs are developed in order to make them more affordable.

REQUIREMENT: Provide an interim capability to do relative cost and schedule estimates for exploration case studies beginning in FY 1989.

2.2. STUDY APPROACH

An overall study approach has been developed to meet the requirements issued by Code Z. The flow chart depicted in Figure 2.2 on page 2-3 summarizes the study approach. The top level describes, in very general terms, the Code Z implementation process. Tied into the implementation process is the work of developing a new cost model. Also included in Figure 2.2 is

2.2 STUDY APPROACH



the process required to develop affordability approaches for new initiatives.

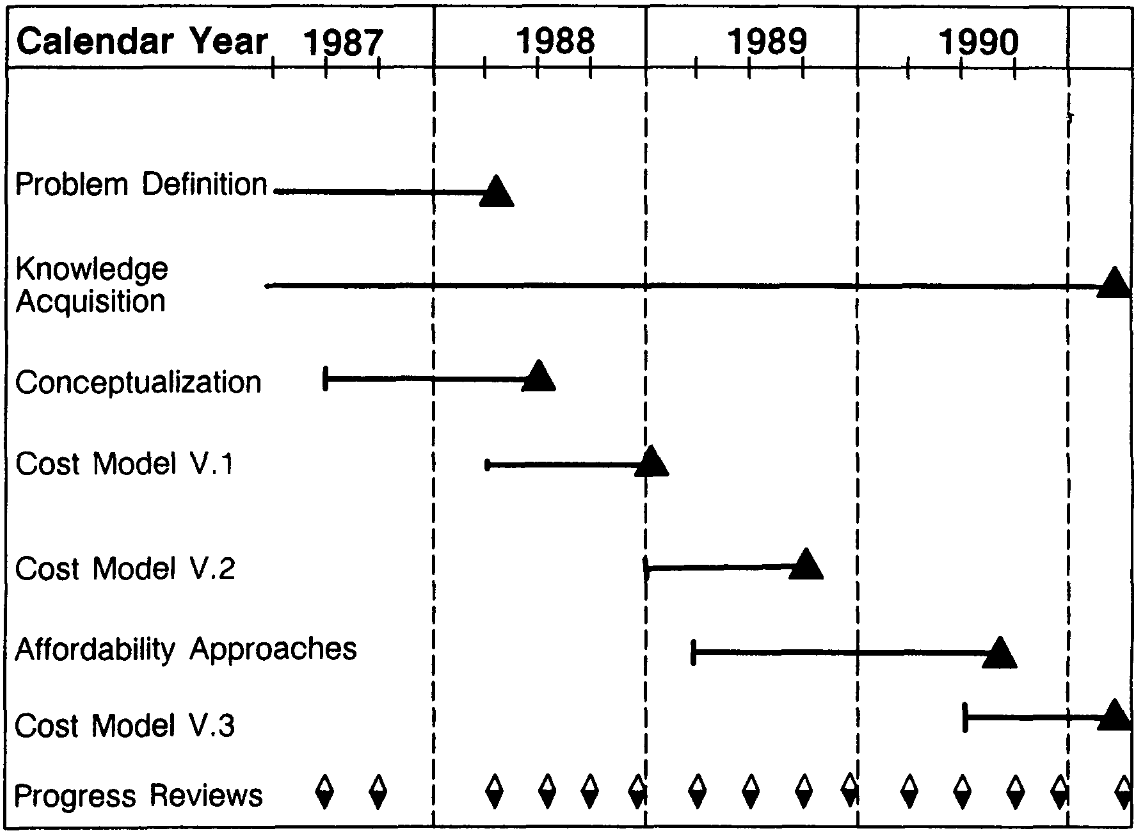
2.3. LONG RANGE STUDY PLAN

The long range study plan for cost understanding is given in Figure 2.3 on page 2-4. Cost model version 1 is a prototype for testing purposes. The major milestones are the delivery of the version 2 cost model in the fall of 1989 and the version 3 model in 1991. Version 2 is an operational test model and Version 3 is the final operational model.

2.4. NEAR TERM STUDY PLAN

The near term study plan is given in Figure 2.4 on page 2-5. The near term schedule is intended to link up with the cyclical definition process established by Code Z. However, due to funding limitations, the current schedule for cost model development will lag slightly behind the Code Z process. The major milestone, again, is the delivery of cost model version 2 in the fall of 1989 with the first "real" cost estimate following.

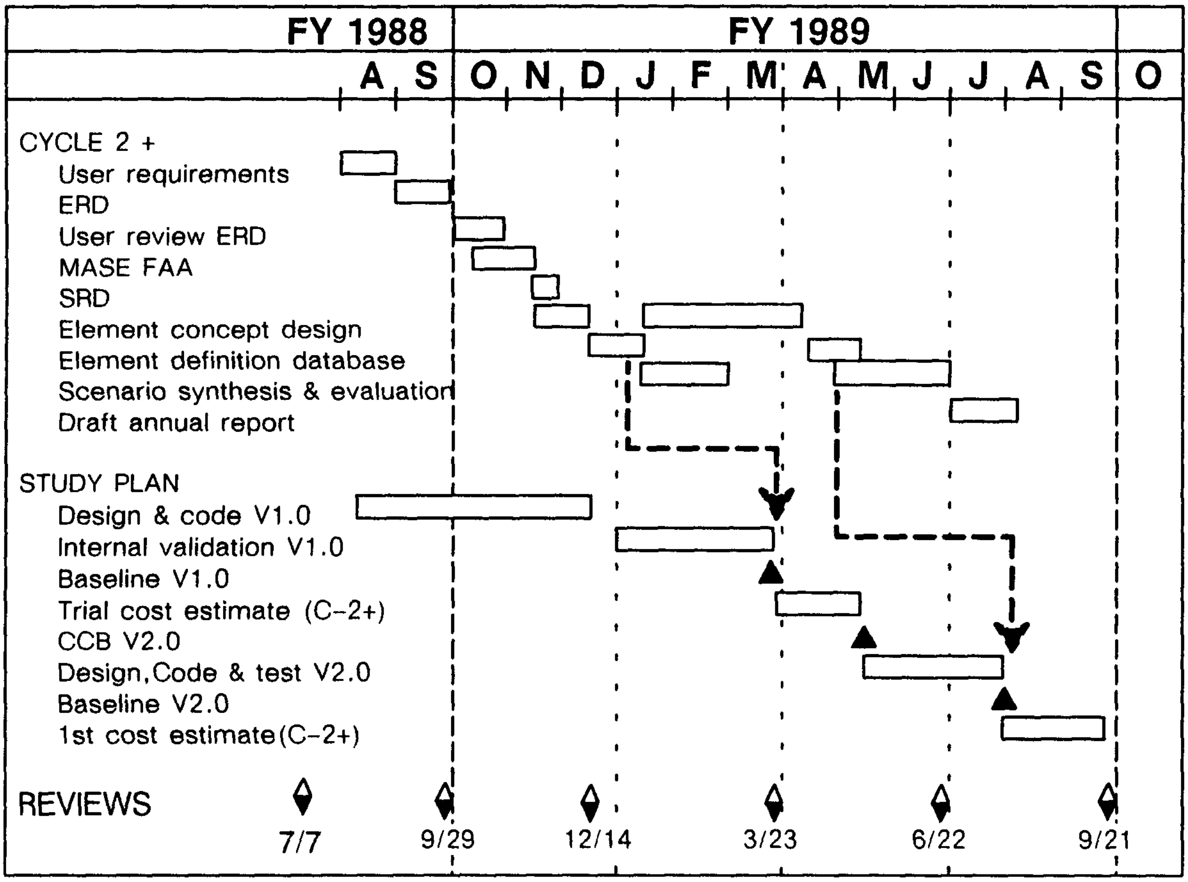
2.3 LONG RANGE STUDY PLAN



2.5. STUDY GROUND RULES

For large scale human exploration missions, current cost estimating methods do not give satisfactory results. Therefore, no cost estimates will be produced for external distribution until fiscal year 1990. This will allow adequate time for development and testing of new cost methods.

2.4 NEAR TERM STUDY PLAN



3. PROBLEM IDENTIFICATION

What is a cynic? A man who knows the price of everything and the value of nothing.
- Oscar Wilde

REQUIREMENT: "In-depth assessment of the costing methodology, with the objective of updating the assumptions and art of costing major initiatives that have a characteristic of being at the concept stage, with implementation far into the future where experience and techniques are very different."

The majority of the space activities in the United States are funded by the Federal Government, which means that growth in the space program is highly constrained. The cost of new space initiatives, especially those involving manned exploration of the solar system, may be very high. Compounding this problem is the fact that most NASA spacecraft are now designed to have a very long operational life. Therefore, the cost of any new developments must be added on to the ongoing cost of operating and maintaining existing systems.

In some worst case scenarios, it is expected that the NASA budget would have to triple by the year 2000 in order to accommodate all of the projected missions. Since the general public does not perceive a great deal of benefit from the space program, it is unlikely that there will be strong public support for a large, sustained increase in public expenditures on civilian space initiatives.

A number of trends in the development of space hardware have been identified. Projects tend to be bigger, more expensive, and take longer to develop. On the other hand, more cost effective approaches to development have been implemented. The industry is more efficient due to advances in worker productivity. Projects are developed with fewer prototypes and less testing. Traditional cost estimating methods that do not take these changes into account will produce inaccurate cost estimates. Furthermore, the factors that really drive cost, such as institutional and management factors, are not usually included in the variables that drive traditional cost models.

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Some of the initiatives may also require technology that has not been developed yet; hence, there are no historical analogies on which to base estimates. Finally, the programmatic factors, rather than hardware characteristics will probably be the major cost drivers. The cost estimating techniques being developed will be able to deal with all of these considerations.

The first task of understanding cost is to define the specific problems that must be addressed. In the following sections will deal with understanding the environment in which new civilian space programs are developed, understanding cost, and deriving a new cost method.

3.1. UNDERSTANDING THE ENVIRONMENT

In order to begin discussing the problems of estimating cost for new initiative, it is necessary to understand the environment in which new programs are developed. The major factors that need to be considered are the budget, the high cost of programs, and the public perception of the space program. Following the discussion of these factors, the problem faced by the Office of Exploration management will be defined.

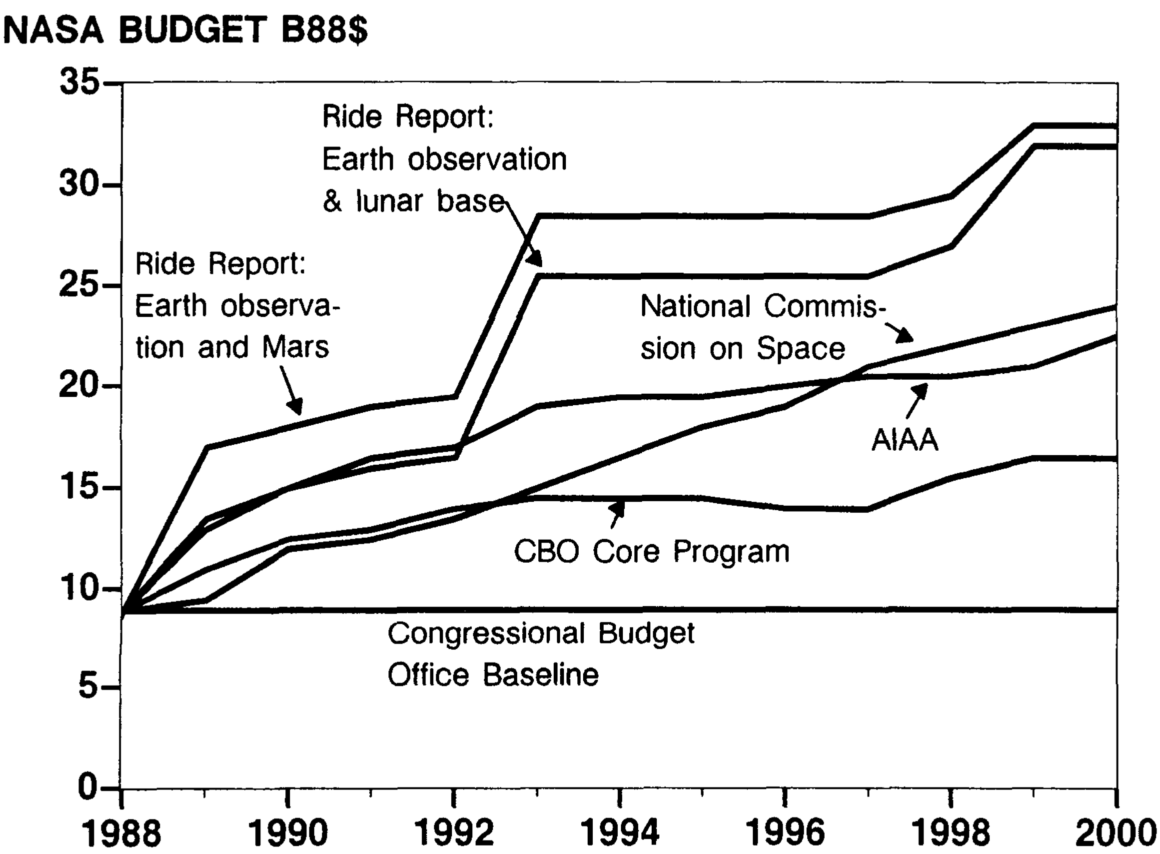
3.1.1. BUDGET CONSIDERATIONS

Since NASA is an agency of the United States Federal Government, it is dependent on the Government for most of its budget. This means that NASA must compete with many other worthy Federal programs for limited financial resources. Each year, the President, Office of Management and Budget, and Congress must decide how much money to allocate to various agencies and programs.

For the past 15 years or so, the NASA budget has been relatively constant at about $8 billion, adjusted for inflation. This is approximately one third of the peak Apollo era budgets. A number of recent studies have tried to estimate the cost of future human initiatives in space in terms of the overall NASA budget. These estimates are summarized in Figure 3.1. The line designated CBO Core Program represents the Congressional Budget Office estimate of the continuing cost of programs that are currently operational, or under development. As shown, the core program alone will require a doubling of the NASA budget (corrected for inflation) by the end of this century.

Some of the more ambitious programs suggested by others will require almost triple or quadruple the present budget. Furthermore, these programs require a sus-

3.1 BUDGET CONSIDERATIONS



tained budget outlay, unlike the temporary spurt of the Saturn/Apollo program.

3.1.2. HIGH COST OF PROGRAMS

NASA manned programs have been very expensive. The Saturn/Apollo program cost was over $96 billion. The cost of the Space Shuttle is $67 billion to date. The planned Space Station program may cost in excess of $20 billion. Recent unmanned programs have also been relatively expensive. The total Viking project cost was $3 billion. Space Telescope cost is $2.2 billion so far, and the Tracking and Data Relay Satellite System (TDRSS) cost is $1.9 billion to date. (note: all costs in FY88$)

3.1.3. PUBLIC PERCEPTION OF THE SPACE PROGRAM

Public confidence in NASA is at an all time low. Only 30% of the general public thinks that funding for NASA should be increased. Most people think NASA is much more expensive than it really is and few people recognize the benefits of the space program. The 'crisis of confidence' that led to the Apollo program is unlikely to be repeated.

3.1.4. THE PROBLEM DEFINED

With no change in the way NASA does business, the budget for manned planetary exploration missions will have to be three or four times greater than the current budget. It is not likely that the American public will support such an expensive program. The choice is: change the way NASA does business or forget about going to the Moon or Mars.

3.2. UNDERSTANDING COST

Life can only be understood backward,
but it must be lived forward.
- Kierkegaard

In order to fully understand cost, it will be important to look at some cost and schedule trends. This will be followed by a brief look at cost drivers. The various factors that influence cost will also be examined in greater detail in Chapters 5 and 6. Since Culture is a principle factor in the new understanding of cost, a section will be devoted to defining culture. Some of the special problems costing new initiatives will be discussed; and, finally, the drawbacks of existing models will be addressed.

3.2.1. COST AND SCHEDULE TRENDS

NASA is realizing diseconomies of scale. Overall, the cost of programs is increasing; and, because of budgetary constraints, programs are becoming longer and longer. This means that, under a fixed budget, with no cultural change, the time between major programs is approaching generations of people. Finally, only a small portion of the budget actually produces spacecraft flight hardware.

3.2.1.1. NASA IS REALIZING DISECONOMIES OF SCALE

NASA cultural norms were set during Apollo and all of the Apollo era NASA installations still exist. NASA has been under a constant-value budget for 15 years which is roughly 65% less than peak Apollo era levels. NASA manpower costs during the same period have been reduced by only about 40 percent. However, spacecraft budgets have been reduced 70-80% from peak Apollo levels.

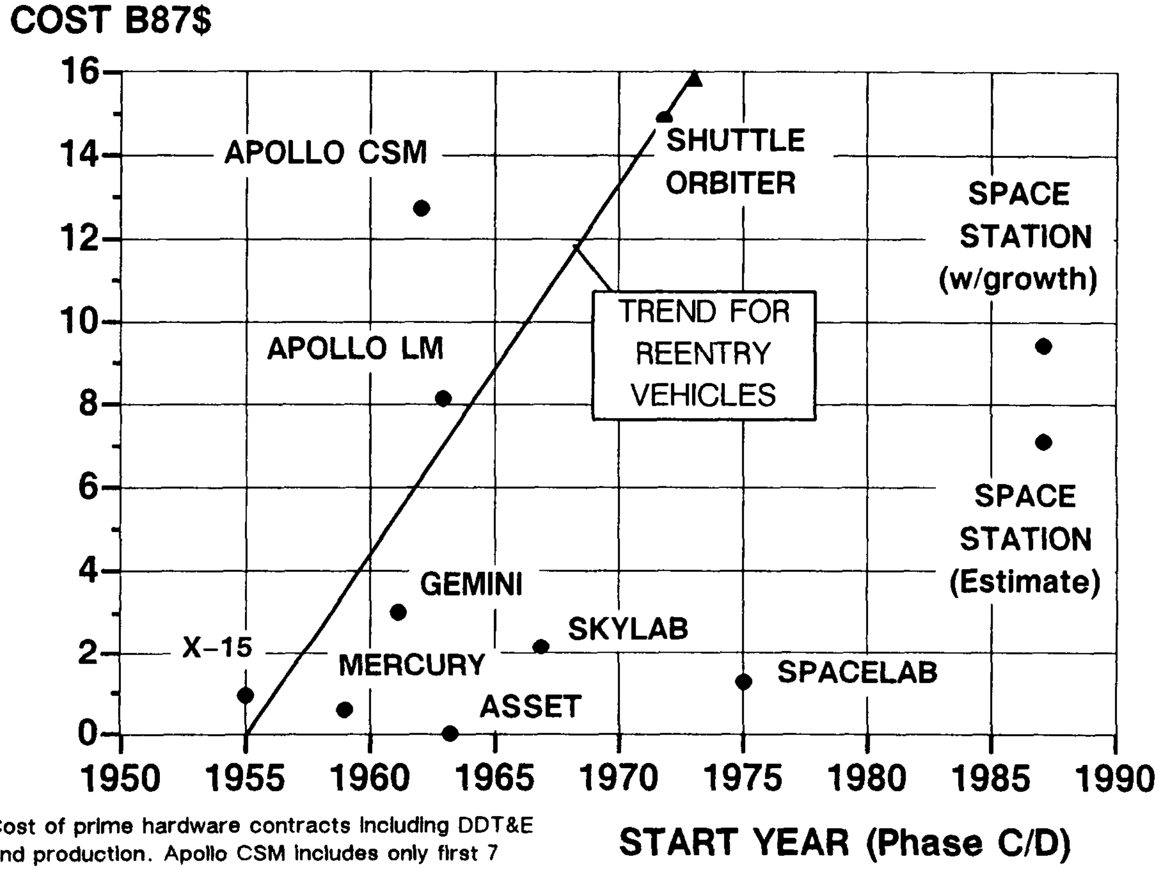
3.2.1.2. PROGRAM COST IS INCREASING

On average, the size of flight hardware is increasing to accommodate ever more complex mission requirements. And, even when normalized for size and quantity, the cost is increasing for most systems. Hardware and software is becoming more complex and performance is increasing. Meanwhile, annual funding is tighter which causes program schedules to become longer and diseconomies of scale in the institutions. These trends are being mitigated somewhat by improvements in productivity; but, the trend is toward higher cost. (see Figure 3.2)

3.2.1.3. PROGRAMS ARE BECOMING LONGER

Because of limited resources and increasing requirements, programs are generally becoming longer. Longer programs exacerbate the problem because of inefficiency and longer programs create the expectation of longer programs, which becomes a self-fulfilling prophecy. Because of sched-

3.2 PROGRAM COST IS INCREASING



Cost of prime hardware contracts including DDT&E and production. Apollo CSM includes only first 7 flight articles. Main engine excluded from Orbiter.

ule stretch out, the time between new starts is increasing dramatically. In addition, the Space Transportation System (STS) and Space Station have very long operations periods, which will further reduce new start wedges. (see Figure 3.3)

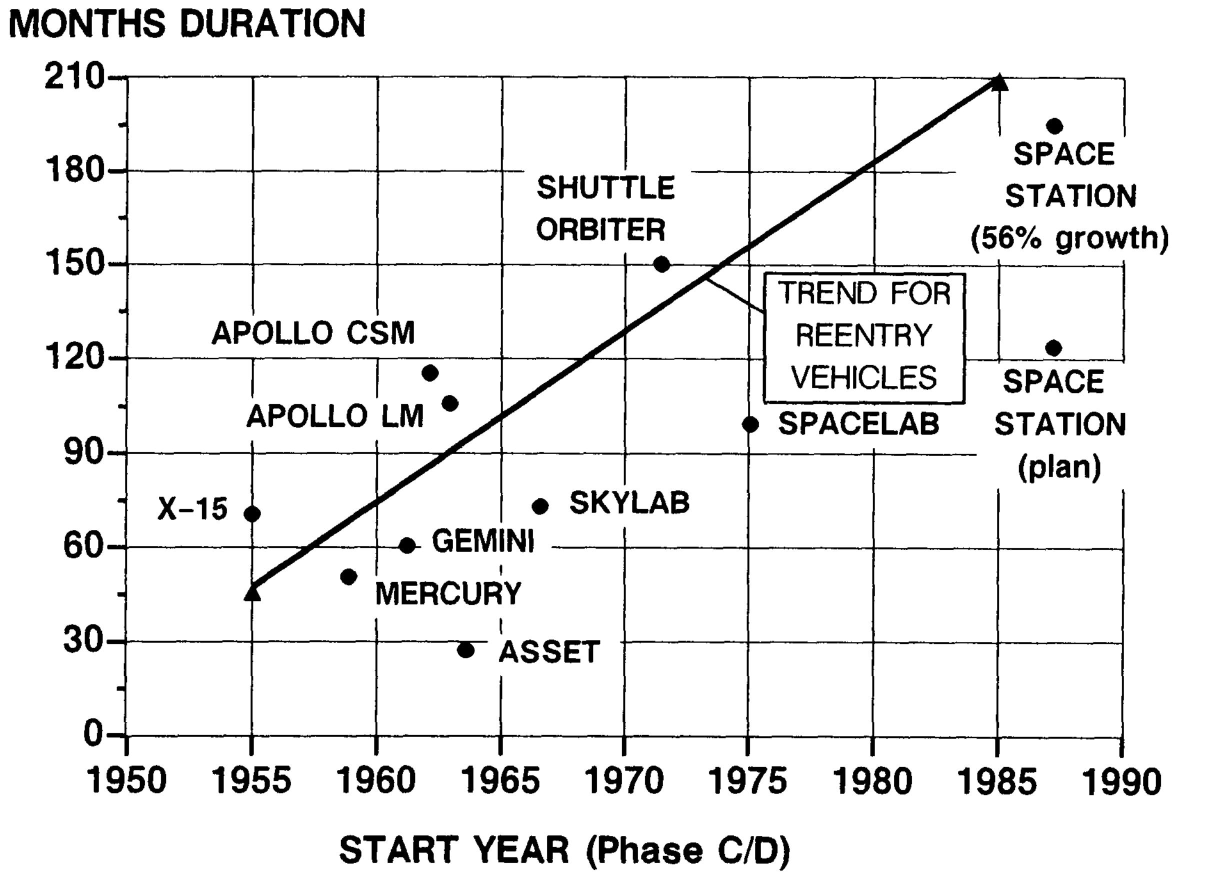
3.2.1.4. THE TIME BETWEEN MAJOR PROGRAMS IS APPROACHING GENERATIONS OF PEOPLE

Since programs have become much longer, and operations are being extended, the amount of time between new starts for major categories of systems (e.g. manned projects) is approaching generations of people. This problem is made worse by poor planning which does not begin to consider new programs until the current projects are complete. The result is that new programs will be run by inexperienced managers. Without change, this could be a path to disaster for NASA. (see Figure 3.4)

3.2.1.5. ONLY A SMALL PORTION OF THE BUDGET PRODUCES SPACECRAFT

Approximately 70% of the NASA Research & Development budget is spent directly on major spacecraft programs. Of this, about 45% is spent on the actual spacecraft and experiments. The rest goes for launch ve-

3.3 PROGRAMS ARE BECOMING LONGER



hicles and operations. The amount of money actually spent on flight hardware (as opposed to development) is even smaller. (see Figure 3.5)

3.2.2. UNDERSTANDING COST DRIVERS

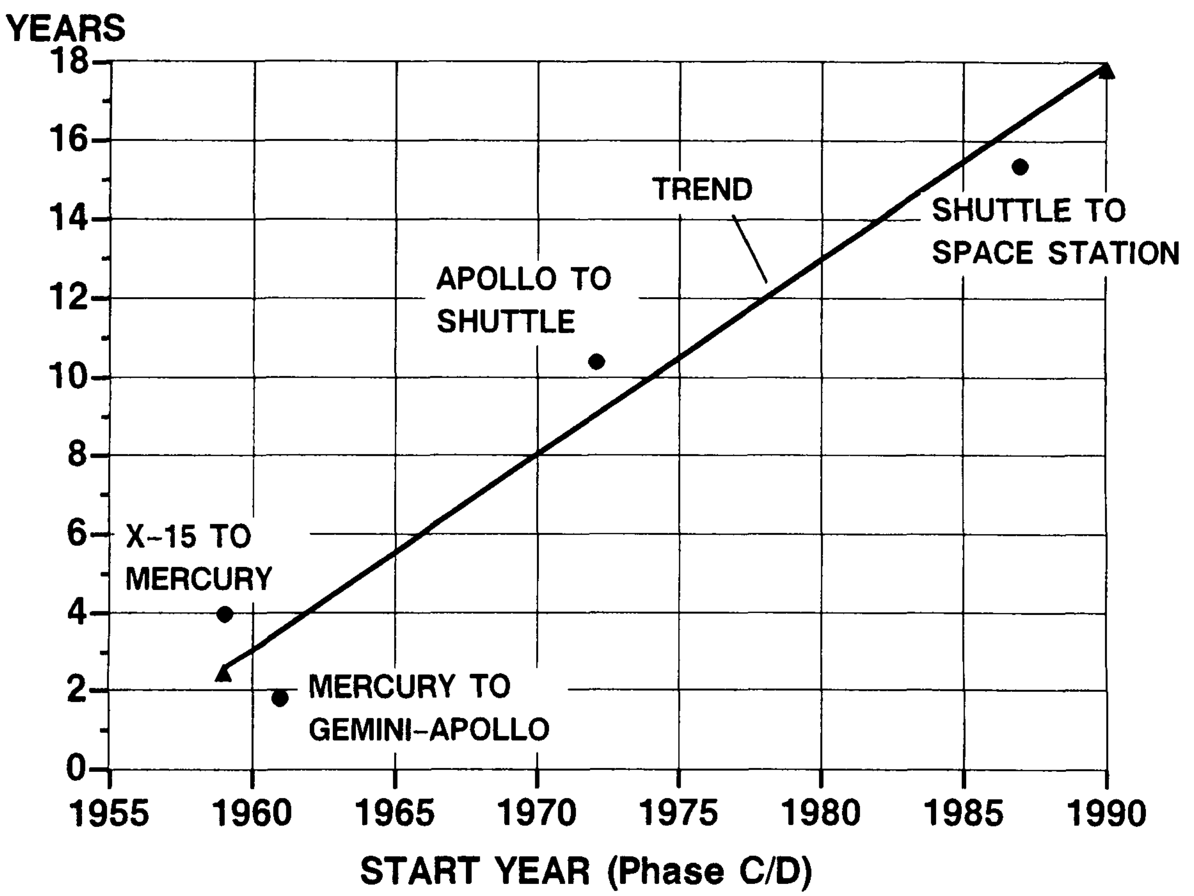
Many factors drive cost including: requirements, specification, quantity, size, performance, level of technology, environment, schedule, budget, team experience, management, safety, degree of risk, productivity, inheritance, et cetera. For conceptual design work, most of the cost drivers can be boiled down to a few key factors: size (e.g. weight), quantity, time (representing technology and productivity), inheritance, culture (or environment) and relative complexity. For any given mission, the design parameters will probably not be very flexible. Therefore, culture is the factor most readily available for change.

3.2.3. DEFINING CULTURE

The fault, dear Brutus, is not in our stars–but in ourselves...
– William Shakespeare

Culture is a condensation of many factors. It establishes the level of specification, thereby defining the conditions of the product's use and the inherent level of required

3.4 TIME BETWEEN PROGRAMS



quality assurance. However, changes in specification level do not explain all of the cost differences between programs.

The residual cost differences have been attributed to the organization's manner of doing business. In other words, organizational work habits do not readily change as a result of imposing different specification levels.

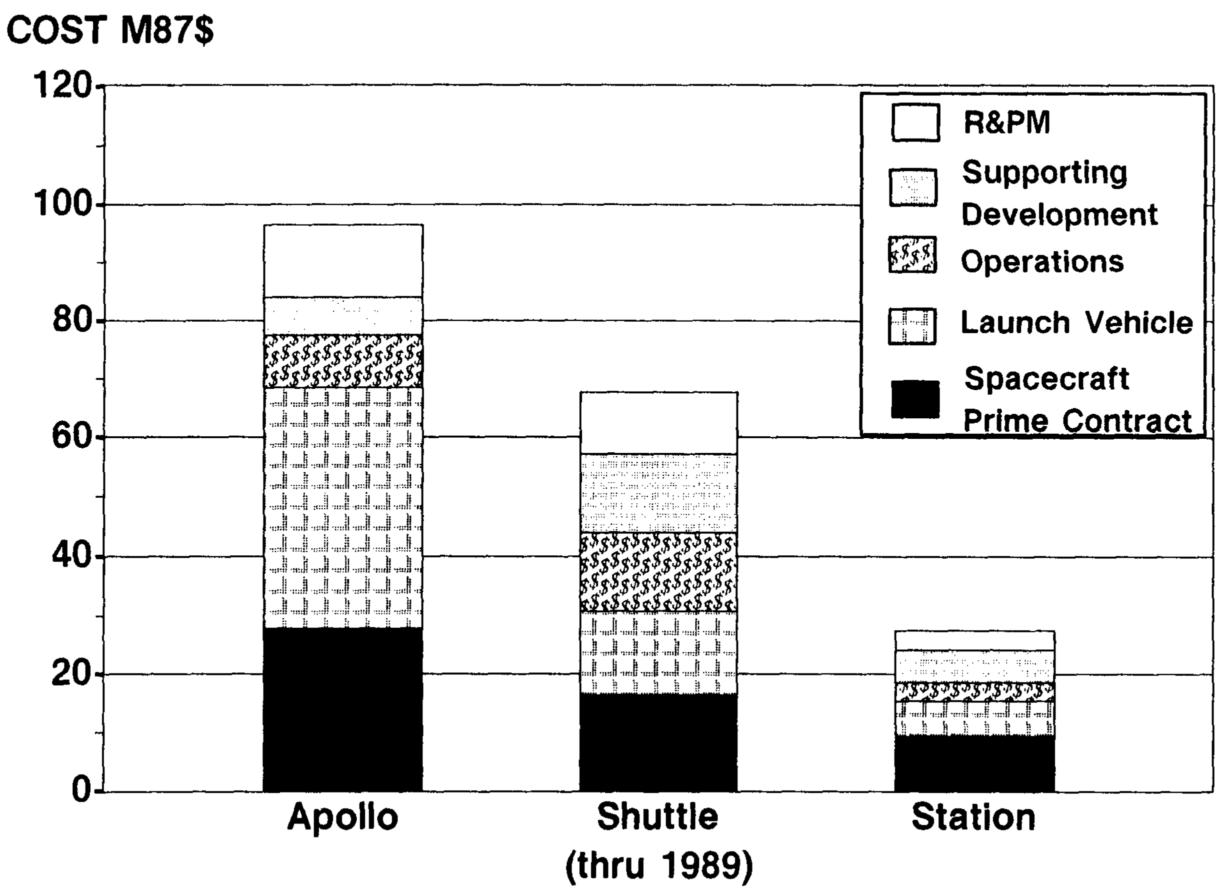
3.2.4. SPECIAL PROBLEMS COSTING NEW INITIATIVES

Current methods give estimates that are too high because they are based on old ways of doing business with no adjustment for change. Forecasting cost far into the future (20-50 years) is very difficult because of the changing environment. Predicting the cost of new technology is difficult because the relationship between cost and other known variables may change. Estimating programmatic cost is difficult because it is not well understood.

3.2.5. DRAWBACKS OF EXISTING MODELS

It ain't so much the things we don't know that get us in trouble. It's the things we know that ain't so.
- Artemus Ward

3.5 SMALL PORTION FOR SPACECRAFT

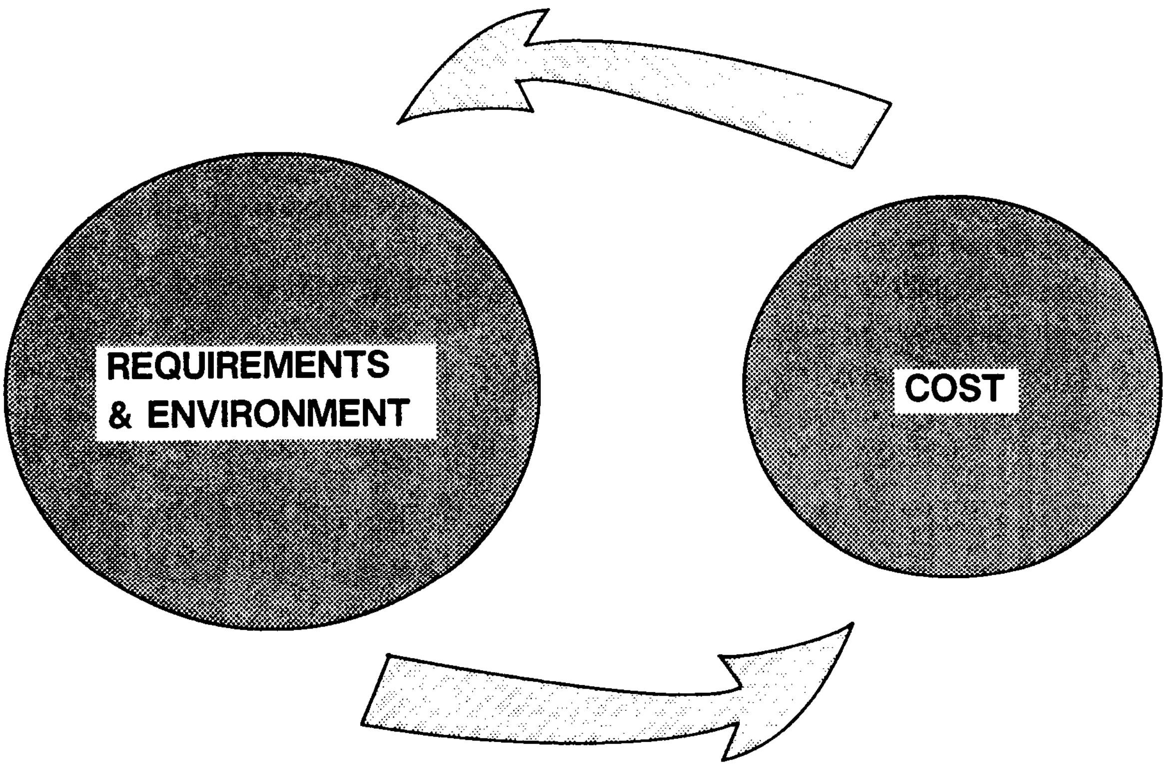


Existing models are dependent on few data points and very few (usually one) explanatory variables. Existing models use historical data from the existing culture, and are therefore predictors of cost only in the same culture. Existing parametric models predict cost as a function of size and a complexity factor. Errors in the subjectively chosen complexity factor will have a far greater influence than errors in objective parameters

3.3. A NEW COST METHOD

The preceding discussion leads to only one conclusion; that it will be necessary to develop a new cost methodology for new civilian space initiatives that involve human exploration of space. A method which defines the conditions under which a certain cost outcome might result. (see Figure 3.6)

3.6 A NEW COST METHOD



4. KNOWLEDGE ACQUISITION

Knowledge is the only instrument of production that is not subject to diminishing returns.
- J. M. Clark

REQUIREMENT: "Task involves collecting experience from agency-wide and industry-wide sources, fitting that experience to the environment of the initiatives."

The knowledge acquisition process consists of two major tasks; acquisition of general problem solving knowledge and acquisition of information specifically related to the problem of cost analysis for new space initiatives. The general problem solving knowledge includes: statistical analysis tools; decision making theory; cost estimating; schedule estimating; cost spreading; project management; systems engineering; forecasting; chart-making; and, other general skills. Fortunately, most of these functions are readily available in various micro-computer products. The major function of knowledge acquisition is the collection of domain specific knowledge. This includes data pertaining to the domain, algorithms for solving specific problems, and heuristic methods.

Collection of cost data is a continuous process. So far, data has been collected on almost every NASA and unclassified DOD spacecraft. In addition, data has been collected on many other types of large scale research and development projects such as aircraft, missiles, ships, and tanks. The raw data is loaded in a computer database and then standardized to a common format. The knowledge of algorithms and heuristics is being acquired by the following: the study technical manager and study leader are cost estimating experts; frequent reviews are held with other cost experts around NASA; members of the study team participate in professional societies, review literature and attend conferences on related subjects.

4.1. NORMALIZING THE DATA BASE

Data bases comprise cost, technical and programmatic data from a wide range of time periods and contractors. However, the data can't be used for parametric analysis without being normalized. In particular, the following specific questions must be answered:

- o Are the costs comparable in currency, year, and elements of cost?
- o Are the products grouped into homogeneous categories?
- o Are programs discriminated as to culture (specification level)?
- o Are programs discriminated as to state-of-the-art?
- o Are consistent sizing and technical parameters used?

- o Are consistent non-recurring/recurring cost splits observed?

Typical steps in the normalization process are listed in Figure 4.1 on page 4-2 . Each of these steps is outlined in the following paragraphs.

4.1.1. NORMALIZING COST DATA

The objective of this effort is to produce total costs that are consistent for purposes of comparison. The first concern is to derive cost numbers that cover the same standard set of elements. In the AMCM data base, Figure 4.2 on page 4-3 shows that reported costs omit fee but include every other element of cost; fee is shown once per program in a 'below the line' display.

To keep consistency in cost units, Figure 4.3 on page 4-4 shows that two steps are taken: 1) no cost is cited without the corresponding year of economics; a standard set of NASA historical escalation tables is used to translate then-year to constant-year base dollars, 2) for foreign currencies, conversion tables are used to generate US dollar figures at convenient economic conditions.

4.1 DATA NORMALIZATION PROCESS

NORMALIZING COST DATA;
1 = Making Elements of Cost Consistent
2 = Making Units of Cost Consistent
3 = Making Year of Economics Consistent

NORMALIZING THE SIZING DATA;
1 = Dry vs Wet Weight
2 = Weight Contingency Application
3 = Percent Electronics

NORMALIZING PRODUCTS BY MISSION APPLICATION;
1 = Grouping Vehicles by Complexity Regime
2 = Calibrating Like Vehicles

NORMALIZING END ITEMS FOR HOMOGENEITY;
1 = Accounting for Absent Cost Items
2 = Removing Inapplicable Cost Items

NORMALIZING RECURRING/NONRECURRING COSTS;
1 = Prime Contractors' Estimates
2 = Time Phased Costs
3 = Flight Article Equivalent Units

NORMALIZING STATE-OF-DEVELOPMENT VARIABLES;
1 = Mission Uniqueness
2 = Product Uniqueness

NORMALIZING CULTURES;
1 = Manned Space
2 = Unmanned Space
3 = Other Cultures

4.2 CONSISTENCY IN THE ELEMENTS OF COST

FACTORY LEVEL	+	SELL LEVEL FACTORS	=	PRIMARY LEVEL OF AMCM REPORTING
DIRECT LABOR OVERHEAD MATERIAL SUBCONTRACT BURDENS		G&A ALLOCATED PRIME COSTS		

FEE IS "BELLOW THE LINE" NUMBER;
REPORTED JUST ONCE IN ANY PROJECT

4.1.2. NORMALIZING THE SIZING DATA

In parametric analysis the sizing variables are significant; for example, it is common to think of cost per pound of hardware or cost per line of software source code. However, these measures are meaningless unless both the size and cost dimensions are consistent. With respect to weight, the AMCM program uses only dry weight of systems; this removes the extraneous influence of propellant and fluids costs, which have low bulk price. The sole exception to this rule is that solid rocket motor sizes are reported as fully loaded weights, because cast solid propellants do have significant costs. Weight contingencies, more common in preliminary or conceptual programs, are always included in the totals so that weight uncertainty is offset.

In the AMCM data base, the variable Electronic Composition Factor is compiled for historical programs because model building has shown this factor to be a significant cost driver. For consistency, this factor is defined in the most simple way as weight of subsystems containing electronics divided by total dry weight. Those subsystems assumed to be all-electronic

4.3 MAKING COST UNITS CONSISTENT

FROM FISCAL YEAR	TO 1987	FROM FISCAL YEAR	TO 1987
1959	5.604	1977	1.984
1960	5.373	1978	1.841
1961	5.206	1979	1.681
1962	5.006	1980	1.519
1963	4.837	1981	1.369
1964	4.629	1982	1.252
1965	4.476	1983	1.180
1966	4.223	1984	1.114
1967	4.026	1985	1.071
1968	3.819	1986	1.038
1969	3.614	1987	1.000
1970	3.380	— Exchange Rates —	
1971	3.180		
1972	3.008		
1973	2.846		
1974	2.655		
1975	2.396		
1976	2.190		
TO	2.153		

include Command and Data Handling (C&DH), and Guidance, Navigation and Control (GN&C). Although electrical power subsystems do contain some electronics, they are not included with the latter.

Consistency in software sizing is maintained by reporting the total machine executable instructions (deliverable) as the primary dimension. Numbers of comment statements or other non-deliverable software may be reported as 'below the line' parameters.

4.1.3. NORMALIZING PRODUCTS

In parametric analysis it is critically important to analyze homogeneous products. This homogeneity must exist across product groupings and within individual products. The AMCM approach to finding homogeneous product groupings is to classify missions by application and objective. The standard mission category identification code system is shown in Figure 4.4 on page 4-5 . In this system, the first digit of the ID code separates manned and unmanned missions; this digit also dis-

4.4 MISSION CATEGORY ID CODES

x x x

First Digit: Source
1 = Historical, Manned
2 = Historical, Unmanned
3 = In Development, Manned
4 = In Development, Unmanned
5 = Conceptual, Manned
6 = Conceptual, Unmanned

Second Digit: Operating Environment
1 = Space, Low-Earth Orbit
2 = Space, High-Earth Orbit/Geostationary
3 = Space, Lunar
4 = Space, Planetary/Interplanetary
5 = Surface, Stationary
6 = Surface, Mobile
7 = Marine
8 = Atmospheric
9 = Extra Atmospheric

Third Digit: Mission Objective
1 = Test-Bed Missions
2 = Scientific Explorers
3 = Scientific Observatories
4 = Communications/Navigation
5 = Earth Surveillance
6 = Exploration
7 = Habitation
8 = Exploitation
9 = Transportation

criminates completed missions from developmental and conceptual programs. The second digit separates products for the space environment from those for other environments, and also discriminates the flight regimes within space flight. The third digit classifies missions by end objective; these classifications are fairly straightforward except that the grouping Test-Bed Missions was added for the category of flight programs whose purpose was to demonstrate the feasibility of new applications.

Homogeneity within individual products is maintained by the Work Breakdown Structure (WBS), which controls the end-item dimension of cost. The AMCM standard WBS is shown in Figure 4.5 on page 4-6 . The WBS is a checklist for products and services normally found in a space vehicle. For instance, displaying Hubble Space Telescope costs against this WBS shows no appreciable costs for Power Source (WBS 01.01.01.02.01); this is because the solar array for this program was a Government-furnished item, and its costs were paid for by ESA. In this case the WBS is warning us not to compare HST costs - especially not electrical power system costs - against programs with complete subsystem sets. In the same way the WBS

4.5 STANDARD WORK BREAKDOWN STRUCTURE

- 01 Space Segment
  - 01 Spacecraft Platform
    - 01 Bus Hardware
      - 01 Structures/Mechanisms/Thermal
        - 01 Structures
        - 02 Mechanical
        - 03 Thermal
      - 02 Electrical Power
        - 01 Power Source
        - 02 Power Conditioning
        - 03 Power Distribution & Control
      - 03 Attitude Control
        - 01 Attitude Determination
        - 02 Attitude Control
      - 04 Propulsion
        - 01 Main Propulsion (Less Engines)
        - 02 Secondary Propulsion
        - 03 Reaction Control Propulsion
        - 04 Common
        - 05 Instrumentation
      - 05 Communications & Data Handling
        - 01 Tracking
        - 02 Telemetry
        - 03 Command
        - 04 Data Management
        - 05 Instrumentation
      - 06 ECLSS/Crew Accommodations (If Manned)
      - 07 Booster Adapter
    - 02 Non-Bus Hardware
    - 10 Non-Bus Software
  - 02 Common Services
    - 01 Management & Support
      - 01 Program Management
      - 02 Financial/Schedule Control
      - 03 Configuration Management
      - 04 Data Management
      - 05 Manufacturing Plans
    - 02 Systems Engineering
      - 01 Requirements Definition & Allocation
      - 02 System Analysis
      - 03 Reliability, Maintainability, Quality
      - 04 Other "Llities"
      - 05 Test & Ops Planning
    - 03 Integration & Test
      - 01 Integration Management Plans
      - 02 Mission Support
      - 03 On-Orbit Servicing Operations
    - 04 GSE
    - 05 Support
      - 01 Other Direct Cost
    - 06 Facilities
  - 03 Operations Support
    - 01 Launch Support
    - 02 Mission Support
    - 03 On-Orbit Servicing Operations
  - 04 Segment-Level Integration & Test
    - 01 Integration Hardware
      - 01 Fairing
      - 02 ASE
  - 51 Payload

will force us to remove inapplicable hardware. Consider the problem of upper-stage hardware. Some classes of space vehicles - Agena derivatives and Comsats for example - carry the upper stage hardware for the ascent vehicle. The WBS deals with this problem by creating a separate block of numbers (WBS 01.01.02) for non-spacecraft-bus hardware.

4.1.4. NORMALIZING NON-RECURRING/RECURRING COST BREAKDOWN

Past practice in parametric analysis has been to model separately the non-recurring and the recurring costs. While it is not necessarily true that future models will require such breakdowns, it is nonetheless prudent to maintain non-recurring and recurring costs in consistent format for current use.

The preferred method for non-recurring/recurring separation is detailed analysis at lowest levels of cost collection, preferably by the prime contractor. This provides the best insight into design, manufacturing and support cost transitions. For consistency, Government guidelines assist the categorization of costs as DDT&E or Production phase; such guidelines were available during the NASA Cost Analysis Task Force era of historical studies, for example.

Lacking detailed assessments, there are two alternative methods for assigning non-recurring and recurring costs. Figure 4.6 on page 4-8 shows a time-phasing method. In this method the stream of costs is referenced to some agreed-on schedule milestone, commonly Critical Design Review, and all costs before the milestone are considered non-recurring and afterward, recurring. To some extent, this method is followed even for detailed cost analysis.

The second method of recurring/non-recurring cost separation is shown in Figure 4.7 on page 4-9 . This method performs a pro rata of total acquisition costs using an equivalent units calculation. In this method every prototype article is assigned an equivalent quantity of flight units, and an equivalent is also estimated for design and development activity (taking into account difficulty, inheritance, etc.). The ratio of DDT&E to Production units then determines the ratio of non-recurring to recurring costs.

4.1.5. NORMALIZING THE STATE OF DEVELOPMENT

Data bases tend to contain programs that reflect wide ranges of development effort - from simple follow-on buys to complex, multi-path developments. Each level of development history invokes a different non-recurring cost, so it is vital to normalize data with a variable that characterizes the wide range of possible development environments. The selected state-of-the-art variable is shown in Figure 4.8 on page 4-10. This variable allows the user to characterize the set of development conditions that would have existed at the time a given program began.

Two attributes are considered in this evaluation. The first is design status, that is, familiarity with materials, parts and

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4.6 TIME PHASING METHOD

ACCT PERIOD	MATERIAL DOLLARS ACT/EST	MATERIAL DOLLARS BUDGET	SUBCONTRACT DOLLARS ACT/EST	SUBCONTRACT DOLLARS BUDGET	TOTAL BURDEN DOLLARS ACT/EST	TOTAL BURDEN DOLLARS BUDGET	OTHER DIRECT COST DOLLAR ACT/EST	OTHER DIRECT COST DOLLAR BUDGET	GEN& AD-MIN DOLLARS ACT/EST	GEN& AD-MIN DOLLARS BUDGET	PROGRAMLESS MGT TOTAL DOLLARS ACT/EST	PROGRAMLESS MGT TOTAL DOLLARS BUDGET
DEC 76 A							235	235	36630	36630	26858(	292065
JAN 77 A									3509	3509	25841	27522
FEB 77 A									3959	3959	29217	31048
MAR 77 A									3979	3204	29367	25143
APR 77 A									6001	4309	44295	33830
MAY 77 A									4218	3813	30904	29930
JUN 77 A									3267	3883	23958	30476
JUL 77 A									2866	4220	20991	36163
AUG 77 A									2644	3531	20079	32004
SEP 77 A								3000	1434	3465	10425	34577
OCT 77 A		2000				628	59	3000	4780	4039	38949	39291
NOV 77		3000				942	4000	4000	2605	2605	28080	28830
DEC 77		3000				1041	4000	4000	2731	2731	28574	28574
JAN 78		3000				1041	4000	4000	2643	2643	28399	28399
FEB 78							4000	4000	2286	2286	22039	22028
MAR 78	3000	3000			1074	1074	4000	4000	2351	2351	22631	22631
APR 78	2000	2000			760	760	2710	2710	2682	2682	23837	23837
MAY 78	2948	2948			729	729			2138	2138	16769	16769
JUN 78									1430	1430	11203	11203
JUL 78					116	116			1671	1671	13223	13223
AUG 78					116	116			928	928	7340	7340
SEP 78					79	79			888	888	7026	7026
OCT 78									1123	1123	8920	8920
NOV 78									661	661	6249	5249
DEC 78									751	751	5966	5966
JAN 79									945	945	7481	7481
FEB 79									712	712	5640	5640
MAR 79									734	734	5814	5814
APR 79									572	572	4531	4581
MAY 79									502	502	3975	3975
JUN 79									528	528	4180	4180
JUL 79									247	247	1973	1973
PTD A							294	6235	73287	263649	543106	612049
TO GO							22710	22710	29138	29138	261649	263649
QTR4 76 A							235	235	36630	36630	268580	292065
QTR1 77									11447	10672	84425	83713
QTR2 77									13486	12005	99157	94236
		11000				3652						
	7948	7948			2874	2874						

processes. The second is familiarity with the mission, i.e. whether an analogous program has ever been undertaken. Based on evaluation of these attributes, each program in the data base is assigned a numerical value from the scale.

4.1.6. NORMALIZING THE PROGRAM CULTURE

In any extensive data base, the programs included may cover a variety of developmental cultures. In this sense the word 'culture' is synonymous with 'program practices' or 'specification levels'. It connotes the levels of management, reports and controls that the customer (usually the Government) habitually requires. Normalization of cultures across programs is controlled by assigning values of a single AMCM variable to each program in the data base.

In the AMCM data base the specification-level (culture) variable is a non-linear factor shown in Figure 4.9 on page 4-11. At present each environment corresponds to

4.7 EQUIVALENT-UNITS METHOD

ACQUISITION COSTS

Project	Midpoint of Expenditures for Inflation Adjustment	Actual or Current Estimate Total Cost	Inflation Adjusted	GSFC In-House Prime Effort	JPL Project Monitoring Staff	Adjusted Project SCP Costs
1	1965	141.4	177.0			177.0
2	1965	45.0	56.5			56.5
3	1965	14.1	17.0	+16.3		33.3
4	1967	7.7	9.0	+5.1		[illegible]
5	1966	9.9	10.9	+9.8		20.7
6	1968	4.7	5.3	+7.7		13.0
7	1970	15.7	15.7			15.7
8	1967	84.7a/	99.0			99.0
9	1965	89.6	112.3			112.3
10	1967	74.5	88.0			88.0
11	1969	13.1	16.2			16.2
12	1969b/	24.0c/	25.7			25.7
13	1969b/	30.8c/	33.0			33.0
14	1969	65.0	70.0			70.0
15	1970d/	89.0	59.0		-9.4	79.6
16	1970d/	26.6	26.6		-3.1	23.5
17	1970d/	117.5	117.5		-13.5	104.0

EQUIVALENT-UNIT CALCULATION

Category	OGO A-C	OGO D-F	OSO A-C	OSO D-G	Explorer (IMP) A-C	Explorer (IMP) D&E	Explorer (IMP) F&G	RAE A	SAS A&B	SSS A	ATS
Nonrecurring:											
Initial design & development a/	2.0	0.5d/	0.4	0.7d/	1.0d/	0.4d/	0.4c/	1.5	0.6	1.5	2.0a/
Thermal/mechanical unit	.1				.1	.1	.1	.1	.1	.1	.2
Engineer unit	.5										.7
Other test units					.1	.1	.1				.2
Prototype units	1.7	.2		.2	1.4	1.4					1.5
Protoflight units			1.4				1.7	1.8	2.5c/	1.8	
Total Initial NR	4.3	.7	1.8	.9	2.6	2.0	2.3	3.4	3.2	3.4	4.6
Follow-on Flight redesign											
Prototype											1.3
Redesign total	1.4	.9	.6	.9	.2	.1	.2		.3		1.2a/
Redesign average bet. flights	(.7)	(.4)	(.3)	(.3)	(.1)	(.1)	(.2)		(.3)		(.3)
Total Nonrecurring	4.7	1.6	2.4	1.3	2.8	2.1	2.5	3.4	3.5	3.4	7.1
Recurring											
Flight Units	3.0	3.0	3.0	4.0	3.0	2.0	1.0				5.0
Spares	.5	.5	.3		.4	.4	.5	.6	.1	.5	.3
Total	9.2	5.1	5.7	5.8	6.2	4.5	4.0	4.0	3.6	3.9	12.4
	14.3		11.5								

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4.8 STANDARDIZING THE STATE-OF-THE-ART

RANK	DESIGN NEWNESS	MISSION UNIQUENESS
1.	OFF THE SHELF; DRAWINGS EXIST	EXISTING
2.	MINOR MOD; DRAWINGS RENUMBERED	"
3.	MODERATE REVISION; EXISTING PRODUCT LINE	"
4.	EXTENSIVE REVISION; EXISTING PRODUCT LINE	"
5.	COMPLEX REVISION; EXISTING PRODUCT LINE	"
6.	NEW; EXISTING PRODUCT LINE	FAMILIAR
7.	NEW; FAMILIAR MATERIALS/COMPONENTS/ PROCESSES	"
8.	NEW; NEW MATERIALS/COMPONENTS/ PROCESSES	"
9.	NEW; NEW MCP PLUS DIFFICULT INTEGRATION	NO ANALOGY
10.	ADVANCED SOTA; CRITICAL FUNCTIONS DEMONSTRATED	"
11.	ADVANCED SOTA; MULTIPLE DESIGN PATHS	"
12.	ADVANCED SOTA; PRINCIPLES OBSERVED	"

a step in the scale; as data are refined, this scale may expand significantly.

4.2. THE NORMALIZED DATA BASE

The outcome of the normalization process just described is to produce a data base of consistent and directly comparable information. The AMCM data base is maintained in hard-copy and computerized format. The hard-copy data is stored in its original format, which encompasses a diverse range from formal post-program studies to informal analyst notes. The computerized data, drawn from the hard-copy sources, exists in three separate forms:

1) Original data in contractor's format; 2) Detailed data extracted from item 1) and organized by the standard WBS; 3) Summary level data also extracted from item 1)

The organization of the data in items 2) and 3) is by program and by WBS. A supplemental breakdown by 'product categories' is available for item 3) only; this categorization of products and services is generic so as to permit cross-cultural comparison of the cost effects of specifications and standards.

4.9 CONSISTENCY IN SPECIFICATION (CULTURE)

SPECIFICATION VARIABLE	CONTENT
1.00	GROUND BASED EQUIPMENT
1.15	SHIPBORNE OR GROUND MOBILE
1.30	AIRBORNE, MIL-SPEC
1.67	SPACEBORNE, UNMANNED
1.80	SPACEBORNE, MANNED

Costs for all programs in the computerized data base are expressed in original currencies and original economic conditions. In this way all costs can be escalated or converted at a single calculation across the data base.

4.2.1. PROGRAM COVERAGE OF THE DATA BASE

Programs that now make up the AMCM data base are listed in Figure 4.10 on page 4-12. At present only space programs are in the collection; they are manned and unmanned. However, the new data base strategy will mandate expansions in product coverage. To encompass programs with alternative cultures (i.e. different from space) the data base will add projects in aircraft, helicopters, missiles and ground vehicles. Also, the space system data set will be enhanced with new and more ambitious missions in order to analyze the effects of performance growth on cost. The planned new data acquisitions are listed in Figure 4.11 on page 4-13.

4.2.2. CONTENTS OF THE SUMMARY DATA BASE

The variables included in the AMCM summary data base are shown in Figure 4.12

4.10 PROGRAMS IN THE DATA BASE

PROGRAM	PROGRAM IDENTIFICATION	PROGRAM	PROGRAM IDENTIFICATION
X-15	1	GEMINI AGENA TARGET VEH.	67
Mercury	2	AE	68
Gemini	3	TIROS M	70
Apollo CSM	4	TIROS N	71
Lunar Module	5	LANDSAT	72
Lunar Rover	6	SMS	73
Orbiter	6	DMSP 5-1	74
Skylab	8	SURVEYOR	81
Spacelab	9	LUNAR ORBITER	82
Space Station	10	MARINER MARS 64	85
Heao A/B	50	MARINER VENUS	86
HST	51	MARINER MARS 69	87
SEASAT	54	MARINER MARS 71	88
ATS A-E	58	MARINER VENUS MERCURY 73	89
ATS F-G	60	VIKING ORBITER	90
OAO-B	61	VIKING LANDER	91
OGO	62	PIONEER F/G	92
OSO A-G	63	HELIOS A/B	100
OSO-I	64		
STP 71-2	65		

on page 4-14. There are two data screens per program. Screen one summarizes cost, weight, complexity, and quantity data; screen two presents schedule data and also makes any needed adjustments in cost and weight (for example, as a result of Government-furnished equipment). Note that the costs and quantities cover not only Development and Production in their traditional meanings, but also the more recent trend of programs with protoflight articles; these latter may be thought of as some Development (prototype testing) and some Production (flight use).

The costs on screen one include separate breakouts for the bus and payload, for common services and for operations. The equivalent standard-WBS nomenclatures for these entries are as follows:

Bus	= WBS 01.01
Common Services	= WBS 01.02
Operations	= WBS 01.03
Payload	= WBS 01.51

The entry for the item labeled 'Spec' represents the program culture variable, as was discussed earlier.

4.11 PROGRAMS BEING ADDED

NAME	ID NO	NAME	ID NO.
SATURN	20	ESRO–I	112
SATURN S–II	21	ESRO–II	113
SATURN S–IVB	22	ESRO–IV	114
SATURN S–IB	23	COS B	115
SATURN IU	24	TD	116
SHUTTLE ET	25	DIAL	117
SHUTTLE SRB	26	AZUR	118
CENTAUR	30	AEROS	119
F–1	40	SYMPHONIE	120
J–2	41	UK–3	121
RL–10	42	UK–4	122
H–1	43	MX	200
SSME	44	PATRIOT	201
VOYAGER	93	C–5A	300
GALILEO	94	C–141	301
MAGELLAN	95	B–70	302
HEOS–1	110	M–1 ABRAMS	400
GEOS–2	111	M–2/M–3 BRADLEY	401

The adjusted weights and costs shown on screen two are of special importance. These adjustments are the way in which incomplete or non-homogeneous cost data are normalized.

4.2.3. MAKING USE OF THE DATA BASE

The object of building and normalizing the AMCM data base is to be able to use the information so assembled to derive parametric relationships. Using the methodology (particularly the calibration features) of AMCM, the first conclusion that becomes apparent through analysis is that space programs are not homogeneous; not even all manned, or all unmanned programs, have common complexity patterns. Instead, space programs appear to fall into smaller groups whose common thread is similar mission objective.

Figure 4.13, Figure 4.14, and Figure 4.15 on page 4–15, page 4–16 and on page 4–17 illustrate how programs group by mission type. These statistical sets show the correlation of the spacecraft complexity/ performance variable to year of first flight. Notice that good correlation is consistently obtained (measured by the value r–squared) for the small mission groups.

4.12 CONTENTS OF SUMMARY DATA BASE

```
                              PROGRAM OVERVIEW - SCREEN 1
IDENTIFIERS
PROGRAM NAME: APOLLO CSM      PROGRAM ID: 4        MSN TYPE ID: 136
QUANTITIES
TEST UNITS=20.00  PROTOFLIGHT=  PRODUCTION. SERIAL UNITS=7          SPARES =
(EQUIV.)                        PRODUCTION. UNIQUE UNITS=           (EQUIV.)

TOP-LEVEL BREAKDOWN   COST IN MISSIONS/FRACTION OF TOTAL            WEIGHT
           DEV.COST   PROTOFLT.COST   REC.COST  TOTAL COST          POUND
BUS        1460.89                    581.64   2042.53  0.64        31231.00
COMMON      865.69                    276.19   1141.88  0.36
SVCS.
PAYLOAD                                           0.00  0.00         1055.00
OPERATIONS                                        0.00  0.00
------------------------------------------------------------------------------
TOTAL      2326.58          0.00      857.83   3184.41              32286.00
COST FEE
PRICE                                          3184.41
AS OF DATE                                     JAN-66
COMPLEXITY:          SPEC    1.80
   MCPLX             E/S     0.10 STATE-0F-ART VARIABLE    6.00 PCT.BUS ELEX.  0.08
                              OVERVIEW - SCREEN 2
SCHEDULES:
AUTH TO PROCEED    =JAN-62     FIRST FLIGHT          =OCT-68  FULL OPNL.CAP.=
CRITICAL DES.REV.  =           INITIAL OPNL.CAP.     =
1ST UNIT DELIV.    =           LAST FLIGHT           =DEC-72
FRACTION OF END ITEMS SUPPLIED GFE (BY WEIGHT):
              TOTAL WEIGHT   GFE ST   FRAT.              TOTAL WT.  GFE  WT.   FRACT.
BUS =         31231.00                0.00   PAYLOADS =  1055.00               0.00
GFE ITEMS: BUS=                              PAYLOADS =
ADJUSTMENTS TO BUS END ITEM WEIGHT/COST:
                WEIGHT           NR COST    P'FLT COST  REC.COST          TOTAL COST
INCREMENTAL   -11676.00          -64.90                  -29.80
CUMULATIVE     19555.00          2261.68       0/00      828.03           3089.71
EXPLANATION OF ADJUSTMENTS: LAUNCH ESCAPE SYSTEM AND SLA ADAPTER
COMMENTS ON DATA SOURCES:
```

Another trend that can be observed in these figures is the mainly positive slope of the complexity versus time plots. This signifies an ever-increasing value of spacecraft complexity - and by extension performance capability - over time.

The next step in the data base analysis will be to select and quantify performance figures of merit. The object will be to track the observed complexity growth against performance growth, and to quantify this mathematical relationship.

4.13 GROUPING LIKE PROGRAMS

SEVEN PLANETARY MISSIONS

RESULTS OF LEAST SQUARES FIT REGRESSION

R SQUARED =	0.833
SLOPE =	0.109
INTERCEPT =	0.751

PROGRAM NAME	MISSION TYPE	IOC FIRST FLT	COMPLEXITY FACTOR
MARINER MARS 64	246	. NOV-64	7.819
MARINER VENUS	246	JUN-67	8.271
MARINER MARS 69	246	FEB-69	8.136
MARINER MARS 71	246	MAY-71	8.376
MARINER VENUS MERCURY 73	246	NOV-73	8.691
VIKING ORBITER	246	AUG-75	8.909
VIKING LANDER	246	AUG-75	9.302

MARINER SERIES ONLY

RESULTS OF LEAST SQUARES FIT REGRESSION

R SQUARED =	0.857
SLOPE =	0.085
INTERCEPT =	2.425

PROGRAM NAME	MISSION TYPE	IOC FIRST FLT	COMPLEXITY FACTOR
MARINER MARS 64	246	NOV-64	7.819
MARINER VENUS 67	246	JUN-67	8.271
MARINER MARS 69	246	FEB-69	8.136
MARINER MARS 71	246	MAY-71	8.376
MARINER VENUS MERCURY 73	246	NOV-73	8.691

4.14 GROUPING LIKE PROGRAMS

LARGE OBSERVATORIES

RESULTS OF LEAST FIT REGRESSION

R SQUARED =	0.804
SLOPE =	0.104
INTERCEPT =	0.651

PROGRAM NAME	MISSION TYPE	IOC/ FIRST FLT	COMPLEXITY FACTOR
HST	213	Y89	10.072
OAO-B	213	NOV-70	8.252
HEAO A/B	213	AUG-77	8.064

SMALL OBSERVATORIES

RESULTS OF LEAST SQUARES FIT REGRESSION

R SQUARED =	0.567
SLOPE =	-0.038
INTERCEPT =	10.807

PROGRAM NAME	MISSION TYPE	IOC/ FIRST FLT	COMPLEXITY FACTOR
OGO	212	SEP-64	8.620
OSO A-G	213	MAR-62	8.244
OSO-I	213	JUN-75	7.924

4.15 GROUPING LIKE PROGRAMS

SYNCHRONOUS AND HIGH ORBIT SATELLITES

RESULTS OF LEAST SQUARES FIT REGRESSION

R SQUARED =	0.608
SLOPE =	0.097
INTERCEPT =	1.343

PROGRAM NAME	MISSION TYPE	IOC/ FIRST FLT	COMPLEXITY FACTOR
ATS A-E	224	DEC-66	7.737
ATS F-G	224	MAY-74	8.153
SMS	225	MAY-74	8.871

WEATHER SATELLITES (LEO)

RESULTS OF LEAST SQUARES FIT REGRESSION

R SQUARE =	0.924
SLOPE =	0.036
INTERCEPT =	4.509

PROGRAM NAME	MISSION TYPE	IOC/ FIRST FLT	COMPLEXITY FACTOR
TIROS M	215	JAN-70	7.070
TIROS N	215	OCT-78	7.386
DMSP 5-1	215	SEP-76	7.229

5. CONCEPTUALIZATION

When you can measure what you are speaking about, and express it in numbers, you know something about it
- Lord Kelvin

REQUIREMENT: "Key product, in addition to the costing techniques analysis, is a tailored method ('cookbook') that can be used by the human initiatives program agents for costing that would include the programmatic and specific agency assumptions on environment in which the initiatives will be developed."

5.1. COST MODEL APPROACH

Cost estimates for new programs are required early in the planning process so that decisions can be made accurately. Because of the long lead times required to develop space hardware, the cost estimates are frequently required 10-15 years before the program delivers hardware. The system design in conceptual phases of a program is usually only vaguely defined and the technology used is often state of the art or beyond. These factors combine to make cost estimating for conceptual programs very challenging.

This section describes an effort to develop a parametric cost estimating method for space systems in the conceptual design phase. The approach is to identify variables that drive cost such as weight, quantity, development culture, design inheritance and time. The nature of the relationships between the driver variables and cost will be discussed. A theoretical model of cost will be developed and tested statistically against a historical data base of major research and development projects. Figure 5.1 on page 5-2 shows the difference between the new approach to cost modeling and previous cost models.

5.2. COST THEORY

5.2.1. MODEL REQUIREMENTS

In order to meet the needs of NASA for a long-range cost forecasting tool, the cost model:

-must have the ability to predict cost over long time horizons (25-50 years)

- must be valid for substantially different types of systems

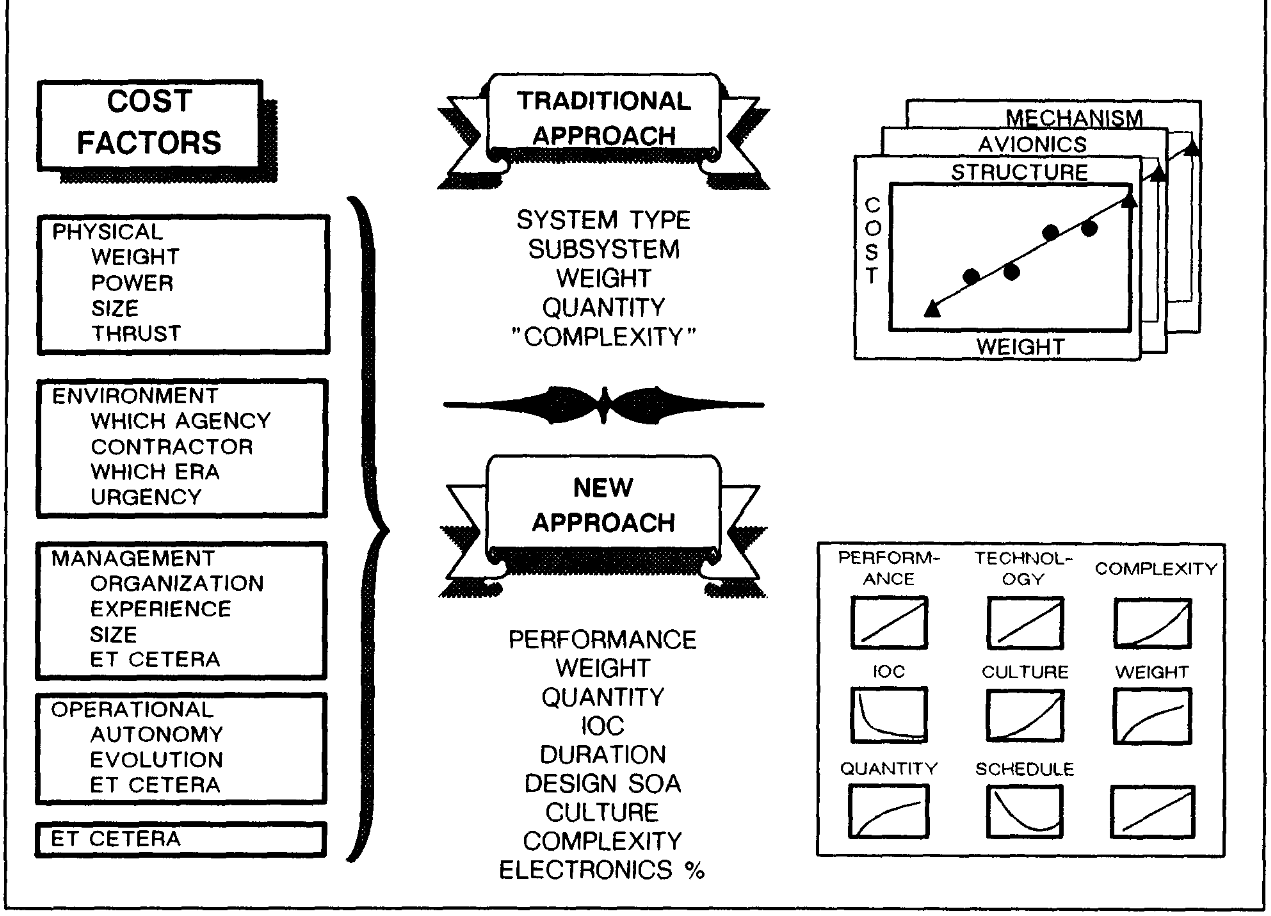
- must be able to predict cost reliably despite significant technological advances

- must be simple to use, requiring few inputs.

5.2.2. MODEL IDENTIFICATION

In order to determine the feasibility of a model that would meet the specified requirements, a proof of concept test was devised. A theoretical model was developed for predicting the total acquisition cost of major hardware development programs. The variables identified for use in the model are described below.

5.1 COST MODEL APPROACH



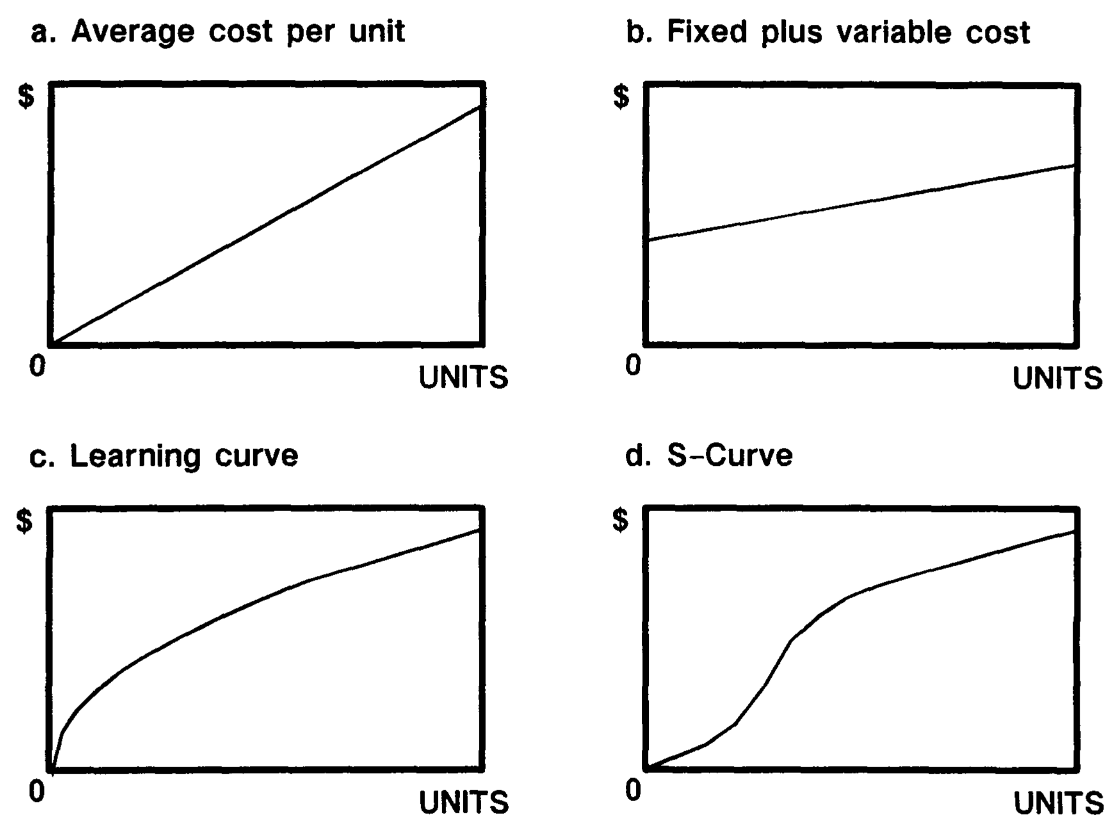
5.2.3. QUANTITY VARIABLE

The relationship between the number of units produced and cost can take many forms. In Figure 5.2 on page 5-3, four of the most common forms are illustrated. Figure 5.2a illustrates the unit or average cost method in which the average cost per unit is used. In this case, the average cost is the same regardless of the quantity produced. This method is most useful for small quantity buys of commercial products where the quantity purchased does not materially affect the cost of production.

A second method of estimating cost, illustrated in Figure 5.2b, is the fixed plus variable cost method. The marginal cost, in this case, is constant. The average cost is higher than the marginal cost, decreases as the quantity increases and approaches, but never reaches, the marginal cost. In this case, the fixed cost is relatively large and changing the quantity produced can substantially affect the average cost. This model represents economy of scale.

The third method, illustrated in Figure 5.2c, incorporates the principle of decreasing marginal cost. In other words, the addi-

5.2 TOTAL COST VERSUS QUANTITY



tional cost of each unit is slightly less than the previous unit. This principle is also known as the learning curve or experience curve. The learning curve also has decreasing average unit cost as the quantity is increased.

A fourth type of quantity relationship is shown in Figure 5.2d. In this case, the marginal cost increases for the first several units, then begins to decrease along the lines of a learning curve as quantity increases further. This example would represent a situation where the first few units were partially operational or low cost prototypes gradually building up to full scale production articles. Once a reproducible configuration is reached, the marginal cost decreases according to learning curve principles.

5.2.4. WEIGHT VARIABLE

Weight has been used for many years in estimating the cost of aerospace systems. It is a most convenient variable since it generally characterizes the size and often performance of a piece of hardware. Weight is also a key engineering parameter; therefore, an estimate of it is usually available, even at the early stages of a program. Although the emphasis here is on weight, the discussion could also be applied to other descriptive parameters such as size, speed, power, et cetera.

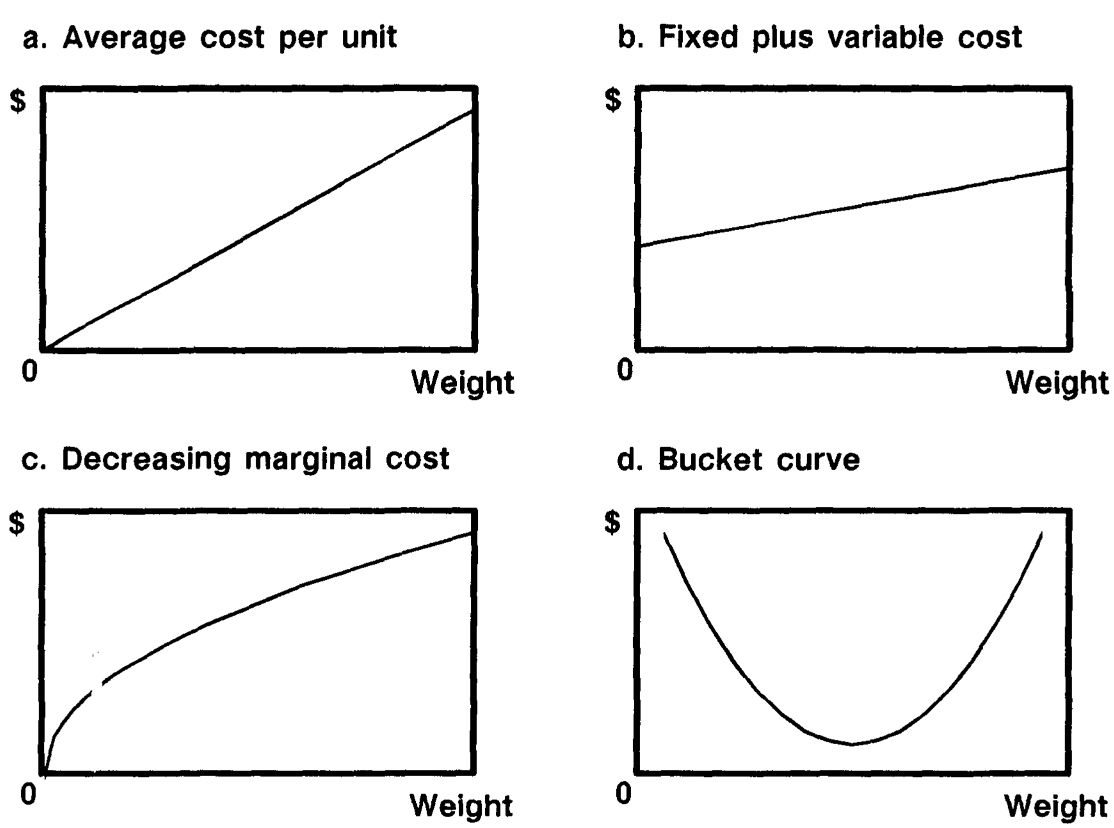
The following discussion will refer to weight as the dry mass of a single unit. Like quantity, weight can be related to cost in several ways. The most common relationships are depicted if Figure 5.3 on page 5-4. In Figure 5.3a, the simple cost per unit weight relationship is illustrated. By definition, the cost per unit weight model has constant average cost per unit weight.

The model in Figure 5.3b has the characteristic fixed plus variable cost. In this case, the average cost per unit weight decreases as the weight increases. The marginal cost is constant and average cost is asymptotic to marginal cost. This is a case of economies of scale with respect to unit weight.

Figure 5.3c illustrates a model in which the marginal cost is decreasing; hence, the average cost is decreasing. In this case, the rate of change in the marginal cost is also decreasing.

The total cost relationship shown in Figure 5.3c is an exponential growth function. The exponent happens to have a special meaning in economics. It is the elasticity of cost with respect to weight. If the elasticity

5.3 TOTAL COST VERSUS UNIT WEIGHT



is greater than 1, then the relationship is said to have decreasing economies of scale. If the elasticity is greater than 0 but less than 1, then there are increasing economies of scale. If the elasticity is exactly 1, then there are constant economies of scale.

Clearly, if there are strong economies of scale, it would be better to build larger (heavier) things. It should be noted, however, that weight and quantity may also be related. The larger something is, the less likely it is to be built in large quantities. The relationship between cost and quantity may also have economies of scale; therefore, the effect of different weights on both cost and quantity should be considered when estimating total program cost.

In the last case, Figure 5.3d, the marginal cost of weight is negative up to a certain weight, then becomes positive. The total cost curve becomes U shaped (also known as a bucket curve). The bucket curve represents a situation where there is an optimum weight for a given type of hardware. Any attempt to decrease the weight below optimum would incur additional cost through the use of exotic materials, additional manufacturing processes, or more complex fabrication techniques. By the same token, attempts to increase the weight above optimum would require additional cost for high performance propulsion, additional structural analysis and testing, specialized tooling, et cetera.

5.2.5. CULTURE VARIABLE

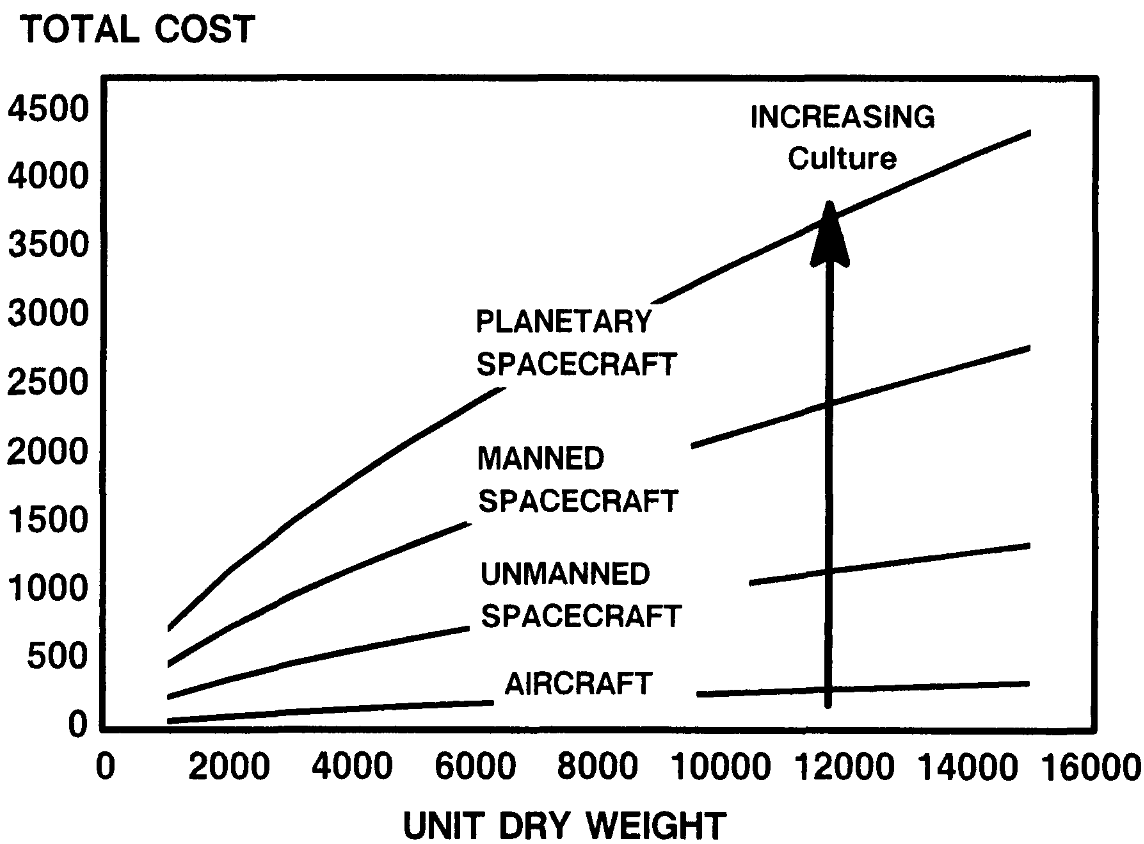
So far, it has been postulated that significant relationships exist among cost, quantity, and weight. It is not likely, however, that the relationships are exactly the same for all different types of hardware. A situation, such as the one in Figure 5.4 on page 5-6, may exist where the cost versus weight curves for several types of hardware have the same elasticity but different multipliers. The Culture variable is defined as a value representing the vertical height of the cost/weight curve for a given subcategory of hardware. If the cost/weight curves were plotted on a log-log graph, the lines would be parallel straight lines and the Culture variable would be a function of the vertical height of each line.

A category is defined as a group of hardware systems that are functionally similar; such as, aircraft, ship, or spacecraft. A subcategory describes a group of systems that perform a similar mission or have the same operational environment. The subcategories of aircraft would include fighter, bomber, transport, et cetera. The classifications used in this chapter are listed in Figure 5.5 on page 5-7.

It must be assumed, for the convenience of regression analysis, that the elasticities are the same for all subcategories. This will prove to be an overly restrictive assumption and future work may focus on techniques to eliminate the need to make it.

The effect of Culture on cost is depicted in Figure 5.6 on page 5-8. The chart shows

5.4 CULTURE VARIABLE



that the impact on cost becomes very dramatic when the value of Culture increases above two. Since most space types of hardware are above a Culture of two, the cost could be influenced substantially by small changes in Culture.

5.2.6. COMPLEXITY VARIABLE

Within a given subcategory, it is possible that the systems may vary considerably in terms of performance, capacity, level of technology, complexity of design, and many other factors. Variations of the type listed within a given subcategory are henceforth referred to by the variable name Complexity. Complexity is obviously very difficult to define and quantify *a priori*.

The potential for overlap between Culture and Complexity can also create confusion. Research and Development organizations tend to group along functional and mission lines, hence the classification scheme used for Culture inherently contains organizational information as well. Organizational differences within a given subcategory may be included in Complexity. Also, specification levels vary along the functional lines in Culture, so only the specification differences within an established

5.5 CULTURE CLASSIFICATION SCHEME

SUBCATEGORY	NO.	Culture
SPACECRAFT	**56**	**2.18**
PLANETARY	13	2.45
MANNED REENTRY	5	2.34
COMMUNICATION	9	2.22
PHYSICS & ASTRONOMY	12	2.2
WEATHER	7	2.19
MANNED ORBITAL	2	2.05
EARTH OBSERVATION	6	2.04
UNMANNED REENTRY	7	2.04
MISC. SPACE	2	1.95
MISSILES	**87**	**1.89**
SURF-SURF, OTHER	4	2.07
AIR-AIR	13	2.04
AIR-ORBIT	1	2.04
SURFACE-AIR	12	1.97
ICBM	11	1.92
ICBM (SUB)	4	1.89
SURF-SURF, LAND	8	1.88
AIR-SURFACE	15	1.81
ANTI-TANK	4	1.78
SHIP-AIR	9	1.74
ROCKETS	6	1.64

SUBCATEGORY	NO.	Culture
AIRCRAFT	**63**	**1.82**
BOMBERS	8	1.99
ATTACK	8	1.96
FIGHTERS	16	1.94
PATROL	5	1.88
ROTARY ATTACK	5	1.88
ROTARY CARGO	5	1.75
COMMERCIAL	3	1.74
FW-TRANSPORTS	10	1.63
TRAINER	3	1.46
SHIPS	**29**	**1.14**
DESTROYERS	5	1.25
SUBMARINES	7	1.24
CRUISERS	4	1.19
FRIGATES	3	1.14
A/C CARRIERS	5	1.11
AMPHIB. ASSAULT	5	0.89
GROUND MOBILE	**16**	**1.15**
RIFLES	3	1.59
TANKS	4	1.24
APC'S	2	0.96
TRUCKS	7	0.82

subcategory should be considered in Complexity.

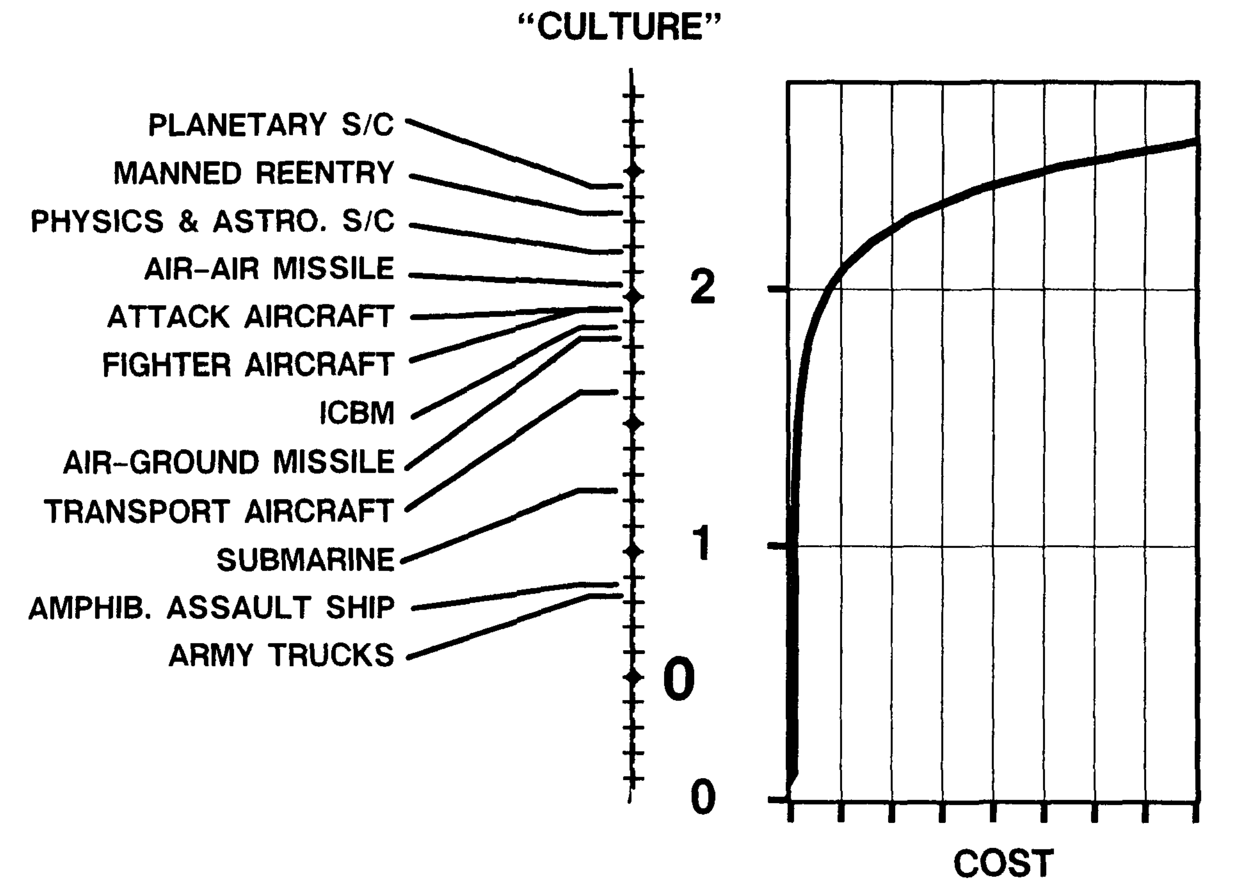
Since there is no readily available means of quantifying Complexity *a priori*, this variable will not be used in the subsequent model derivation. It is discussed here in order to clarify the definition of Culture and to provide a basis for future work to refine quantitative measures of Complexity.

5.2.7. TIME VARIABLE

Another factor that must be considered in estimating cost is the impact of time related phenomenon. Inflation, productivity, technology and performance are just a few of the factors that may change with time. For most cost estimating applications, the effects of inflation are removed by applying standard inflation rates to convert the data to a constant year dollars. The modeling of productivity, performance and technology change is not so easy.

Time related phenomena may change at a fixed rate, like interest on a bond, or they may vary from one time period to another. The method of using a program milestone date as the time variable will result in a fixed rate of change when the model is estimated. Measurement of the variable rate case would require construction of an in-

5.6 EFFECT OF CULTURE ON COST



dex, similar to an inflation index, and then selecting the appropriate index value based on the year of Initial Operational Capability (IOC), mid-point of construction or some other basis. A productivity or technology improvement index could be incorporated in this fashion. For this report, the IOC Year was chosen to represent time.

5.2.8. GENERATION VARIABLE

The design of a new aircraft, spacecraft or missile is often based on a previous design that has already been proven. In the case of aircraft, the new airplane may use the previous airframe with only minor structural modifications. Spacecraft designs may use structural components, electronics, and mechanical systems already tested on a previous design. Designers may work with configurations they are familiar with from previous projects. The result may be considerable savings in the development cost of new hardware. Savings can also be achieved in production since the tooling already exists and manufacturing experience is far down the learning curve from the previous design.

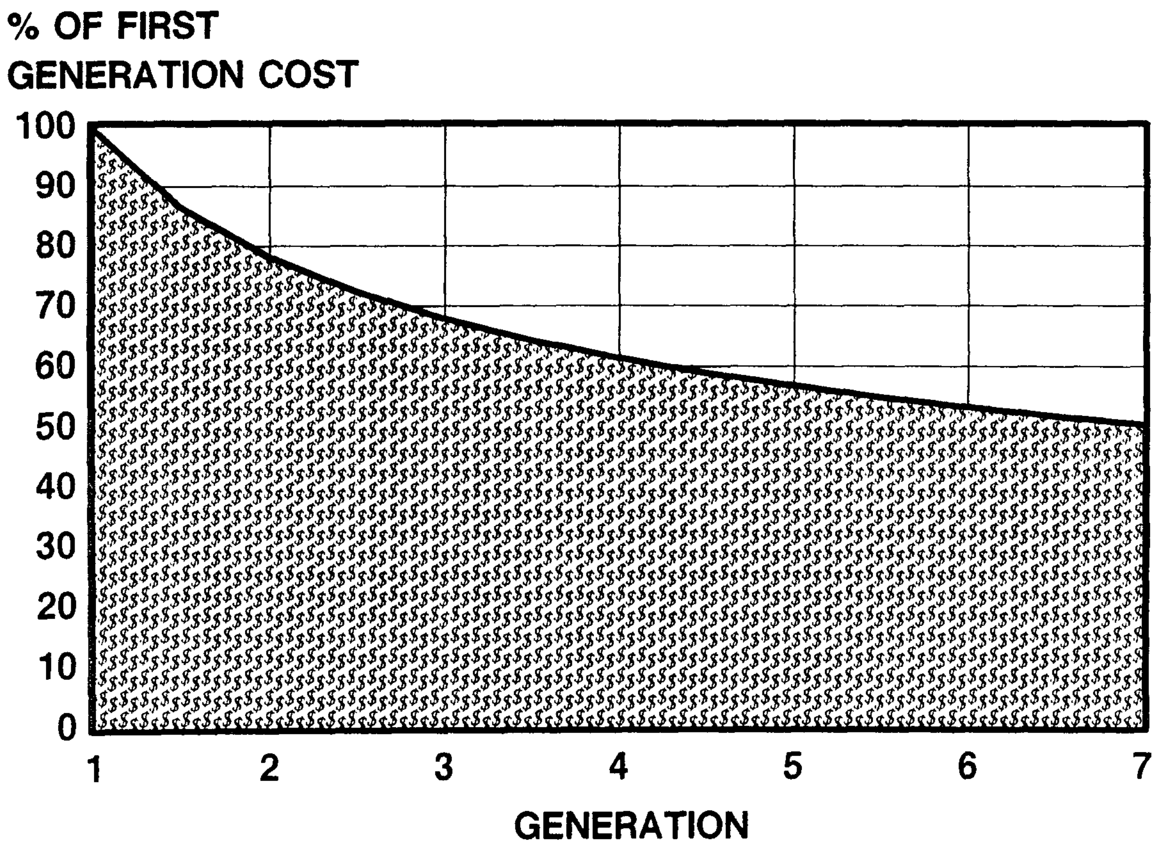
In theory, the cost of each subsequent model should be considerably less than the previous model. The amount of savings, however, would probably decrease as the series progresses. The total cost would be decreasing asymptotically to some level as shown in Figure 5.7 on page 5-9.

The Generation variable used in this report is defined as the sequential number for a given model of a specific piece of hardware. Generation is not used to represent individual units of production, but rather a group of identical units. Subsequent Generations must have very similar characteristics, usually being produced by the same manufacturer or to the same specifications. Individual units of production may be given a Generation number if they differ substantially from previous units but still retain the basic design and total production is small. All programs that do not have readily identifiable predecessors are given a Generation of 1 (one).

5.3. STATISTICAL ANALYSIS

This section will describe the statistical analysis that was conducted to create a model incorporating the variables described above. The intent was to attempt to prove a hypothetical model by regression analysis of historical program cost data. The following sections will describe

5.7 GENERATION VARIABLE



the historical database, the model evaluation process and the validation process

5.3.1. DATA BASE DESCRIPTION

In order to statistically validate some of the theories relating to cost behavior, it was necessary to construct a data base of cost and other variables for many different types of research and development programs. The data base consists of 264 major programs. Most of the programs are U.S. Government sponsored. Many of the Government programs are Defense related weapons and delivery systems. A substantial number of NASA sponsored spacecraft are also included. A small proportion of the data comes from other Government agencies, foreign countries and commercial companies. In total, the data base represents $1 trillion worth of expenditures in 1987 dollars. Programs from the 1930's all the way up to the mid 1980's are included. Major categories include ground vehicles, ships, aircraft, missiles and spacecraft. Data collected for this study includes top level cost data, system weights, program schedule dates, developing organizations, and technical data. A variety of sources were used to gather data and information was confirmed by two or more sources whenever possible.

5.3.2. MODEL EVALUATION

Model evaluation has consisted of three major steps. The first step was to test a model consisting of the variables Quantity, Weight, Culture, IOC Year and Generation against the data base as a whole. Step 2 required the estimation of models for individual subcategories of data. Finally, the elasticities derived from step 2 were compared to the Culture variable derived in step 1.

Step 1 had several major functions. One was to evaluate the theoretical model of Quantity, Weight, Culture, IOC Year and Generation. A second function was to produce estimated values of Culture for different program subcategories. A third purpose was to identify any data observations that may be incorrect or classified wrongly. The final function was to develop estimates for the elasticities of weight and quantity, as well as other presumed constants.

Using total program cost, weight, quantity, and other data, a multiple linear regression analysis was performed. The results are presented in Figure 5.8 on page 5-11. Out of 264 data points, 253 observations were included in the regression model. The remaining observations were rejected due to missing data. The dependent variable is the $\log_{10}$ of total acquisition cost. The independent variables are $\log_{10}$ Weight, $\log_{10}$ Total Quantity, Culture, IOC year and $\log_{10}$ Generation. The coefficient of determination (R squared) is 0.91 and all of the variables are significant according to their t-statistics. Also, the signs and the magnitude of the coefficients are reasonable.

As discussed earlier, the Culture variable is a derived value. The derivation begins by entering an estimated value for each Culture subcategory. The multiple regression is performed using the original value for Culture. The estimation errors for each

5.8 REGRESSION MODEL RESULTS

Dependent Variable: Log_{10} of Total Acquisition Cost

Independent Variables:

		COEF. VALUE	T-STAT	STD ERROR
Constant		-4.7645		
Log Q	Log_{10} Total Quantity	0.5773	47.5	0.0122
Log W	Log_{10} Unit Dry Weight (lbs.)	0.6569	43.5	0.0151
C	Culture	1.7705	31.8	0.0556
Y	IOC Year - 1900	0.0124	9.3	0.0013
Log G	Log_{10} Generation	-0.3485	-7.5	0.0466

Standard Error of Y Estimate	0.2247
R Squared	0.9125
Observations	253
Degrees of Freedom	247
MAPE	45%

$$\textbf{COST} = 0.0000172\, Q^{0.5773}\, W^{0.6569}\, 58.95^{C}\, 1.0291^{Y}\, G^{-0.3485}$$

subcategory are then averaged. The original Culture value for each subcategory is then adjusted by a factor calculated to make the average error for that subcategory equal to zero. A new multiple regression is then performed with the adjusted Culture values. This process is repeated until the regression statistics stabilize. In order to minimize rounding errors, the Culture values are rounded at the second decimal place prior to the regression analysis.

A second regression analysis was done at the subcategory level for a few selected subcategories. This process generally used $\log_{10}$ Total Acquisition Cost as the dependent variable and $\log_{10}$ Weight and $\log_{10}$ Quantity as independent variables. In some cases, IOC Year and Generation were also included. The results of step two are summarized in Figure 5.9 on page 5-12. Note that the R-squared values are good for almost all subcategories. The elasticity of weight and elasticity of quantity are displayed along with the estimated Culture values.

The final step in the analysis was to compare the Culture values to the elasticity values with respect to weight. Recall that Culture is a function of the intercept of the

5.9 SUBCATEGORY MODEL RESULTS

CAT	SUBCATEGORY	CULTURE	WEIGHT ELASTIC.	QUANTITY ELASTIC.	R^2
S/C	Planetary	2.45	0.45	1.02	0.87
S/C	Physics & Astronomy	2.20	0.68	1.17	0.95
MSL	Air-air	2.04	0.69	0.53	0.86
MSL	ICBM	1.92	0.81	0.92	0.93
A/C	Attack	1.96	0.43	0.52	0.92
A/C	Fighter	1.94	0.74	0.46	0.95
MSL	Air-surface	1.81	0.91	0.57	0.81
A/C	Transport	1.63	0.91	0.54	0.87
SHIP	Submarine	1.24	1.18	0.92	0.99
SHIP	Amphib. Assault	0.89	1.30	0.30	0.95

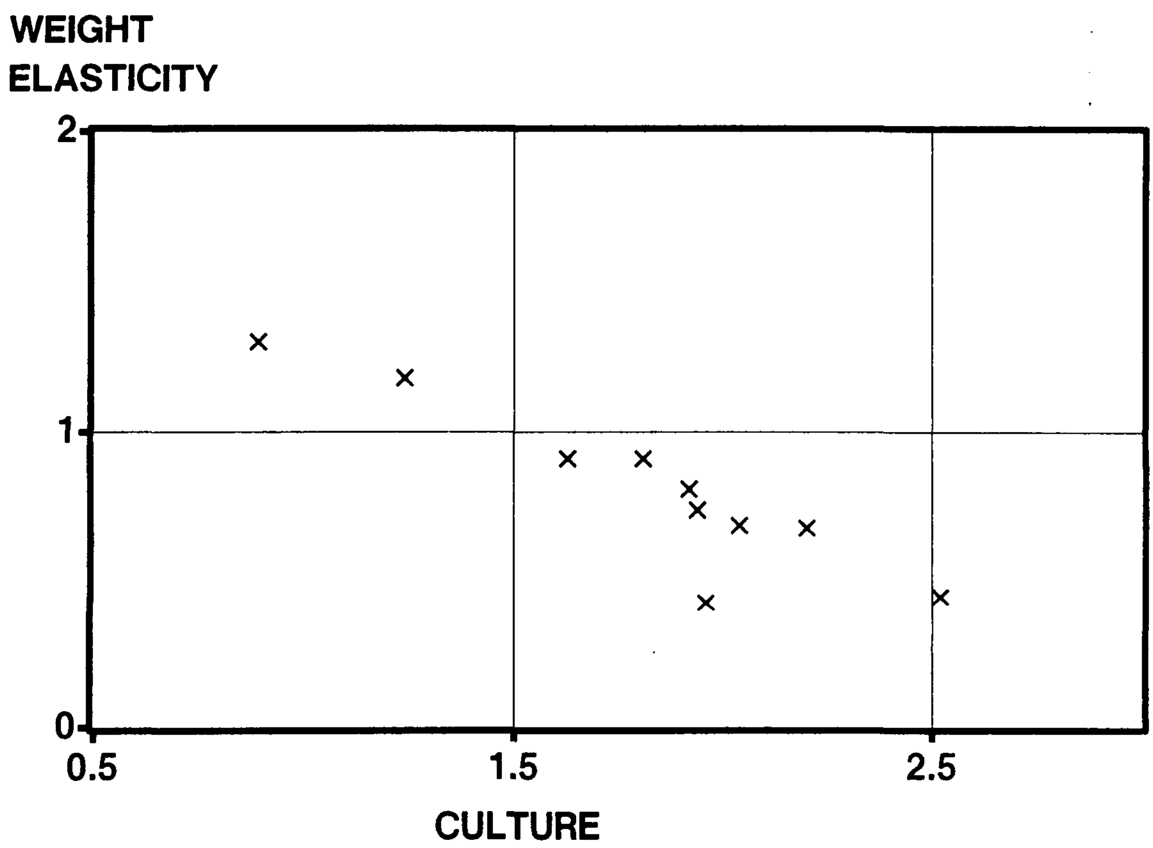
regression lines. and elasticity is the slope of the regression lines in a log-log model. A regression analysis of the dependent variable Weight Elasticity and the independent variable Culture found high correlation with an R-squared of 0.80, or 0.95 with the one outlier removed (see Figure 5.10 on page 5-13).

Furthermore, The coefficient of Culture has a negative sign. This can be interpreted economically as meaning that high Culture programs have greater economy of scale with respect to weight than low Culture programs. Figure 5.11 on page 5-14 illustrates the effect of the latter conclusion on the cost/weight curves. Note that moving down or to the right increases the slope.

It is also noteworthy that two subcategories, submarines and amphibious assault ships actually had weight elasticities greater than one, indicating diseconomies of scale.

An attempt was also made to correlate Culture with Quantity Elasticity but the results were inconclusive. Of particular interest are the quantity elasticities of planetary and physics and astronomy satellites which are 1.02 and 1.17 respectively. The fact that these elasticities are close to or greater than one indicates that the mar-

5.10 WEIGHT ELASTICITY VERSUS CULTURE



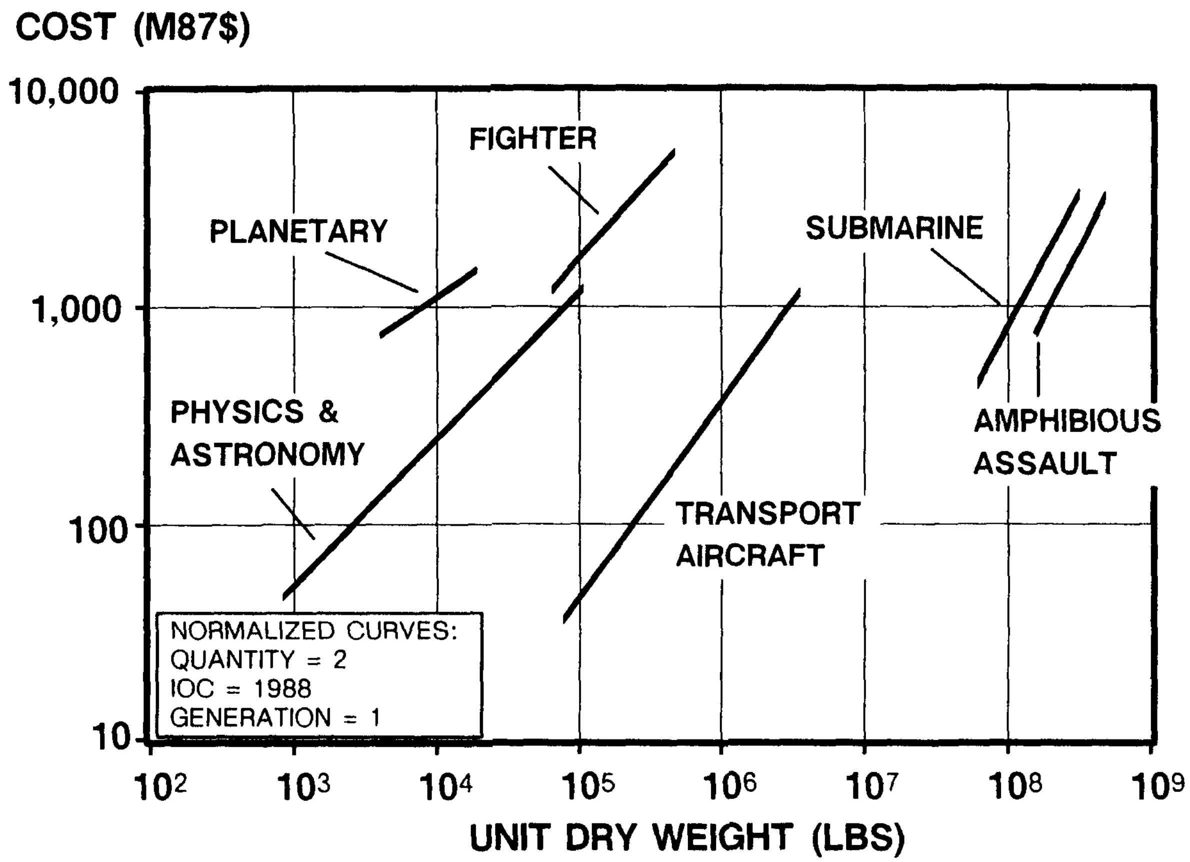
ginal cost is constant or increasing. Since spacecraft generally have very small production runs, and the first few units are generally prototypes or test articles, this is not surprising. The high elasticities may be indicative of the S-curve depicted in Figure 5.2d on page 5-3.

5.3.3. MODEL VALIDATION

A procedure was developed for validating the statistically estimated model. The validation procedure consisted of dividing the data base into two parts. The data was divided at the median IOC Year, 1969. All programs prior to 1970 were used to calibrate a new model using the same variables as the overall model. Values for Culture were also calibrated based on the limited data.

The restricted model was then used to simulate a forecast of the actual programs in the second half of the data base. The result was that the simulated forecast overestimated the total actual cost by approximately 45%. This indicates a bias in the estimating model. An examination of the coefficients showed that all were reasonably consistent between time periods except for the coefficient of IOC Year. The coefficient for IOC Year is 50% higher in

5.11 SUBCATEGORY MODELS



the first period than overall. This difference probably accounts for most of the overestimate.

Several explanations may be offered for the variation in IOC Year coefficients. Different inflation indices were used to normalize the data during different time periods. The indices used may not have been appropriate.

The IOC Year variable used for time also assumes a constant rate of change over the entire time period. It is possible that whatever factor the IOC Year variable is attempting to measure was, itself, changing over time. Productivity changes in the work force are one possible explanation.

Due to the magnitude of the error caused by the IOC Year coefficient, it will be essential to identify the source of error before this model can be used for forecasting. Future work will focus on isolating the problem and developing solutions.

5.4. CONCLUSIONS

In order to accurately make any forecast using mathematical or statistical modeling, several conditions must be meet. First, the structure of the model; i.e., the nature of the relationships must be identified. Second, the parameters of the equation that are expected to vary, as input or outputs, need to be specified. Third, those factors that remain constant must be identified and estimated. Finally, the conditions under which the structural equations and parameters remain stable must be specified and tested. Only when thorough testing has indicated stability and accuracy over the expected range of forecasting requirements can a model be put to operational use.

The model identified in this paper is a fair predictor of general hardware development cost. As such, it proves that using many varied programs as a data base for estimating a cost model is a viable concept. The use of many data points from different technology domains has several advantages. First, it increases the number of degrees of freedom in the statistical analysis which allows more explanatory variables to be used.

Second, the wider range of data available provides a deeper insight into the nature of the relationship between cost and various program factors. For example, a limited analysis of spacecraft data may have led to the conclusion that quantity elasticities are always greater than unity. In fact, production economies of scale should be achieved once the initial prototype stage is passed.

Third, a model based on a wide range of technologies should be more suitable for estimating the cost of new designs that may have no historical analogies. Finally, validating the model over different time periods may improve the confidence in estimates made far into the future. The model described here demonstrates that such a model can be constructed and will estimate cost within fairly reasonable bounds.

In addition, several economic conclusions can be drawn from the data model. The analysis shows that significant economies of scale with respect to weight exist for nearly all types of development hardware. The more complex the hardware, the greater the economies of scale. Also, the lower the weight of a subcategory, the greater the economies of scale are for that subcategory. Some classifications, such as ships, may even have diseconomies of scale with respect to weight. The estimated elasticity of cost with respect to weight ranges from 0.43 to 1.30 with an average value of approximately 0.65.

Economies of scale with respect to unit quantities also are evident. The range of estimated elasticities is very wide, from 0.3 to 1.17 with the average around 0.58. Some types of systems have diseconomies of scale. These are mostly very low production quantity systems such as spacecraft. The conclusion is that a modified learning curve such as Figure 5.2d on page 5-3 may be appropriate.

The use of a Culture variable was proven effective for combining different technologies in the same data base. A methodology for deriving a quantitative measure of Culture was presented and shown to produce good results. For future space developments, Culture may be the most significant variable the cost analyst has to select. Weight and quantities will usually be given, but the particular hardware may not fall into any of the historical subcategories. It may be possible to estimate Culture for future programs using deterministic methods, such as a function of the ratio between weight and quantity. Another possible method of estimating new Cultures would be interpolation or extrapolation of existing Cultures.

The inclusion of a time based variable causes the effects of time to be removed from the other variables in the model. Hence, the model could be used for long-range planning if the future effect of time could be predicted. It was found that the cost of programs is increasing with time, even after the effects of inflation are excised. The time related cost growth is not at a constant rate. The magnitude of cost growth appears to be from 0.0 to 3.0 percent per year. The exact nature of this time related phenomenon is not yet understood although it is believed to be a combination of increasing performance, complexity and technology offset by improving productivity and development methods.

Finally, the benefits of design inheritance were clearly demonstrated. Substantial reductions in cost from using existing designs rather than starting from scratch are evident from the large negative coefficient of the Generation variable. Cost savings of about 22 percent for each subsequent generation are predicted by the model. This fact has been used to great advantage on military acquisition programs and should be incorporated whenever possible in the space program.

The statistical model does have some deficiencies. Most of the problems result from the wide range of estimated coefficients for subcategory models as shown in Figure 5.9 on page 5-12. The model of all data must effectively average these coefficients which results in erroneous estimating at the subcategory level.

In addition, it was found that the modeling of time related behavior (e.g., inflation, productivity, technology, et cetera.) is inaccurate. The model assumes that the rate of change is constant but, in reality, it varies.

The combination of these two deficiencies makes the statistical model unsuitable for long-range cost estimates of advanced space programs. Although the basic technique demonstrated here is sound, it must be refined even further to produce acceptable cost estimates. The specific weaknesses of the statistical model have been identified and potential solutions will be implemented in the future.

6. RDT&E AND PRODUCTION COST

6.1. INTRODUCTION

In addition to the statistical model described in the previous chapter, a cost model based on expert knowledge was also developed. The intent of this section is to outline the results of a study performed to determine an appropriate cost modeling methodology using heuristic methods. Within this report, the advantages and disadvantages of the proposed cost estimating methodology, the algorithms developed for testing, algorithm coefficients and exponents utilized in the development and testing of the aforementioned algorithms will be discussed.

6.1.1. BACKGROUND

Upon the review of current cost estimating methodologies, it was determined that there were three (3) basic approaches to cost modeling. The first approach being the commonly used cost estimating relationship (CER) approach. This approach, being the most unsophisticated approach, is a simplistic, two-dimensional cost analysis equation which relates an independent variable, usually weight, to the dependent variable cost.

The CER approach was rejected as a methodology primarily due to the disadvantage that the CER is specific in nature and only predicts costs for similar items as those utilized in the development of the CER. AMCM requirements dictate the operability of a cost model to estimate items and systems which will be designed and manufactured in the absence of cost history of similar items or systems.

The second methodology is more complex than the CER approach in that it is a series of CER's, each CER being a sub-assembly or subsystem of the aggregate system. This methodology was rejected by the cost model developer for the same reason as the CER approach.

The third methodology, a parametric cost estimating system, is the most complex methodology in that it is composed of many CER's which interact so that a change in one input variable changes the result of more than one CER. This methodology was also rejected due to the extreme complexity and operator training required for the systems available. Also, the reviewed parametric cost estimating systems lacked capabilities desired and/or required for AMCM. These capabilities are as follows:

(1) A visible and verifiable calibration basis.

(2) The ability to perform technology forecasting.

(3) The ability to use various engineering design parameters other than weight.

6.1.2. PURPOSE

The primary design criteria in the development of AMCM is the advancement of NASA cost estimating tools to be utilized during the systems design process when technical/cost trade studies can be performed in a timely and cost efficient manner. The trade study parameters available to the model user shall include parameters known to be major cost drivers in present models as well as parameters determined to be relevant to cost estimation in the next century, but not included in the present models. These parameters shall include but not be limited to system requirement, system design and performance. An emphasis will be placed upon the effects of the technological selection process and the resulting technology forecasting algorithms.

6.1.3. GOAL

The goal of this study is to test and/or prove the feasibility of developing a cost model which incorporates both the advantages of present cost estimating technology and the advancement of that methodology to achieve the purposes outlined in section 6.1.2. Specifically, the features of AMCM methodology to be designed, prototyped and tested will be as follows:

1. To be operable at micro levels. For example, the system should be calibrated and usable at low levels of the work breakdown structure for sub-assemblies such as cables, harnesses, tanks, filters, printed circuit boards, heaters and structural parts.

2. The model should be capable of operating at the assembly level on such items as transmitters, antenna arrays, processors, communication devices, thrusters, gyros, gimbals and rocket motors.

3. The modeling system should be operable at subsystem levels. These items would include subsystems such as: communications, propulsion, structural, tracking, telemetry and command, attitude and velocity control, hydraulics, landing gear and environmental control.

4. The model should have the ability to be operated at system levels such as: the orbiter, satellites of all types, landers, command and service modules, lunar and planetary exploration vehicles and space laboratories.

5. The model should be operable at program levels such as: the space shuttle, aerospace planes, space stations and interplanetary manned missions.

6. The model should have the flexibility of processing increasing levels of information as the data becomes available (initial definitions at system level progressing through sub-assembly subsystem levels). The model should have the capability of determining cost driving coefficients for new systems as a function of the assemblage of calibrated and known sub-assemblies, assemblies and subsystems.

7. The model should incorporate and provide guidelines for technology selection and technology forecasting based upon a sound technical basis. This includes the processing and calibration of historical data to provide analytic rationale for the assessment of the cost of technologies to be developed in the future.

8. The model should be able to assess the feasibility of achieving selected performance parameters in terms of calendar time and the associated cost risk of deviations in the specified date of operational capability and the model predicted feasibility forecasts.

9. The model should have the capability to assess cost as a function of critical engineering performance criteria, as a replacement for the commonly used parameter of weight.

10. The model and relevant data base should be easily calibrated in such cases where responsible personnel determine that any cost model dimension, as constructed, is undesired or requires modification.

11. The model should be less complex to operate than current sophisticated parametric cost estimating systems.

12. The model should be designed in such a manner as to optimize computer calculation speed in the event that Monte Carlo simulation would be utilized for the assessment of cost versus technical risk.

13. The model should have provisions for the use of industry cost model parametric mapping.

6.1.4. RESULTS

The results of the modeling development effort are discussed in this section. The study methodology was based upon two (2) criteria.

The first criteria is a requirement for a rapid prototype. The rapid prototype concept was instituted in order to eliminate potential cost methodologies and to test and evaluate the chosen methodology as to its feasibility in accomplishing the purposes and goals as set forth in sections 6.1.2. and 6.1.3, respectively. The second criteria that the study methodology was based upon was that of a broad but shallow data base. The broad but shallow data base criteria is founded upon the requirement for a cost model to estimate all components, assemblies, subsystems, systems and programs in US or European historical data bases as well as components through systems for future space programs not envisioned at the time of model construction. The aforementioned scope dictates a model with variable coefficients which exceed costing dimensions of items in the historical data base. For example, manned spacecraft with interplanetary capability, manned vehicles or stations in excess of ten million pounds or space rated nuclear reactors in excess of two megawatts.

A summary of the results of the development of AMCM are described in this section with further discussion of resultant methodology, algorithms, example runs, scope, validation and recommendations in section 6.2. The chosen methodology, utilizing nine (9) required input variables and three (3) optional input variables, was successful in obtaining good statistics on the broad and shallow data base to be discussed in section 6.2.5. and section 6.5. The successful modeling approach utilized the parametric cost estimation system concept described in section 6.1.1. has achieved 100 % of the goals specified in section 6.1.3.

In addressing those items denoted as achievability proved, it should be noted that item eight (8), technical feasibility test has been determined as achievable by illustration through the analysis of super computer performance relative to time and complexity over time. However, the mathematical implementation of assessment routines for technical specification and schedule cannot be implemented in the spreadsheet software that was utilized for the fast prototype concept.

In conclusion, the results and tests of the methodology have been successful, and will provide NASA technical personnel with the advantages over currently available systems as follows:

(1) Intermediate steps identifiable and dissected.

(2) Applicable at all levels.

(3) Validity grows with data base.

(4) Can use out-of-house data.

(5) Easy to use.

(6) Emphasizes technology forecasting.

6.2. COST MODEL OVERVIEW

The following section provides an overview of the AMCM methodology, algorithms, an example calibration run, forecasting run, the intended scope of the cost predictions, the resultant validation test and the system designers' recommendations.

6.2.1. METHODOLOGY

AMCM, as opposed to the static concept of CER's, was designed and is operated under the concept of a generalized system. Specifically, the CER is a static concept in that each CER represents a unique and exclusive product in terms of weight or other parameters. The generalized concept of AMCM, in order to satisfy the criteria in section 6.1.3., incorporates the observations of a multitude of CER's and determines a pattern of cost dimensional movement across the entire spectrum of products relative to that dimension. For example, the movement in cost opposed to weight for satellite structures, the movement in the cost of cables relative to weight and the movement in the cost of propulsion relative to weight are not observed as separate and distinct models, but rather as items of different complexity in a continuous complexity plane and the differences in the cost over weight slopes are reference points on a surface of potential slopes for other items in the complexity

GOAL SPECIFIED	GOAL ACHIEVED	ACHIEVABILITY PROVED
Estimating Level		
1) Sub-assemblies	Yes	
2) Assemblies	Yes	
3) Subsystem	Yes	
4) System	Yes	
5) Program		Yes
Utilities		
6) Comp./Decomposition	Yes	
7) Technology Forecasting	Yes	
8) Tech Feasibility Test		Yes
9) Performance Analysis		Yes
Operations		
10) Easily Recalibrateable	Yes	
11) Operational Complexity	Yes	
12) Calculation Speed	Yes	
12) Industry Model Mapping		Yes

plane. This concept allows the interpolation of cost as a function of weight coefficients for items not in the historical calibration library. This concept is a product of the generalized system developed by Mr. Frank Frieman at RCA in the 1960's. During implementation, this concept generates a complexity factor in lieu of a linear relationship by the removal of the independent variable as a relative factor due to normalization.

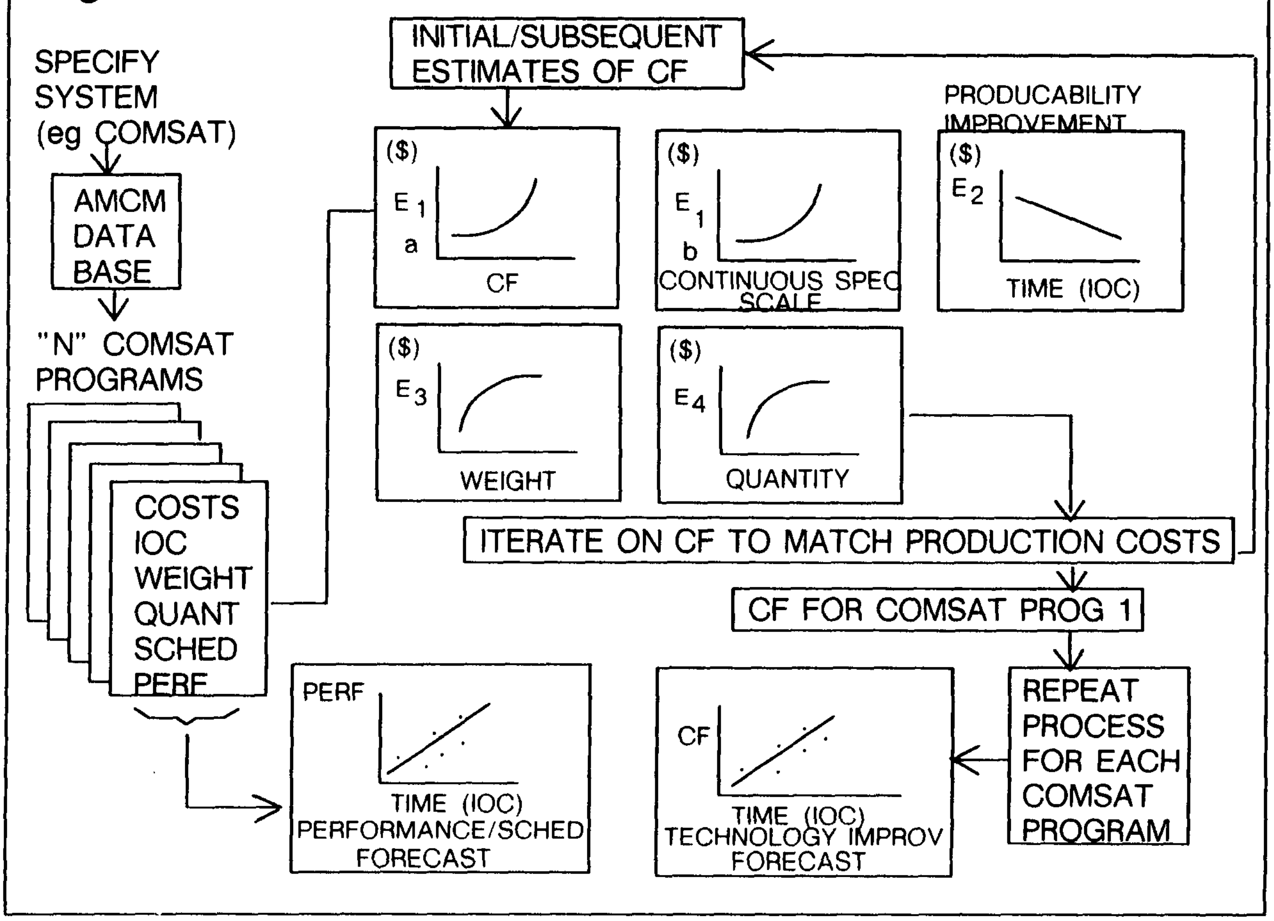
The process of normalization is the extrapolation of an input parameter to a neutral point in the cost hyper-plane as determined by the model designer. For example, in the comparison of the historical costs of two different satellites, the cost analyst modifies the historical costs to a common economic point in time (ie. 1985 dollars) before he performs relative cost complexity comparisons. The analyst would also normalize the historical cost as a function of the build quantity to preclude inaccurate assessments of relative cost, due to the economies of quantity scale.

For example, a build quantity of ten satellites may be reduced to a build quantity of one satellite and a build quantity of three satellites is also artificially reduced to the build quantity of one. In both cases, the modifications to actual history are artificial and for comparative purposes only. In the sophisticated parametric system, all relevant common cost dimensions are neutralized in this manner to an artificial reference point. The same procedures as outlined for economic escalation and build quantity are utilized in the normalization of weight, state-of-the-art (SOA), schedule and other variables to determine the relative cost of the subject item at a single point in the cost hyper-plane. This point, commonly referred to as "cost complexity factor" is then used by the model as a seed for the generation of similar technologies where the cost parameters lie at different distances on each dimension of the cost hyper-plane.

6.2.1.1. CALIBRATION

The process of calibration is a user instigated process where the independent variable, cost, is known and the complexity factor is the dependent variable and is determined by the model by the iterative method as several of the algorithm coefficients are a function of the calibration of the dependent variable, complexity factor. This process is illustrated in Figure 6.1 on page 6-6.

Figure 6.1 EVALUATING COMPLEXITY



6.2.1.2. FORECASTING

The forecasting process in the model operations is the reverse concept of the calibration process in that the complexity factor derived from historical calibration is now the independent variable, and cost is the dependent variable. This process is illustrated in Figure 6.2 on page 6-7.

Example cost model runs of both of these processes will be displayed and discussed in section 6.2.3.

6.2.2. ALGORITHMS DEVELOPMENT

Figure 6.3 on page 6-8 provides the reader with the algorithms developed for AMCM as a result of extensive iterative design and testing. The left hand column of the figure provides the user with the inputs for the example runs discussed in section 6.2.3. The second column provides the processing algorithm. The third column provides the primary output variables, RDT&E costs and production costs, as well as intermediate values utilized by the model for purposes which will be discussed in section 6.4., Model Derivation.

Figure 6.2 ESTIMATION PROCESS

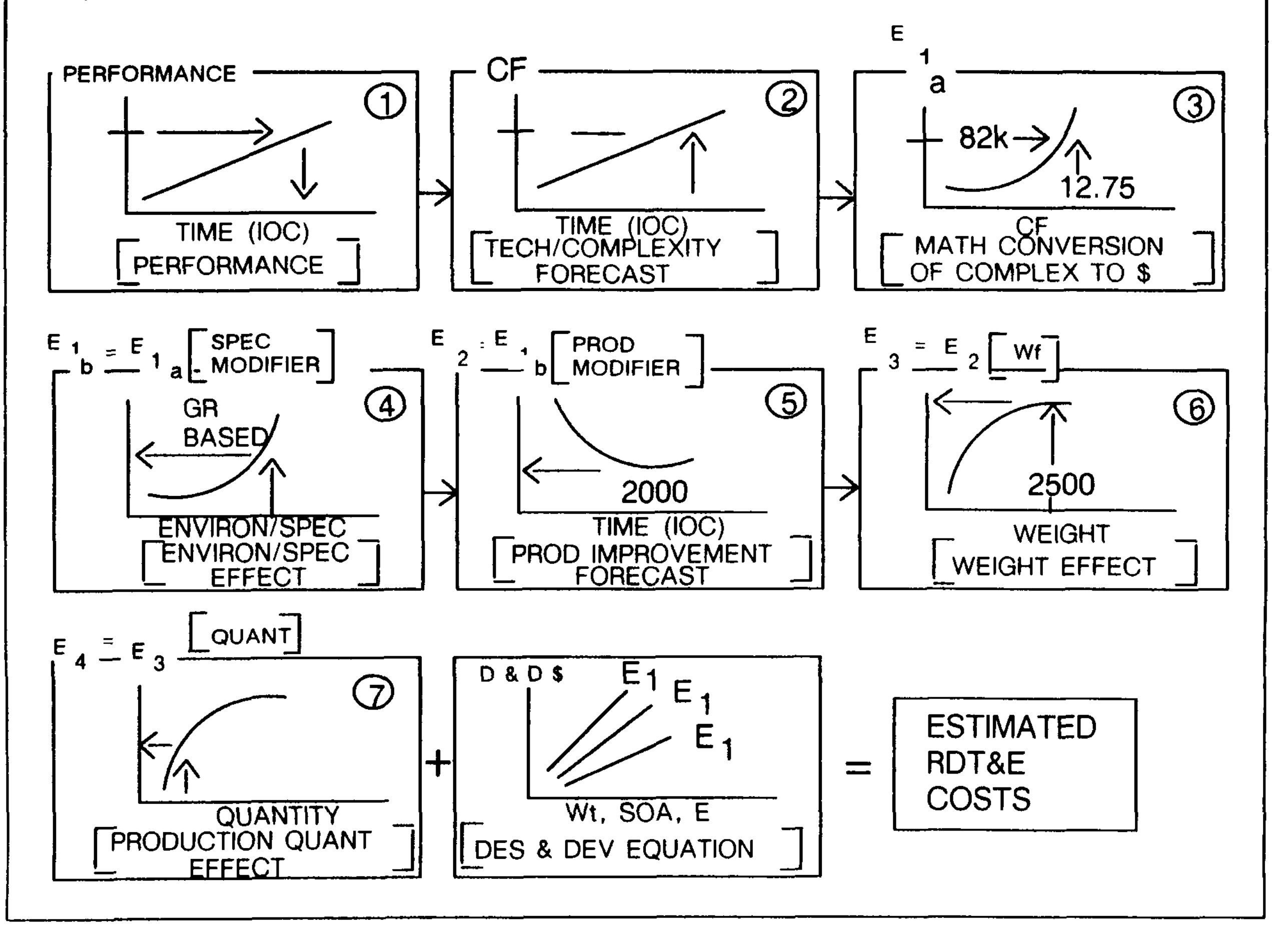


Figure 6.3 AMCM ALGORITHMS

INPUT	SPACE SHUTTLE ORBITER	OUTPUT
cf = 8.263 s = 1.80	$E_1 = 69.323\, e^{.73(cf)} (cf)^{.3436} e^{.2172(s)(cf)^{.6333}}$	264,497
IOC = 1981	$E_2 = E_1 e^{.0036(1987-IOC)E_1^{.2}}$	343,853
f = .2 w = 154,950	$E_3 = E_2 w^{.3[1-e^{-(.15\ln(E_2)-.37)^3}] + .3(f) = .3}$	835,906,643
Q = 2	$b_1 = [.9085Q^{.0065} - .0361Q^{-.1755} \ln(E_2/1000)]$ $E_4 = \dfrac{E_3 Q^{[\ln(b_1)/\ln(2) + 1]}}{1{,}000{,}000}$	.7343 Production $ 1,229 m
SOA = 3	$RDT\&E = .251\, E_1^{-.74} w^{.63} E_1 [.3(SOA) + .1]$	RDT&E $ 12,004 m

Note:

cf	=	Complexity Factor	s	=	Specifications
IOC	=	Initial Operational Capability	f	=	Electronic Fraction
w	=	Weight	Q	=	Quantity
SOA	=	State-of-the-Art	$	=	Constant 1987

6.2.3. EXAMPLE RUN

This section illustrates the current operational Lotus 1-2-3 version of AMCM. For illustrative purposes, a calibration run is provided on the left of Figure 6.4 on page 6-9. It should be noticed by the reader that in this case the base complexity factor is the unknown variable for the calibration of history, and the known variable is cost history (1213.7m$) with the derivation of complexity factor (8.26) being determined by the other technical and programmatic variables processed by the system algorithms provided in Figure 6.3 on page 6-8.

On the right side of Figure 6.4 on page 6-9 resides the forecasting process where the known variable is the complexity factor (8.26) derived from historical calibration and the unknown variable is the cost projection as determined by the same equations used in the calibration process.

This example is fundamental to the AMCM concept in that given the same parameters for calibration and forecasting, the same results would be obtained. In normal estimating circumstances, the technical input variables and programmatic input variables would normally be different as the same project is never performed twice. It should

Figure 6.4 AMCM EXAMPLE RUN

DS: Run 7	Calibration	Forecast
	ORBITER	ORBITER
Input Parameters		
Year (IOC)	1981	1981
Design SOA		3
Economics	1987	1976
ESC Factor (to TY$)	1.00	0.45
Weight (w)	154,950	154,950
Quantity (Q)	2	2
Specs (s)	1.8	1.8
Elect fract (f)	0.2	0.2
Cost ($m)	1213.7	?
Base Complexity	?	8.26
Operations		
Base Complexity	8.26	8.26
Mod-1, Tech Up		
Mod-2		
Operational Cmplx	8.26	8.26
Cost Generation		
E-1	264,522	264,522
Tech Imprv (d)	1.30	1.30
E-2	343,887	343,887
Weight	2,431	2,431
E-3	835,978,163	835,978,163
Quantity	1.45	1.45
CIC %	0.726	0.726
E-4 Prod ($m)	1213.7	1213.7
Design SOA	0.0	1.0
NR E-fctr	0.0-	24.4
NR-wt fctr	1861	1861
RDT&E ($m)	0	12,004
DDT&E + Prod (87 $m)	1,214	13,218
ESCALATED	1,214	6,014

SPACE CAPSULE	IOC YEAR	CALIBRATED COMPLEXITY	PREDICTED COMPLEXITY
Mercury	1962	6.53	6.62
Gemini	1965	7.43	7.17
Apollo CSM	1968	7.62	7.71
Lunar Module	1969	7.94	7.90

also be noted that the complexity factor derived by the calibration process (8.26) is only a reference point from the calibrated project and is usually modified for technological advancement from past to future programs. This subject will be discussed further in the model input variable section, "Complexity Technology Advancement Slope", discussed in section 6.3.10.

6.2.4. SCOPE

The scope of the model described in this report is inclusive of research, design, test and evaluation (RDT&E) and production cost. The model is designed to cover all of the hardware related activities associated with RDT&E and production including systems engineering, design engineering, program management, data, hardware, tooling and test equipment and government associated in-house costs. It should be noted that the costs predicted by the model would only be inclusive of those costs used to calibrate the model or derive complexity factors. In order to make direct comparisons of complexities derived in the calibration process, the data base must be equivalent in the scope of the costs included for each project calibrated.

6.2.5. VALIDATION

In order to validate the model, technology estimating relationships were generated from calibrated runs of similar products. For example, the data base discussed in section 6.5. contains cost history for four (4) manned space capsules.

The complexities derived by the model range from a 6.53 to a 7.94, and these values have been regressed over the initial operational capability (IOC) which ranges from 1962 to 1969. A technology CER was generated in the form complexity =.183 * (IOC - 1900) - 4.73 with an r-squared = .92. The following data is displayed to show the differences between model calibrated values and the predicted complexity values by the technology prediction equation discussed previously.

In order to validate the model, Table 6.1 on page 6-11 was constructed from regressions upon complexity values for twenty-one (21) product categories. The table includes coefficients for technology prediction equations for the twenty-one products and their corresponding r-squared values. Complexities generated by these technology prediction equations were then input to the model in the fore-

Table 6.1 VALIDATION – 40 YEAR DATA BASE

DS: CALIB-1

	N	A	B	R	67cplx	87 cplx
Space	11					
Capsules	4	-4.73	0.183	0.92	7.56	11.23
Space Labs	4	0.89	0.084	0.98	6.49	8.16
Launch Booster	3	2.11	0.054	0.82	5.72	6.79
Aircraft	38	-0.69	0.114		6.97	9.26
Fighters	13	-0.01	0.099	0.94	6.59	8.57
Attack	5	0.96	0.082	0.21	6.46	8.10
Bombers	5	-1.77	0.143	0.91	7.79	10.64
Transport-FW	7	-3.16	0.158	0.86	7.40	10.55
Rotary Wing	8	0.52	0.091	0.61	6.62	8.44
Vehicles	12	1.96	0.025		3.64	4.14
Tanks	4	1.95	0.041	0.86	4.72	5.54
APC's	2	3.41	0.009		4.03	4.22
Trucks	6	0.52	0.025	0.88	2.17	2.66
Ships	29	2.61	0.055		6.28	7.35
Ac Carriers	5	1.55	0.078	0.97	6.80	8.36
Submarines	7	2.59	0.059	0.83	6.56	7.74
Cruisers	4	7.38	0.015	0.57	6.35	6.04
Amphip Assault	5	1.17	0.067	0.85	5.66	7.00
Destroyers	5	3.07	0.050	0.97	6.44	7.44
Frigates	3	-0.10	0.089	0.99	5.88	7.54
Other	7					
Super Computers	2	-0.65	0.163		10.29	13.55
Autos-Commercial	2	0.91	0.016		1.95	2.26
Rifles	3	-0.17	0.014	0.66	0.77	1.05
Missiles	22	-4.30	0.126		4.15	6.67
ICBM's	3	-1.90	0.109	0.89	5.37	7.54
Air-Air	5	-2.25	0.093	0.75	3.99	5.86
Air-Surface	6	-4.18	0.124	0.31	4.11	6.59
Surface-Air	4	-3.77	0.115	0.29	3.96	6.27
Anti-Tank	4	-9.38	0.189	0.38	3.31	7.09

casting mode. The cost predicted by the model was then compared to the recorded cost for each program and plotted in Figure 4.6 on page 6-14 to display the accuracy of the model in the prediction of a wide range of products including spacecraft, launch boosters, four types of aircraft, two types of vehicles, six ship types, and two missile types.

Attack aircraft, trucks, air-to-surface missiles, surface-to-air missiles, and anti-tank missiles were excluded as the categories contained diverse types within the categories and therefore are not directly comparable. An example is the Harrier attack aircraft VERTOL (vertical take-off and landing) capability which makes it quite unique relative to the other attack aircraft. Another example is military trucks. The trucks contain a data base of wide performance ranges as a function of the various drive systems. An example of normalization for these situations is available but not included in this report. It should be noted that for all spacecraft types the r-squared value averaged .91.

In Figure 6.5 on page 6-13, the writer has displayed graphical representation of the span of complexity derivation for the GE/PRICE-H system and AMCM. The average

Figure 6.5 COMPLEXITY RANGES

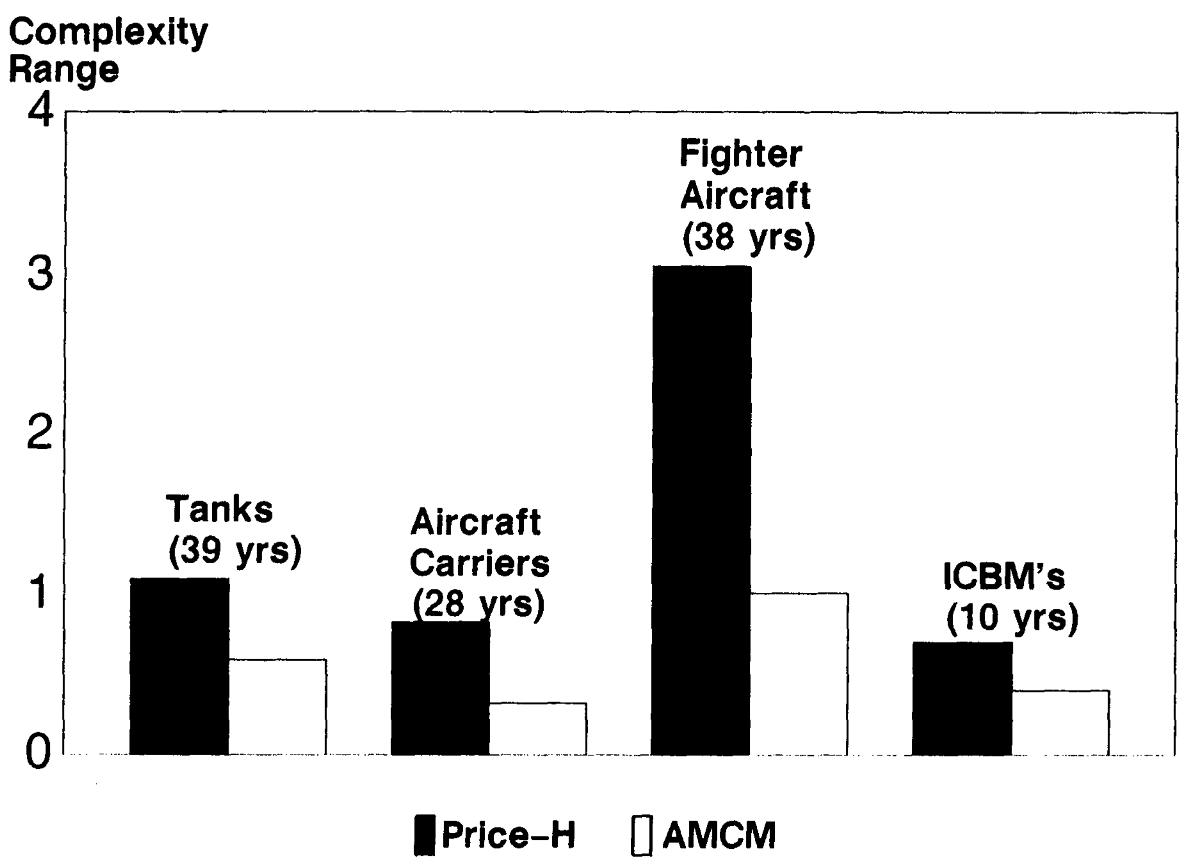
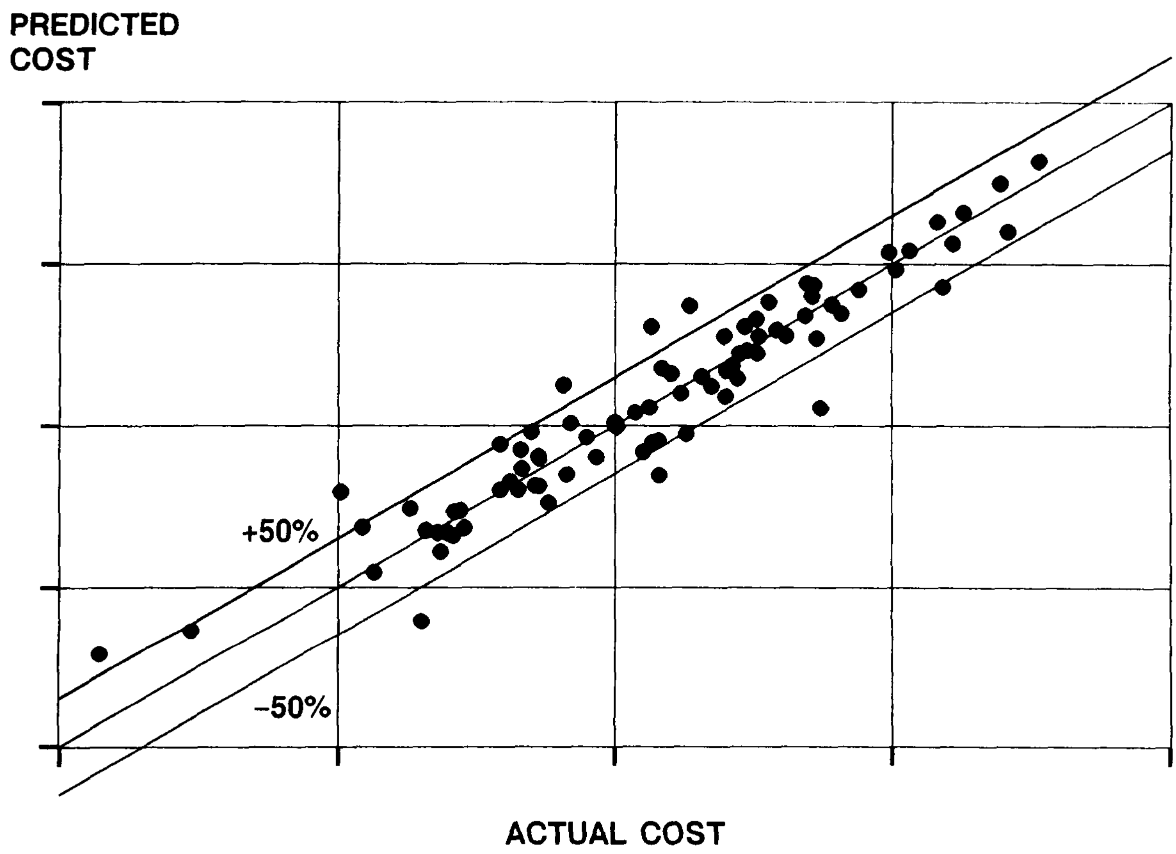


Figure 6.6 AMCM VALIDATION VARIANCE



difference in AMCM developed complexity factors is only 54 % of the difference in the PRICE-H complexity factors. As the difference in complexity factors for similar products indicates cost variances unexplained by the model, AMCM explains substantially more of the cost variance than PRICE-H.

6.2.6. CONCLUSIONS/RECOMMENDATIONS

Upon review of the testing results of the AMCM prototype algorithms relative to the purpose and goals set forth in sections 6.1.2. and 6.1.3., the writer recommends that the chosen methodology be implemented into operational software concurrent with further calibration efforts on a detailed and specific data base, as opposed to the broad and shallow data base used for the prototype version discussed in this report.

6.3. COST MODEL VARIABLES

This section discusses the rationale, definitions and cost effects of the cost driving input parameters for AMCM. Table 6.2 on page 6-14 has been provided as a summary of the cost variables utilized in AMCM. The input variables denoted as common are used in both RDT&E and production cost predictions. The top of the table indicates the suggested source of the

Table 6.2 MODEL VARIABLE SOURCES

MODEL VARIABLE	COST ANALYST	PRG MGMT ENGR	TABLE	MODEL DERIVED
Common				
Cf	x		x	x
E-1				x
Weight		x		
IOC		x		
Specifications		x	x	
Escalation	x			
RDT&E Costs				x
SOA		x	x	
Production Costs				x
Quantity		x		
Elect %	x	x		
E-2				x
E-3				x
CIC	x			x
E-4				x

input parameter in the performance of a cost analysis. In some cases, input parameters such as CIC (section 6.3.7.) or complexity factor are derived by the model or provided by the cost analyst.

6.3.1. ECONOMIC ESCALATION

In the performance of the cost analysis, economic escalation tables are available to normalize the value of monetary units to a common point in time. By the use of funding profiles described by the analyst in conjunction with schedule start and stop dates, economic factors are derived to normalize historical programs to constant 1987 dollars and project model generated costs in 1987 dollars to then year dollars appropriate to the program under study. The scope of this effort is not covered by this report.

6.3.1.1. RDT&E

The escalation rates utilized in the calibration of AMCM algorithms are provided in Table 6.3 on page 6-15.

6.3.1.2. PRODUCTION

For calibration purposes of production programs an ACS derived production normalization table was utilized and is not recommended for use in future calibration efforts.

Table 6.3 NASA R&D INFLATION INDEX

FROM FISCAL YEAR	TO 1987	FROM FISCAL YEAR	TO 1987
1959	5.604	1977	1.984
1960	5.373	1978	1.841
1961	5.206	1979	1.681
1962	5.006	1980	1.519
1963	4.837	1981	1.369
1964	4.629	1982	1.252
1965	4.476	1983	1.180
1966	4.223	1984	1.114
1967	4.026	1985	1.071
1968	3.819	1986	1.038
1969	3.614	1987	1.000
1970	3.380		
1971	3.180		
1972	3.008		
1973	2.846		
1974	2.655		
1975	2.396		
1976	2.198		
TQ	2.153		

6.3.2. SPECIFICATIONS

The specifications input variable is an input parameter used to determine the amount of reliability and documentation requirements required for contract performance. The variable presently is indicative of the environment that the product is intended to operate in.

The values listed below are reflective of calibration undertaken for the comparison of PRICE-H generated costs at their respective platforms and then calibrated by AMCM.

ENVIRONMENT	AMCM SPECS VALUE
Ground	1.00
Mobile	1.15
Airborne	1.30
Space	1.67
Manned Space	1.80

The primary advantage of AMCM methodology is that in the calibration mode the model neutralizes cost differences for the respective environments so that data obtained from the industry operating in different environments can be utilized in the analysis of space transportation systems.

For example, a visual display calibrated from a fighter aircraft can be utilized by the analyst as a basis for estimating a visual display in a space laboratory environment.

It is recommended that an extensive study be undertaken to investigate the effects of the environmental test program, safety, configuration control, maintainability and parts quality effects on this variable. Figure 6.7 on page 6-16 illustrates these variables and how they may interrelate to derive the specifications variable in the future.

6.3.3. WEIGHT

Weight is an input variable in AMCM as an indicator of the relative size of projects under study. It has been determined that cost increases with weight as displayed in Figure 6.8 on page 6-17. The coefficients of the cost per weight equations have been modeled in Figure 6.9 on page 6-18 for electronic and structural items. The derivation of these equations will be discussed in section 6.4.3. Weight is input into AMCM in US pounds. However, it should be noted that the goal of AMCM is to reduce weight as a required input pa-

Figure 6.7 SPECIFICATION EVALUATION

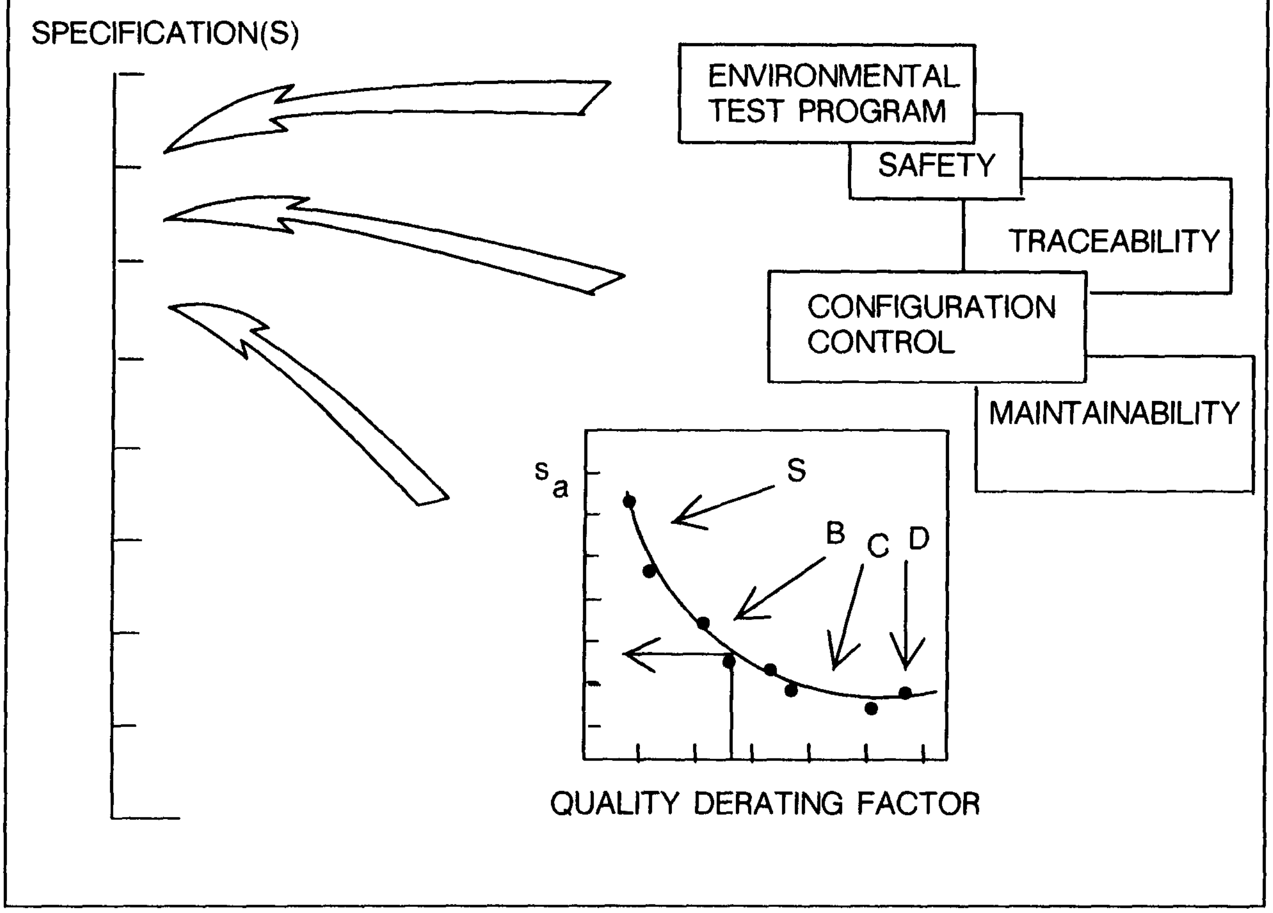
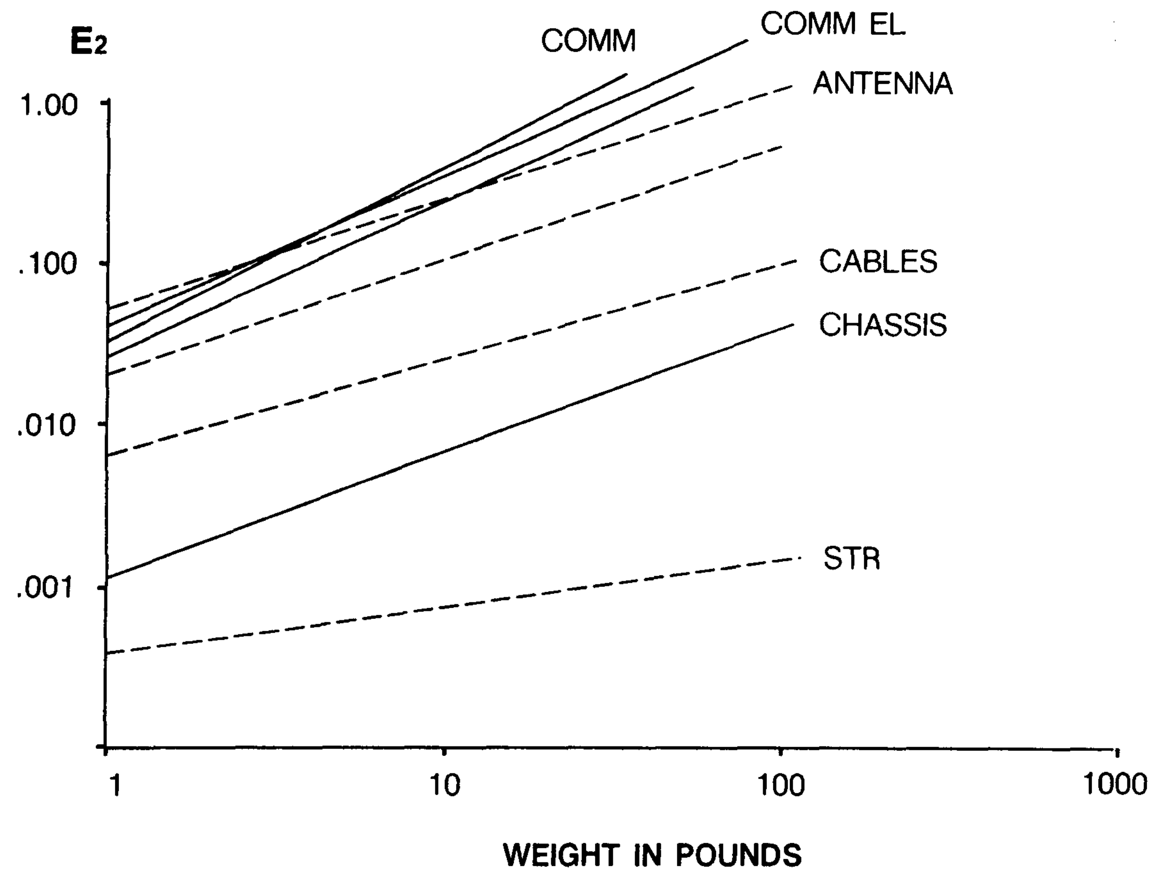


Figure 6.8 PRODUCTION WEIGHT SLOPES



rameter and incorporate the substitution of performance parameters.

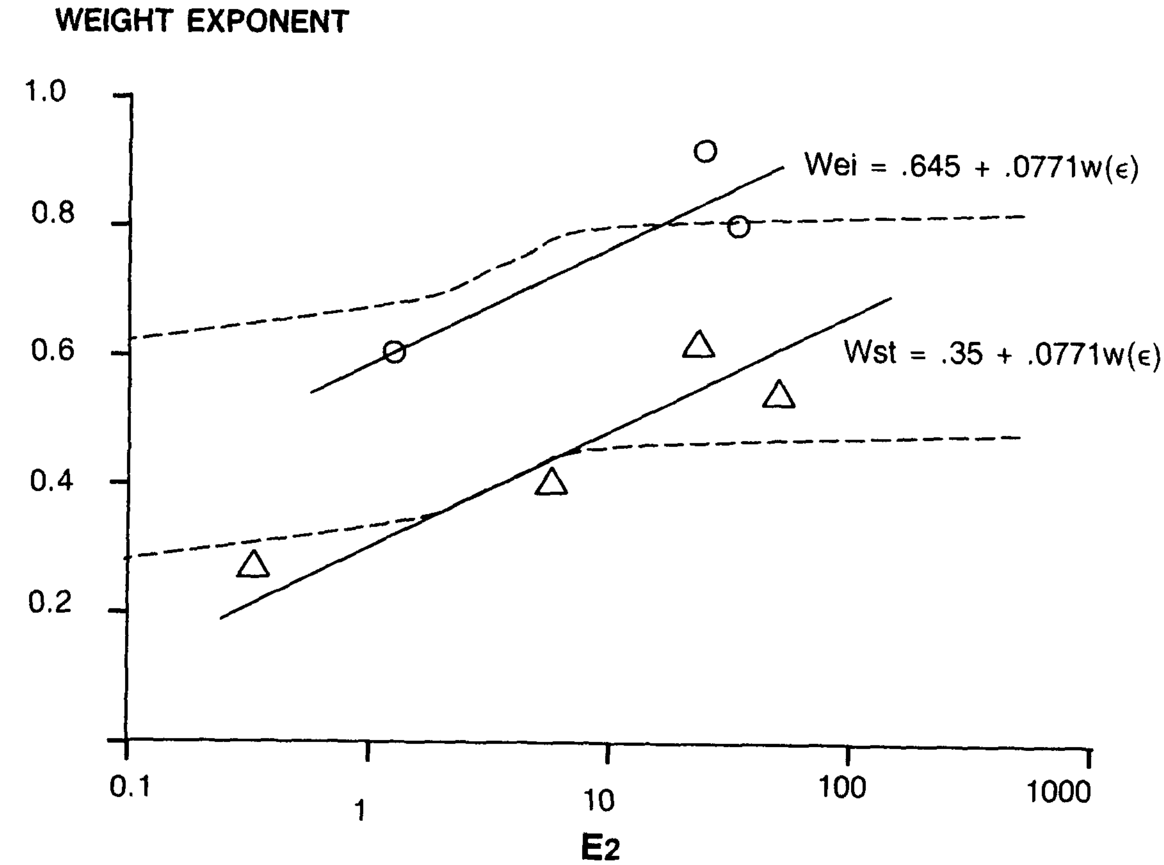
6.3.4. ELECTRONIC COMPOSITION FACTOR

The electronic composition factor is an input to AMCM which enables the model to interpolate between the weight slope function of electronics and structures. In viewing Figure 6.9 on page 6-18, the reader will notice one function for electronic products and one function for structural products. The electronic composition factor indicates the percentages of the product being estimated which are electronic in nature, structural in nature, or a combination of both. For example, the electronic composition factor of 1.0 would instruct the model to use the top function in Figure 6.9. An electronic composition factor of 0 would direct the model to use the bottom function in Figure 6.9. An electronic composition factor of .70 would instruct the model to use a weight slope exponent 70% of the distance between the electronic and structure functions.

6.3.5. STATE-OF-THE-ART FACTOR

The state-of-the-art (SOA) input variable indicates to the model the relative differences in the complexity of the engineering task. For example, in Figure 6.10 on page

Figure 6.9 WEIGHT SLOPE ESTIMATION



6-19 a SOA input variable of "1" denotes to the model that the design task only includes sustaining engineering support for the manufacture of off-the-shelf items and denotes the lowest possible RDT&E effort. The opposite end of the spectrum is the "principles observed" input variable "12" which indicates a component or system which has not been tested or manufactured in the subject application. The intermediate input variables denote the various classifications between these two extremes and are provided in Figure 6.11 on page 6-20

6.3.6. BUILD QUANTITY

This parameter describes to the model the quantities of hardware that will be built and applies the method of engineering tooling and test equipment. For example, if three identical satellites are built, the quantity variable is equal to 3.0. Another example is where one flight processor is built after a developmental test model. In this case, non-integer values are allowed and the analyst would input quantity equal to 1.7.

Figure 6.10 STATE-OF-THE-ART INPUT VARIABLE

JULY 1988	SOA	SEPTEMBER 1988	PRICE-H
Same/Simple Mod	1	Off-the Shelf	Mod-Simple
Extensive Mod	2	Mod-Minor	- Routine
State-of-the-Art	3	- Moderate	- Difficult
Advance SOA	4	- Extensive	- Complex
	5	- Complex	New Simple
	6	New Simple	- Routine
	7	- Normal	- Difficult
	8	- New MCP	- Complex
	9	- Diff I&T	Adv SOA-Routine
	10	Adv SOA-CFD	- Difficult
	11	- MDP	- Complex
	12	- Principals	

CFD - Critical Functions Demonstrated
MDP - Multiple Design Paths
MCP - Materials, Components and Processes
MOD - Modification

6.3.7. QUANTITY COST IMPROVEMENT CURVE

This variable is an optional input to the analyst in that the model predicts a cost improvement curve percentage as a function of the item complexity and build quantity. The curve is based upon the straight line average (SLA) philosophy. For example, an eighty-seven percent cost improvement curve is input as .87. Figure 6.12 on page 6-21 shows how the model determines the cost improvement curve as a function of the model determined complexity factor, E-1 (See section 6.4.1.) and build quantity.

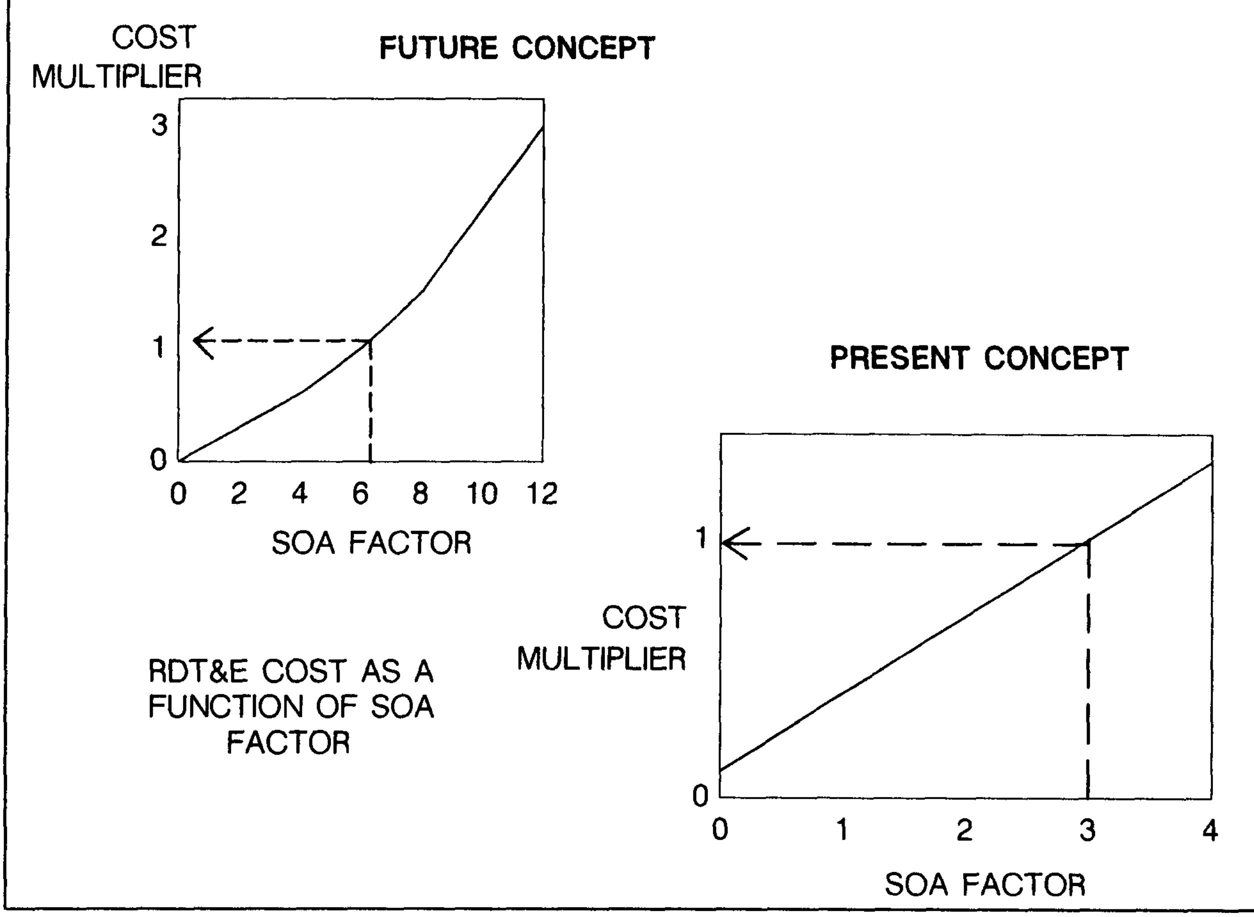
6.3.8. INITIAL OPERATIONAL CAPABILITY

Initial operational capability (IOC) indicates to the model the maturity of a products development in terms of improvements in methods and processes, along with performance. IOC is input as a calendar year numeric. For example the shuttle orbiter IOC is equal to 1981.

6.3.9. COMPLEXITY FACTOR

The AMCM complexity factor is a model peculiar resource numeric used to drive the cost model and differentiate one technology from another with respect to resources, both labor and materials, re-

Figure 6.11 STATE-OF-THE-ART FACTOR



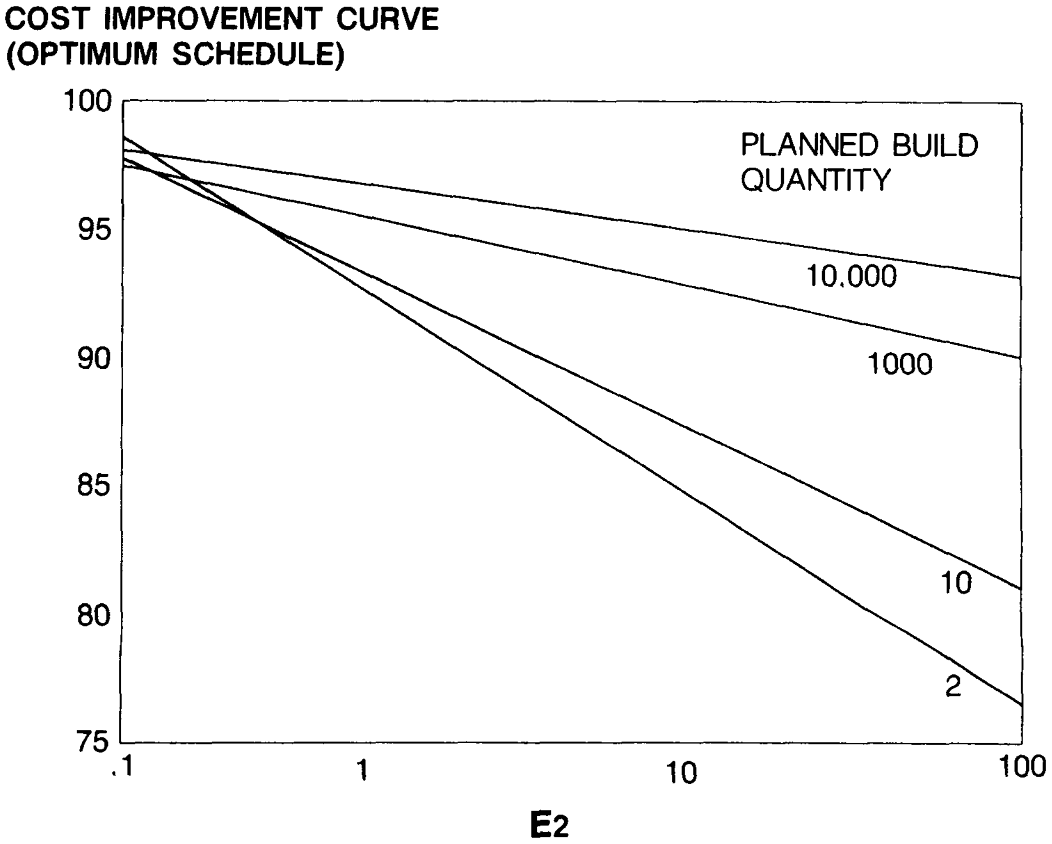
quired to produce the subject item. The complexity factor can be simply interpreted as the intercept of a multidimensional CER on the cost axis. Figure 6.13 on page 6-22 provides a generalized picture of the movement in cost as a function of the increases or decreases in the complexity factor. Figure 6.14 on page 6-23 is an illustration of the relative complexities of AMCM calibration upon the estimates for space station and results generated from mean values of other models.

The analyst must be careful in that intuitive assessments of complexity by engineers or the analyst are usually distorted in that complexity is not cost. For example, one would intuitively believe that a printed circuit card with an integrated circuit is more complex than the integrated circuit itself. However, the printed circuit card is a low technology mixed with a high technology integrated circuit to provide a moderate technology composite and a complexity for the loaded printed circuit card lies between the complexity of the bare printed circuit card and the integrated circuit.

6.3.10. COMPLEXITY TECHNOLOGY ADVANCEMENT SLOPE

The technology slope is the factor that indicates to AMCM the level of technological

Figure 6.12 COST REDUCTIONS WITH QUANTITY



progression for the technology under study. Figure 6.15 on page 6-24 illustrates the methodology for plotting calibrated complexities opposed to IOC. Regression upon these parameters provides the slope of complexity increase per year for a product line. This slope represents the cost for increasing performance over time. Table 6.4 on page 6-25 provides the slope per year for products ranging from super computers to automobiles as a reference in determining technology slopes. Figure 6.16 on page 6-26 compares the technological slopes of complex items (super computers) with slopes of simple items (trucks). It should be obvious that high complexity items contain proportionally higher technology slopes when compared to low complexity items. Figure 6.17 on page 6-27 is a graphical representation of the technology slopes studied to date and should be used as a guideline in assessing this critical model input for the determination of future program costs.

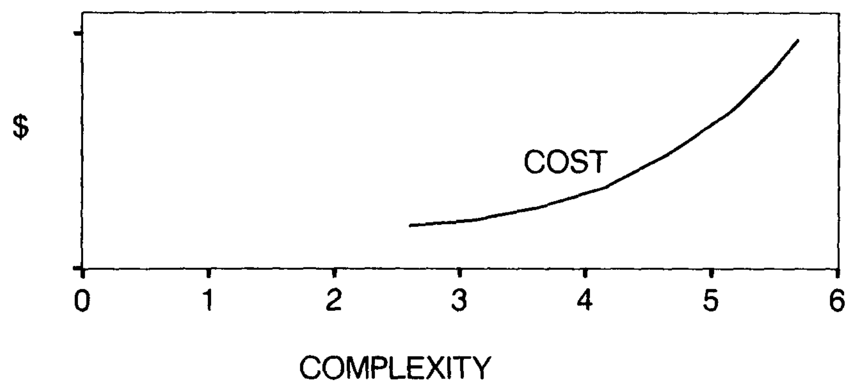
6.3.11. SCHEDULE

Schedule is a cost determinant in three ways:

1. It dictates the appropriate inflation index values for economic escalation/de-escalation.

Figure 6.13 COST COMPLEXITY

- DEFINITION
  - A MODEL PECULIAR RESOURCE NUMERIC USED TO DRIVE THE COST MODEL
  - IT REPRESENTS THE RESOURCES REQUIRED TO PRODUCE AN ITEM WHERE ALL COMMON COST DIMENSIONS ARE NEUTRALIZED
  - INTERACTNG WITH THE MODEL COST, COMPLEXITY CUSTOMIZES THE CER SLOPES AND PROVIDES MULTIPLE INTERCEPTS
  - COST COMPLEXITY SIMPLIFIES MATHEMATICS AND GRAPHICS



2. It determines technology availability (IOC).

3. It determines the impacts of accelerated or stretched-out schedules.

It is envisioned that AMCM will utilize RDT&E go ahead, RDT&E completion, hardware manufacturers start and production completion dates. From these dates the model will calculate time span in months and compare those specified months to month predictions by the model. Upon comparison of the specified and predicted time spans, a schedule acceleration/stretch-out impact assessment will be made. Figure 6.18 on page 6-28 displays three cost impact concepts in those cases where the schedule is abnormal. This consists of moderate cost increases for schedule stretch-out and two opposing viewpoints for schedule acceleration.

This dimension is not active in AMCM to date as the schedule prediction algorithm will be a function of complexity factor and therefore is not yet completed due to a lack of source data.

6.4. MODEL DERIVATION

This section describes the development of the model algorithms and the utilization of input variables relative to the calibration

Figure 6.14 COMPLEXITY COMPARISONS

Space Station Estimates		Calibration From Other Models	
US Co-orbit Plt	9.45		
Space Station	9.17	10.20	Att Determin
EVA Systems	9.14	9.70	ATT Cntrl Sys
Press Log Carr	8.73	9.07	Platform
Resource Node	8.57	8.97	Att & RC
US Lab Mod	8.31	8.93	EPS-Subsyn
Airlock	8.14	8.90	Comm Elect
US Polar Pltfm	8.13	8.86	Satellite-Comm
Habitation Mod	8.09	8.79	Comm Antennas
Mech & Joints	7.76	8.72	EPS-Synchronous
Propul Assy	7.25	8.65	Communications
Solar Pwr Mod	7.22	8.51	TT&C
Unpress Log Carr	7.12	8.22	Str.TC, Intrstge
Servicing Fac	6.70	7.13	Dispenser
Truss Assy	6.37	7.08	AKM
Att Payld Acces	6.04		

data base. Intermediate values used in the models' calculations for purposes of observations, testing and system interactions are denoted as epsilon values. The epsilon value derivations are explained in the following paragraphs.

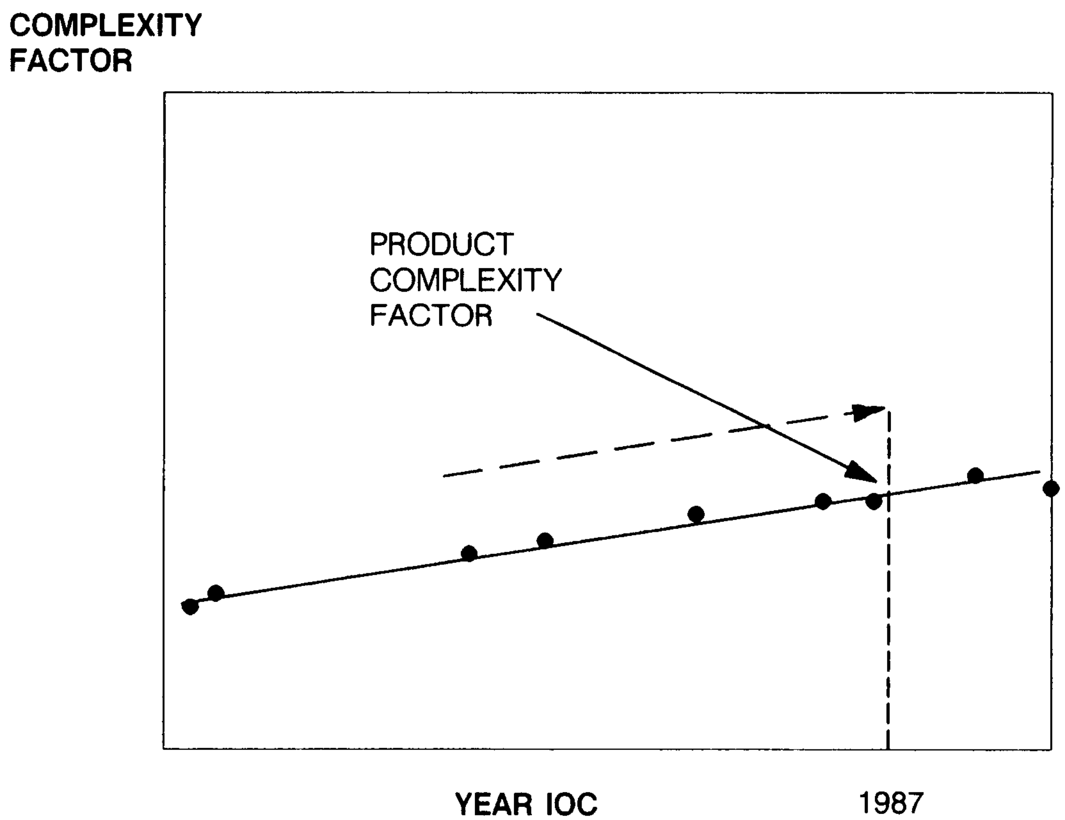
6.4.1. EPSILON-1

The purpose of the epsilon-1 intermediate value is to generate cost as a function of complexity and to modify that cost for the level of specifications to be imposed on the product or program.

The primary goal of this function is to retransform complexity to cost in the reverse process that was performed to transform cost to the model generated complexity factor. The coefficient and exponent is calibrated to simulate PRICE-H mapping in order to achieve objective number 13 in section 6.1.3.

The secondary objective of the algorithm is to normalize the specifications imposed on the product or program in order to enable the utilization of cost calibration from other environments or cultures. The higher order exponent modifies the cost as a function of the movement in the specification plane and the original complexity. The rationale is that higher complexity items

Figure 6.15 TECHNOLOGY NORMALIZATION



undergo more severe increases in cost as specifications are increased than lower complexity items.

The algorithm E-1 was obtained by manual iteration and interactive simulation of the calibration data base. (See Figure 6.3 on page 6-9)

6.4.2. EPSILON-2

The purpose of the epsilon-2 intermediate value is to modify the cost derived by epsilon-1, as described in paragraph 4.1, for technology improvements over time. These technology improvements are those cost reductions in the producability of an item due to advances in manufacturing methods, processes and yields. This parameter should not be confused with AMCM functions which increase complexity over time as a result of increases in performance and capabilities of a product type.

The goal of this function is to reduce the costs of implementing a designated technology of a past technology over time as a function of the IOC described in paragraph 6.3.8. The rationale of the algorithm mechanism is that high valued epsilon-1's which denote complex and advanced technologies sustain dramatic and contin-

Table 6.4 TECHNOLOGY SLOPE ANALYSIS

DATA POINTS	SLOPE/YEAR
Super Computers	0.163
Capsules	0.183
FW-Transports	0.158
Fighters	0.099
Space Labs	0.084
Submarines	0.059
Destroyers	0.050
Air-Air Missiles	0.093
Tanks	0.041
Trucks	0.025
Automobiles	0.016

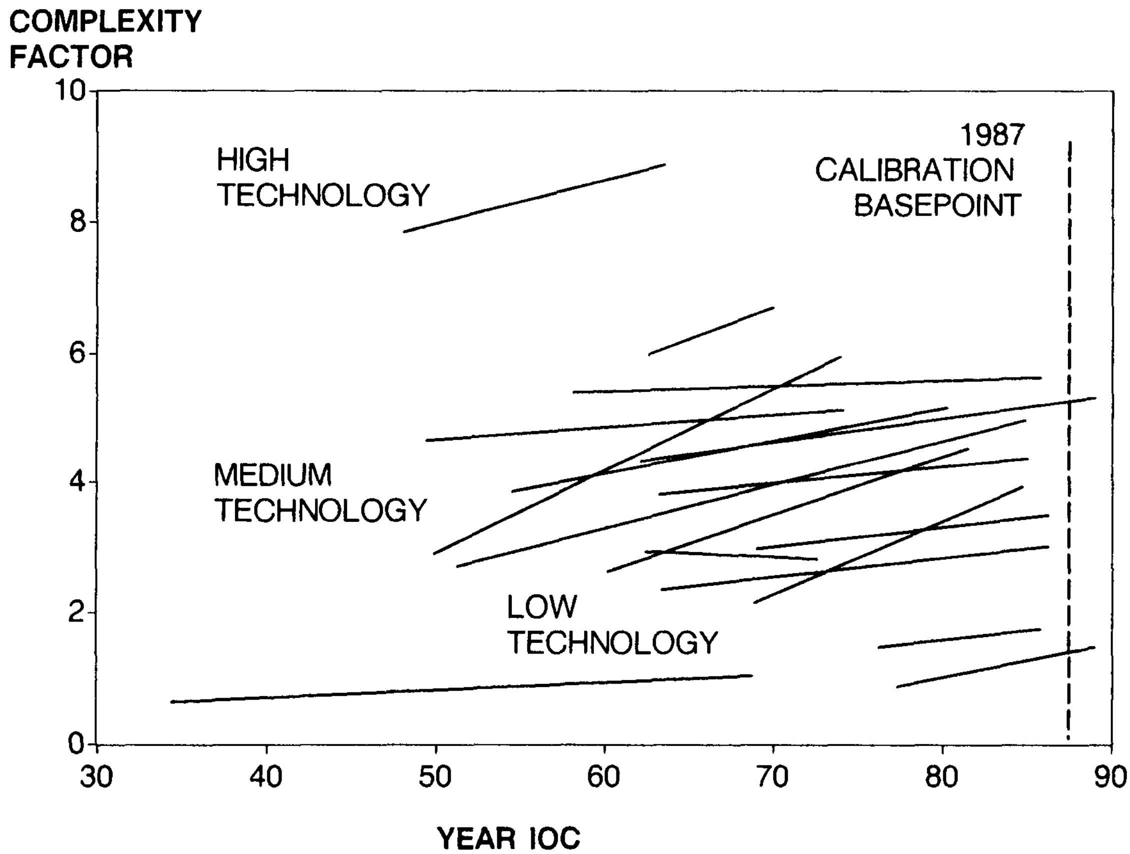
ual technological processes and methods improvements over time whereas low epsilon-1 values are relatively mature and therefore realize very small technological improvements over time in terms of cost reduction.

Figure 6.19 on page 6-29 is a graphical depiction of the epsilon-2 algorithm. The source data for the algorithm resides in the files of Rockwell International as backup for a paper presented by Mr. Darryl W. Webb entitled "Electronic Cost Complexity: Trends and Analysis", presented to The International Society of Parametric Analysts, April 26-28, 1983.

The mechanics of the epsilon-2 algorithm are as follows: 1. The operation 1987-IOC indicates the number of years difference between the product IOC and the model base point of 1987. 2. The coefficient .0036 and the exponent .2 determine the cost reduction slope as a function of epsilon-1 (complexity). (See Figure 6.3 on page 6-9)

As an example, the Mercury capsule in the data base yields a time span of twenty-five (25) years, 1962-1987, combined with an epsilon-1 value of 56.002 yields a combined exponent for e of .801 to provide a

Figure 6.16 AMCM TECHNOLOGY SLOPES



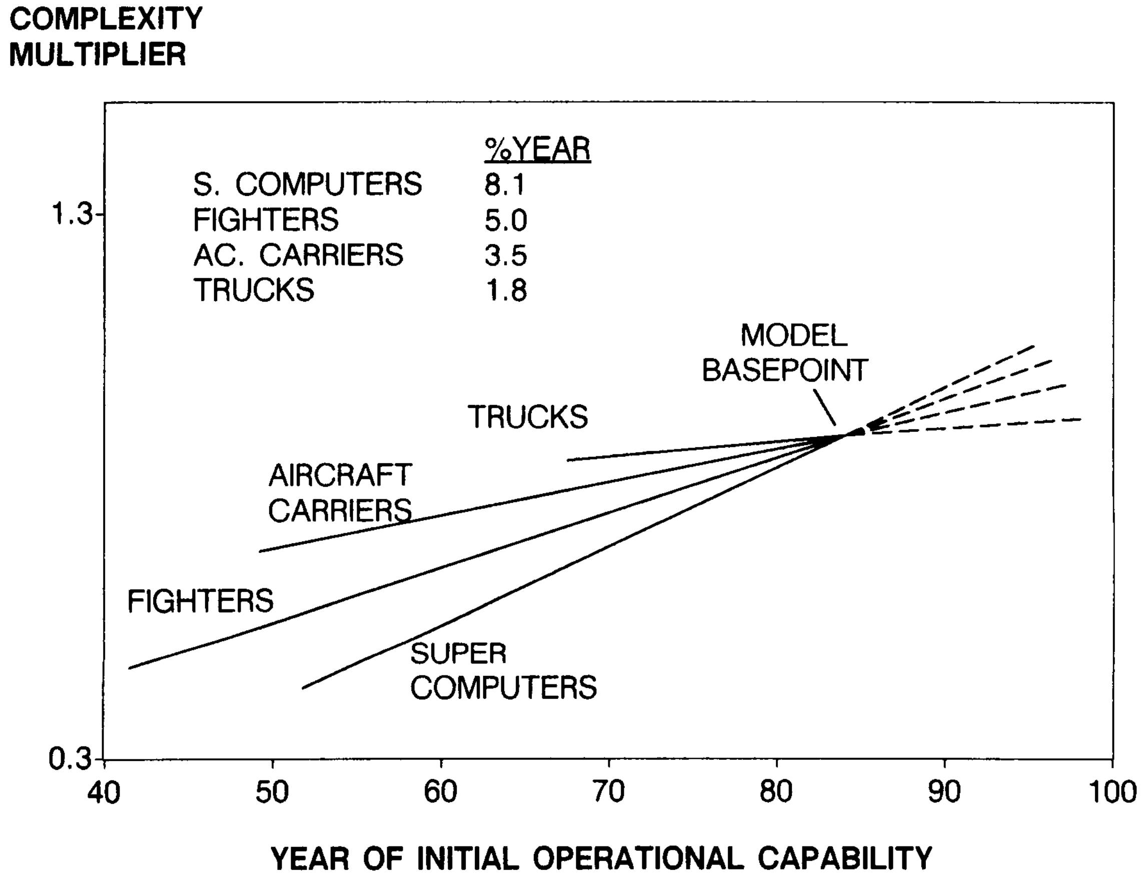
technological multiplier of 2.23 to be applied against epsilon-1.

6.4.3. EPSILON-3

The purpose of the epsilon-3 intermediate value is to modify the cost derived by epsilon-2, as described in section 6.4.2., for differences in product weights. These weight differences reflect the amount of mass of various products independent of differences in complexity.

The goal of this function is to normalize the costs of the relative size of products as a function of the weight parameter described in section 6.3.3. The rationale of the algorithm mechanism is that high valued epsilon-2's which denote complex and advanced technologies realize smaller cost reductions as weight increases than lower epsilon-2 valued items. The slopes for structural and electronic items illustrated in Figure 6.8 on page 6-17 and discussed in section 6.3.3. have been plotted as a function of epsilon-2 in Figure 6.9 on page 6-18. Observing Figure 6.9. the weight exponents for structural and electronic items from the data base described in section 6.3.3. have been regressed in semi-logarithmic form. However, upon testing of the data, the analyst has instituted lower and upper limits upon the exponents derived as

Figure 6.17 COST COMPLEXITY FORECASTING



a function of epsilon-2, and these limits are shown with dotted lines in Figure 6.9 on page 6-18 with the defining algorithm shown below. (See Figure 6.3 on page 6-9)

The mechanics of the epsilon-3 algorithm are as follows:

1. The epsilon-3 value is a function of epsilon-2 times weight to a derived exponent.

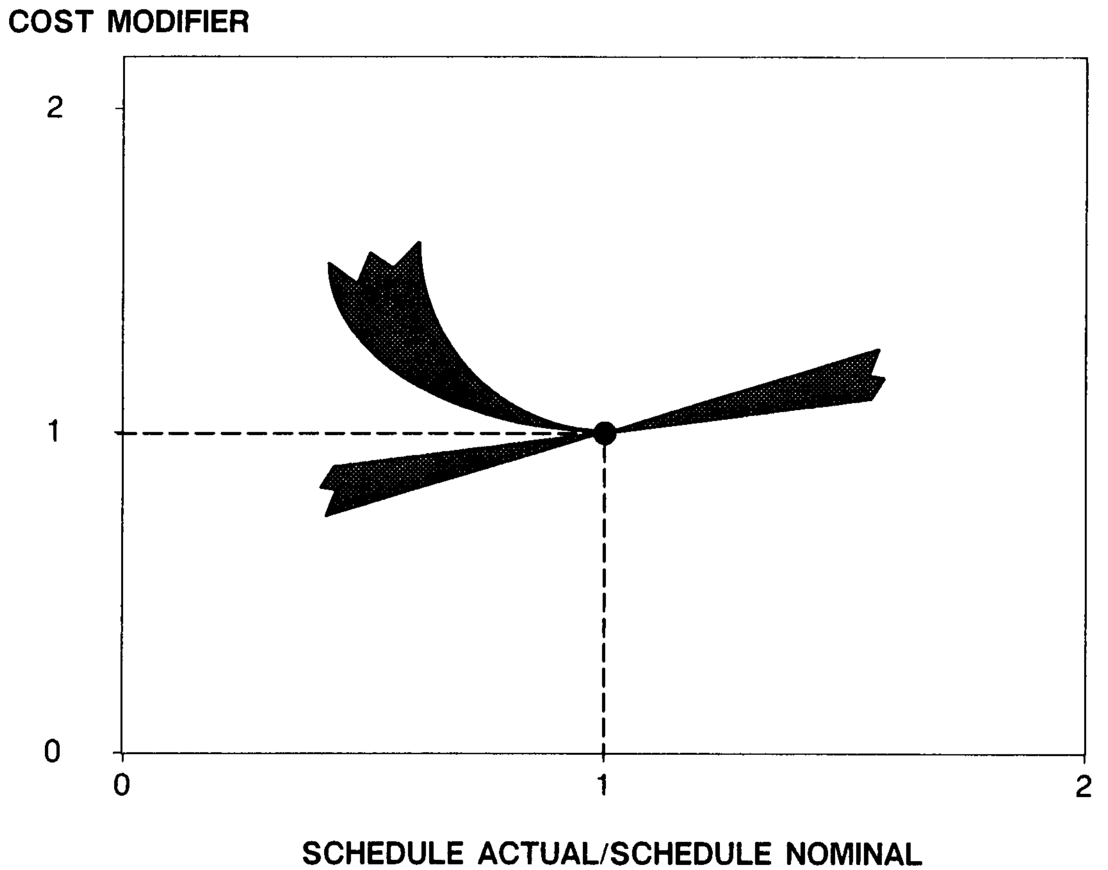
2. The derived exponent is a function of the epsilon-2 complexity of the product and the electronic composition factor described in section 6.3.4.

3. The exponent derived is merely an S-curve generated for a 100% structural item modified by the electronic composition factor which dictates an increase in height for the S-curve to ration it between the upper electronic curve and the lower structural curve displayed in Figure 6.9 on page 6-18.

6.4.4. EPSILON-4

The purpose of the epsilon-4 intermediate value is to modify the cost derived by epsilon-3, as described in section 6.4.3., for differences in product production quantities. These production quantities differ-

Figure 6.18 SCHEDULE MODIFIER



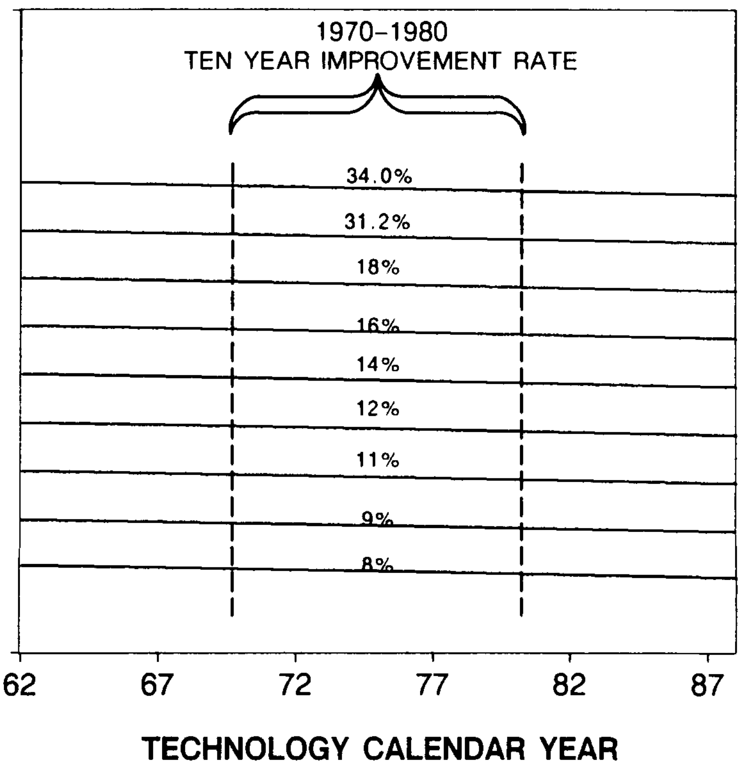
ences reflect the differing resource requirements applied to the products.

The goal of this function is to normalize the costs of the relative build quantities of products as a function of the build quantity parameter described in section 6.3.6. The rationale of the algorithm mechanism is that high valued epsilon-3's which denote complex and advanced technologies realize larger cost reductions as build quantity increases than lower epsilon-3 valued items. The slopes for quantity reduction illustrated in Figure 6.12 on page 6-21 have been plotted as a function of epsilon-3 in Figure 6.12. Observing Figure 6.12, the build quantity exponents for high and low epsilon-3 values have been regressed in semi-logarithmic form as a function of the intended build quantity. The rationale is that items built in very small quantities, such as two, are labor intensive and contain extreme cost efficiencies in terms of minimum buy and setup charges. Items built in extremely large quantities, such as 10,000, are tooled and tested with automated production processes to achieve the production rates necessary for items built in large quantities.

The mechanics of the epsilon-4 algorithm are as follows:

Figure 6.19 IMPROVEMENTS IN METHODS/PROCESSES

E1



1. The basis of the epsilon-4 equation is the mathematical simulation of the cost improvement curve prediction illustrated in Figure 6.12 on page 6-21. The equation predicts the cost improvement curve for an item as a function of the specified build quantity and epsilon-2, described in section 6.4.2. (See Figure 6.3 on page 6-9)

2. The straight line average cost improvement curve, predicted above, (example: 87% = .87) is then transformed by the equation below into the cumulative exponent for quantity and multiplied by epsilon-3 to provide cost as a function of build quantity.

3. For example, the Mercury value of epsilon-2 = 124,817 and the build quantity of 20 produces a cost improvement curve of 82.3% (b1=.823) per Figure 6.12 on page 6-21 and the equation provided above in section 6.4.4., the b1 equation. Progressing to the next equation, the cost improvement curve percentage of 82.3 (b1=.823) is transformed into a quantity exponent to yield a build quantity exponent of .72, to yield a

build quantity multiplier of 8.63. This effectively multiplies the unity value of epsilon-3 to provide epsilon-4, which is the cost of building 20 units of the subject item.

6.4.5. EPSILON-5

Not Presently Used.

6.4.6. RESEARCH, DEVELOPMENT, TEST & EVALUATION (RDT&E)

This section describes the data base, mathematical formulation and rationale for the prediction of RDT&E costs by AMCM. The purpose of the RDT&E section of the cost model is to provide a basis for the cost estimation of future systems or components where calibration of conventional cost estimating relationships is not possible due to the lack of experience with the subject technology. To accomplish this task, the movement of cost opposed to chosen input variables will be modeled in a generalized manner instead of the development of a CER specific to an equipment type. The scope of the estimated costs is inclusive of systems engineering, design engineering, quality assurance, program management, data and one set of development hardware.

6.4.6.1. DATA BASE NORMALIZATION

The data base utilized in the development of the RDT&E equations is shown in Table 6.5 on page 6-31.

6.4.6.2. ESCALATION

The source data has been normalized from then year dollars to constant 1987 year dollars by the use of standard escalation normalization techniques in the NASA inflation rate table shown in Table 6.3 on page 6-15.

6.4.6.3. WEIGHT

The source data after normalization for inflation, has been normalized for differences in weight. To accomplish this, the CER's in the US Air Force Unmanned Spacecraft Cost Model (Fifth Edition, July 1981) were observed to determine an approximate exponent for the movement in RDT&E cost as a function of unit weight. The CER's observed are as follows:

CER	Exponent
Communications	.56
Structure	.66
Communications Electronics	.70
Communications Antennas	.59

Other CER exponents were eliminated due to their equation type being inconsistent with observations made by the analyst on many other systems. The average of the exponents listed above is .63 and therefore this factor is utilized in AMCM to normalize different spacecraft systems for differences in weight. Plots of the observed CER's are shown in Figure 6.20 on page 6-32.

6.4.6.4. OTHER VARIABLES

Other variables have been normalized by AMCM in the development of Epsilon-1

Table 6.5 RDT&E DATA BASE

ITEM	R&D $M	Yr$	fctr	1987$	WT	WT .63	R&D Norm	E-1	Ef
CRAY-2	100	85	1.09	109.0	5,500	227.2	0.480	9,103,200	0.053
B1-B	9,000	83	1.19	10710.0	140,00	1746.1	6.134	937,000	6.546
ORBITER	6,000	77	2.10	12600.0	154,95	1861.3	6.769	264,520	25.591
LUNAR MODULE	1,298	67	3.98	5166.0	33,294	706.5	7.313	198,700	36.802
APOLLO CSM	2,327	66	4.20	9773.4	31,335	680.0	14.373	148,700	96.658
GEMINI	468	63	5.10	2386.8	6,934	262.9	9.078	126,000	72.049
NAV SATELLITE	240	83	1.19	285.6	1,800	112.4	2.541	107,000	23.744
MERCURY	72	60	5.25	378.0	3,985	185.5	2.038	56,000	36.394
JAGUAR	300	86	1.12	336.0	3,200	161.5	2.080	2,700	770.426

(E-1). For a discussion of these variables see section 6.4.1. The rationale for using E-1 in the RDT&E equation is that E-1 has normalized various factors to result in a measure of the relative hardware complexity and design configuration of spacecraft systems.

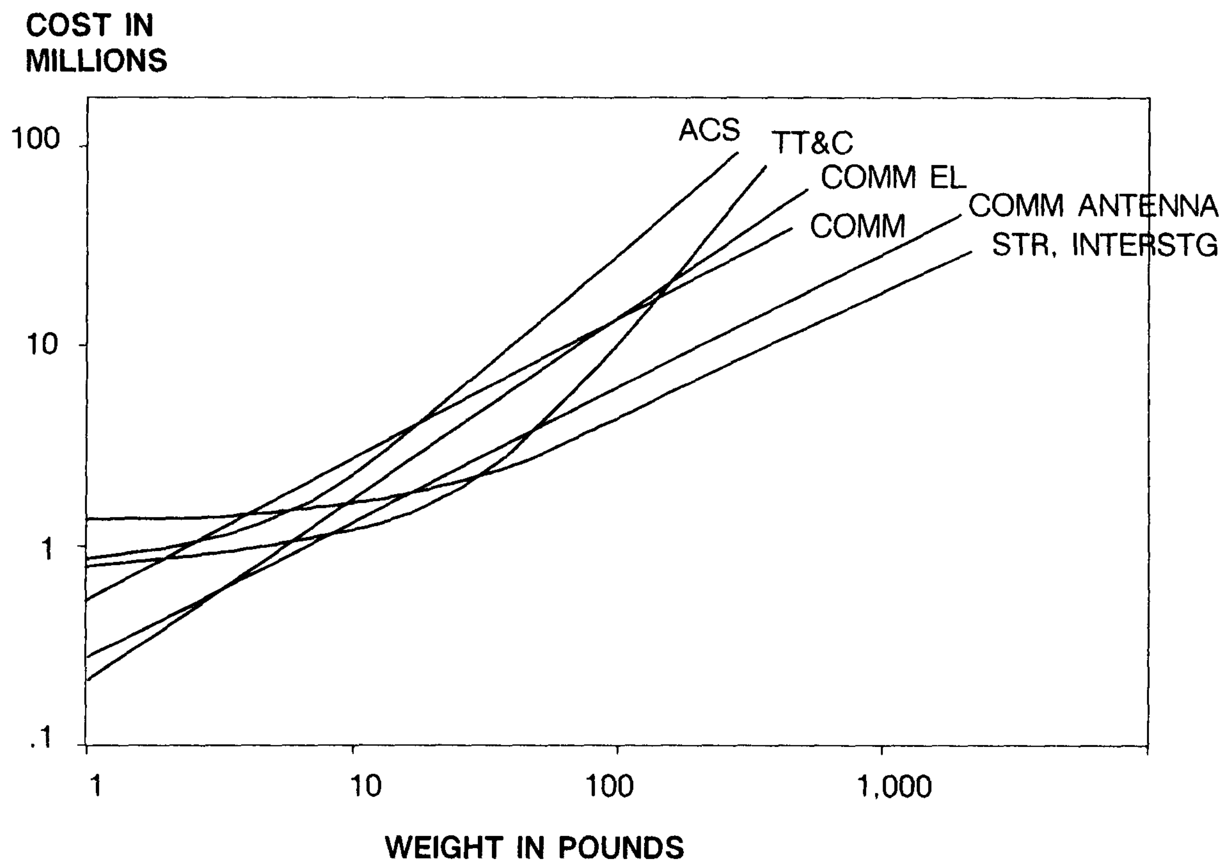
6.4.6.5. MATHEMATICAL EXPRESSION DEVELOPMENT

The process of equation development will be described in this section.

Upon regression of the normalized RDT&E costs exhibited in Table 6.5 on page 6-31 (column 7), opposed to E-1 values exhibited in Table 6.5 (column 9), the equation coefficients were provided with an r-squared value of .82. (See Figure 6.3 on page 6-9)

Subsequently, the normalization dimensions of cost versus weight and SOA are reinstituted to form the total RDT&E equation.

Figure 6.20 RDT&E COST SLOPES



6.4.6.6. CONFIDENCE STATISTICS

TBD

6.4.6.7. OBSERVATIONS

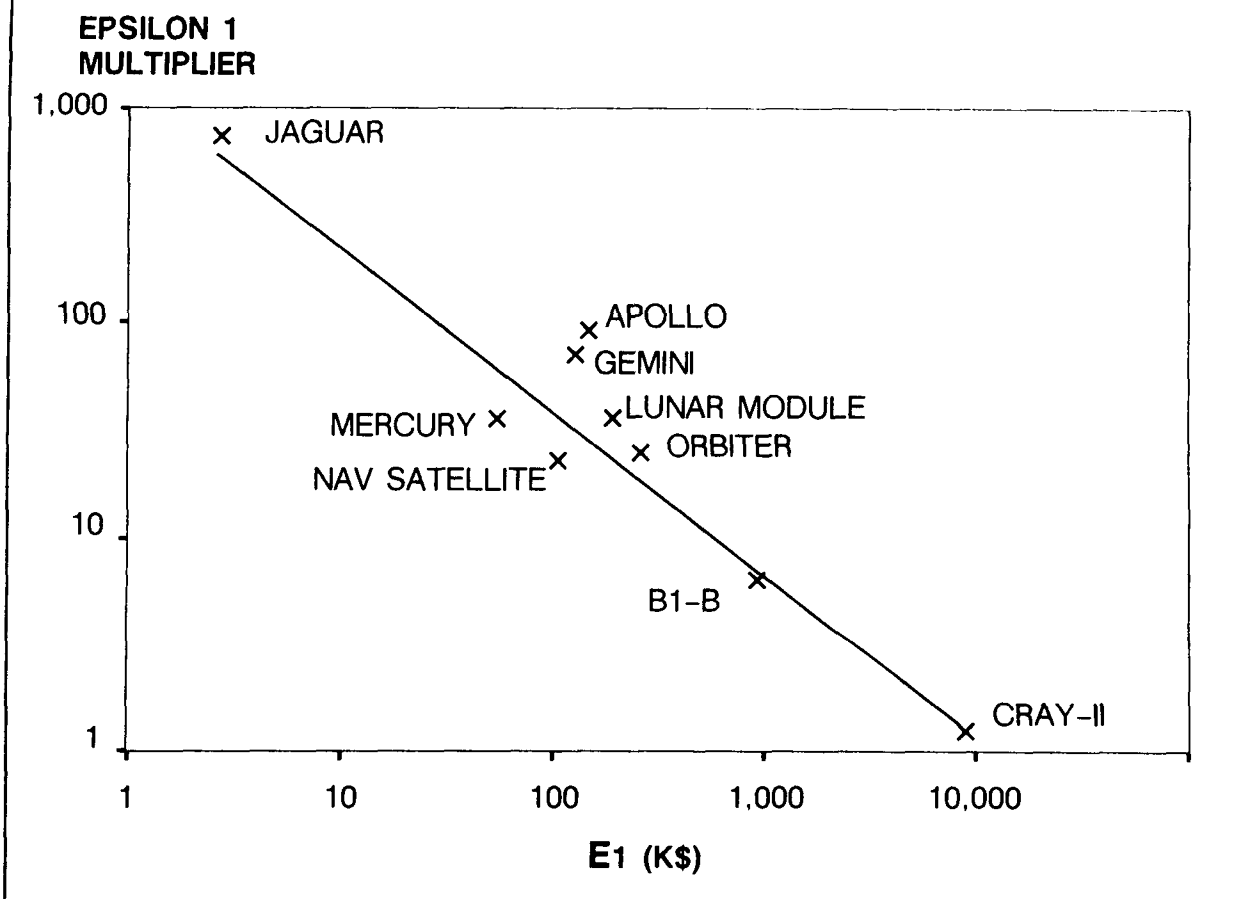
This section is a summary of the analysts comments and suggestions for future RDT&E equation development.

Observing the data plot in Figure 6.21 on page 6-33, it is apparent that the data plot versus the regression indicates possible variable omission which would be required to improve the statistics and confidence intervals. One possible variable is the project IOC, with the exception of Mercury, the analyst can see downward trends in the RDT&E factor opposed to E-1. Another possible variable is government specification versus best commercial practice which would provide a lower level equation coefficient for commercial practice than government specification as well as alteration in the equation exponent.

6.4.6.8. RDT&E CONCLUSION

The aforementioned RDT&E methodology, based upon limited available data, has great potential for application to the prediction of RDT&E costs as a generalized predictor for space systems to be encountered in the future. The primary limitation is the lack of data points for a three (possibly

Figure 6.21 RDT&E CALIBRATION



six) dimensional equation. A possible low cost solution to this problem is the incorporation of processed NASA data bases included in Red Star, Scientific Instruments Cost Model, Spacecraft Cost Model and the AMCM data base not utilized for this analysis.

6.4.7. PRODUCTION

The production costs for AMCM are the cumulative effects of the operations described in sections 6.4.1. through 6.4.4. The rationale being that the variables and relationships utilized to derive epsilon-4 are inclusive of the presently known parameters desired for a production phase, as well as the multiple production of units manufactured in the RDT&E environment. Future considerations for the RDT&E and the production environments should differentiate between these phases considering variables such as dedicated manufacturing processes such as robotics, differences in culture and RDT&E peculiar requirements such as qualification testing, engineering development models, brassboards and simulators.

6.5. THE RAPID PROTOTYPE DATA BASE

This section describes the data base used in the design, calibration and testing of the

AMCM prototype. The data base is intentionally shallow in any specific technology. For example, the number of spacecraft and space related products is limited to a total quantity of 64. To compensate for a shallow data base, the scope of coverage in technologies is very broad (ie. manned and unmanned spacecraft, aircraft of several types, land vehicles (earth and lunar), computers, ships, missiles and launch boosters. This scope also extends through the time dimension for 52 years including those projects analyzed from cost projections. The goal in this concept is to develop common cost estimating relationships and a technical basis as to why projects cost what they do over a broad basis. This will enable the future analyst to possess a tool for the cost evaluation of products not yet defined and those utilizing technologies, methods and manufacturing processes not to be developed for years to come.

Table 6.6 on page 6-35 summarizes the products analyzed for AMCM by category. This table is limited to those items where more than one product of the same type has been designed and manufactured. The value of this subset in the data base is in the technology forecasting of same, similar or different items as displayed in Figure 6.16 on page 6-26. In the respective columns of Table 6.6 are the number of items analyzed in each category, the coefficient of the linear equation provided by linear regression, the complexity factor slope per year for that linear equation, the r-squared value for those categories exceeding a quantity of three items and the complexity factor of each category predicted for the year 1987.

Table 6.7 on page 6-36 and Table 6.8 on page 6-37 provide the resultant complexity factors, base technological year and the predicted 1987 complexity factors for those items in the calendar year 1987. These two tables provide relative reference and calibration points for like or similar products where the technological complexity slopes have not been derived due to a lack of like or similar product history over limited or large time spans.

6.6. UTILITY ROUTINES

This section illustrates one of many possible AMCM utility routines which has been developed and tested to insure the accomplishment of the objectives in section 6.1.3., primarily the useful and accurate operation of the model in early system design phases when detailed design parameters are not available or speed of the estimate execution is necessary.

As an example of a flexible and powerful utility tool for AMCM operations, a technology composition modification or restructuring routine was devised to enable the analyst to modify a historical project to new requirements while maintaining the value of the historical data base as a technological reference point. For illustrative purposes, Table 6.9 on page 6-38 has been provided to demonstrate the capability of test software in utilizing the NASA developed lunar rover to derive complexity factors for a lunar based excavator.

Observing Table 6.9, the reader will notice that the lunar rover, developed prior to

Table 6.6 CF ESTIMATING RELATIONSHIPS

	N	A	B	R^2	87cplx
Space	11	-1.28	0.089		6.47
Capsules	4	-4.73	0.183	0.92	11.23
Space Labs	4	0.89	0.084	0.98	8.16
Launch Booster	3	2.11	0.054	0.82	6.79
Aircraft	38	-0.69	0.114		9.26
Fighters	13	-0.01	0.099	0.94	8.57
Attack	5	0.96	0.082	0.21	8.10
Bombers	5	-1.77	0.143	0.91	10.65
Transport-PW	7	-3.16	0.158	0.86	10.55
Rotary Wing	8	0.52	0.091	0.61	8.44
Vehicles	12	1.96	0.025		4.14
Tanks	4	1.95	0.041	0.86	5.54
APC's	2	3.41	0.009		4.22
Trucks	6	0.52	0.025	0.88	2.66
Ships	29	2.61	0.055		7.35
AC Carriers	5	1.55	0.078	0.97	8.36
Submarines	7	2.59	0.059	0.83	7.74
Cruisers	4	7.38	-0.015	0.57	6.04
Amphip Assault	5	1.17	0.067	0.85	7.00
Destroyers	5	3.07	0.050	0.97	7.44
Frigates	3	-0.10	0.089	0.99	7.54
Other	7				
SuperComputers	2	-0.65	0.163		13.55
Autos-Commercial	2	0.91	0.016		2.26
Rifles	3	-0.17	0.014	0.66	1.05
Missiles	22	-4.30	0.126		6.67
ICBM's	3	-1.90	0.109	0.75	7.54
Air-Air	5	-2.25	0.093	0.89	5.86
Air-Surface	6	-4.18	0.124	0.31	6.59
Surface-Air	4	-3.77	0.115	0.29	6.27
Anti-Tank	4	-9.38	0.189	0.38	7.09

Table 6.7 SINGLE POINT CALIBRATIONS - A

BROAD/SHALLOW DATA BASE SINGLE POINT CALIBRATIONS	YR	CF	1987 CF
SPACE STATION	96	9.92	9.17
US LAB MOD	96	9.06	8.31
HABITATION MOD	96	8.85	8.09
ATT PAYLOAD ACCESS	96	6.79	6.04
PRESS LOG CARR	96	9.48	8.73
UNPRESS LOG CARR	96	7.88	7.12
AIRLOCK	96	8.89	8.14
SOLAR PWR MOD	96	7.97	7.22
TRUSS ASSY	96	7.12	6.37
PROPUL ASSY	96	8.01	7.25
RESOURCE NODE	96	9.32	8.57
MECH & JOINTS	96	8.51	7.76
EVA SYSTEMS	96	9.89	9.14
SERVICING FAC	96	7.45	6.70
US POLAR FLATF	96	8.88	8.13
US CO-ORBIT PIT	96	10.20	9.45
ARMOURED VEHICLES			
M-113 A1	64	4.01	4.22
LVTP-7	71	4.07	4.22
SP-GUN (175MM)	62	6.65	7.69
PRICE-H MAPPING			
M=4, PLT=1.0	87	1.10	
M=4, PLT=1.8	87	1.22	
M=4, PLT=2.0	87	1.35	
M=8, PLT=1.0	87	5.37	
M=8, PLT=1.8	87	5.47	
M=8, PLT=2.0	87	5.62	
M=12, PLT=1.0	87	10.33	
M=12, PLT-1.8	87	10.42	
M=12, PLT=2.0	87	10.40	
M=4, PLT=1.0	87	1.25	
M=4, PLT=2.5	87	1.52	
M=8, PLT=1.0	87	5.50	
M=8, PLT=2.5	87	5.82	
M=12, PLT=1.0	87	9.83	
M=12, PLT=2.5	87	10.05	

Table 6.8 SINGLE POINT CALIBRATIONS - B

BROAD/SHALLOW DATA BASE
SINGLE POINT CALIBRATIONS

	YR	Cf	1987 Cf
MISC			
LUNAR ROVER	71	8.36	11.29
ORBITER	81	8.26	8.76
AGENA	66	6.34	7.48
CENTAUR	66	6.92	8.06
AGENA-W/O PROP	66	7.72	7.72
CENTAUR-W/O PROP	66	8.11	8.11
REACTOR PS-20KW	66	7.23	8.91
REACTOR PS-100KW	66	7.38	9.06
GRD REACTOR, 1MW	84	7.86	8.10
GRD REACTOR, 100MW	84	7.85	8.09
GRD REACTOR, 1000MW	84	7.85	8.09
SATELLITE SUBSYSTEMS			
STR, TC, INTRSTGE	72	6.72	8.22
TT&C	72	7.02	8.51
COMMUNICATIONS	72	7.16	8.65
COMM ANTENNAS	72	7.29	8.79
COMM ELECT	72	7.41	8.90
AT&T CNTRL SYS	72	8.20	9.70
ATT DETERMIN	72	8.71	10.20
ATT & RC	72	7.48	8.97
EPS-SUBSYN	72	7.44	8.93
EPS-SYNCHRONOUS	72	7.22	8.72
PLATFORM	72	7.57	9.07
DISPENSER	72	5.64	7.13
AKM	72	5.59	7.08
SATELLITE			
SATELLITE-COM	72	7.36	8.86
ASSET	63	5.55	7.94
OGO-1	64	6.55	8.84
VELA-IV	64	6.58	8.87
ATS-1 (A)	66	6.76	8.86
ATS-2 (B)	67	6.51	8.50
ATS-5 (E)	69	6.48	8.28
INTELSAT-III	69	6.34	8.14
IDCSP-A	70	6.62	8.31
M-35	71	6.85	8.45
DSCS-II	72	6.47	7.97
DMSP	76	6.10	7.20
HEOS-1		5.87	
HEOS-2	72	7.41	11.84
ESRO-1 A/B		5.86	
ESRO-II		5.82	
ESRO IV	72	7.50	11.93
COS-B	75	7.96	11.50
TD	72	8.18	12.61
DIAL		5.10	
AZUR		6.62	
HELIOS A/B	74	8.81	12.65
AEROS		6.21	
SYMPHONIE	74	9.15	12.99
UK-3		5.35	
UK-4		4.76	

Table 6.9 LUNAR BASE EXCAVATOR

PROGRAM	INPUT WT-%	INPUT $-%	CF	ENVIRON	WT LBS	PREDICTED EPSILON
LUNAR ROVER (1971)						
FRAME	15%	5%	5.62	2.55	76	589
MOBILITY	29%	28%	6.65	2.55	150	3,297
CREW STATION	5%	6%	6.13	2.55	28	706
NAVIGATION	4%	30%	7.05	2.55	20	3,532
POWER	29%	26%	6.58	2.55	446	3,064
SPACE SUPT EQUIP	17%	5%	5.56	2.55	89	589
	100%	100%	7.02	2.55	509	11,774
EXCAVATOR						
COMPONENTS COMMON TO ROVER						
MOBILITY			6.65	2.40	850	5,845
CREW STATION			6.13	2.40	35	
NAVIGATION			7.05	2.50	25	529
POWER			6.58	2.40	1,150	1,623
SPACE SUPT EQUIP			5.56	2.40	180	6,394
						570
MODIFIED ROVER COMPONENT FRAME			5.20	2.40	635	664
COMPONENTS UNIQUE TO EXCAVATOR						
EXCAVATORS			5.50	2.40	700	5,815
AUGER			5.30	2.40	100	296
BUCKET			4.80	2.40	320	252
AUGER MOTOR/DRIVE			6.30	2.40	150	1,384
CAMERA			6.90	2.50	35	1,753
			6.61		4,180	25,132

1971, was calibrated to an AMCM complexity factor of 7.02 under the column entitled "cf". In this problem, the analyst desired to estimate the sub-assembly complexities for the lunar rover to enable him to formulate a new sub-assembly composition for the lunar excavator. To accomplish this, the analyst, through research and engineering estimates, input the weight percent and cost percent estimates for the lunar rover assemblies. Upon execution, the AMCM technology composition modification utility routine derived the complexity factors which would generate the input mass/cost resource distribution. This process is commonly referred to, in the parametric community, as demultiplexing.

In order to derive the system complexity for the lunar based excavator, in the bottom half of Table 6.9 on page 6-38, the analyst inputs the weights and estimated complexity factors for the excavator utilizing the assembly complexity factor generated by the lunar rover as reference points, to either be used directly or indirectly with modifications. The reader should also notice that the environmental factor, referred to in section 6.3.2. under the column entitled "ENVIRON", can also be changed at the assembly level to reflect differences in specifications by assembly as compared to the lunar rover. The end resultant complexity factor of 6.61 is substantially different than the complexity factor of 7.02, as derived for the lunar rover, which should be expected due to the differences in mission, ruggedness and size of the respective vehicles.

For the reader's interest, this utility routine has been used to modify satellite subsystem complexities as a function of the elimination of subsystem components. For example, if the AMCM complexity factor for a communications subsystem is determined to be non-applicable to the requirement for a subsystem complexity factor without a transponder, this utility routine would be utilized to extract the cost of the transponder from the subsystem. The utility routine has also been used for complexity modification where assemblies are desired to be added to the referenced system, subsystem, assembly and sub-assembly.

7. FORMALIZATION

REQUIREMENT: "Methodology provided to each program 'agent' for a distributed cost estimate of the program or scenario pieces."

7.1. MODEL INPUTS

The Advanced Missions Cost Model input requirement is the data that is needed to generate a cost estimate. The input types are:

1) User-defined and entered input data
2) System supported data

7.1.1. USER-DEFINED AND ENTERED INPUT DATA

7.1.1.1. NEW MISSION DEFINITION DATA

The user is required to input data that defines the Advanced Mission to be estimated. Typical information required to define a new Mission includes but is not limited to:

1) Dry weight of system
2) Initial Operating Capability date of system
3) Number of Prototype units to be produced
4) Number of Production units to be produced
5) Standardized WBS number for system

The user may enter data at different levels of detail. The different levels of detail include but are not limited to:

1) Total program information differentiating between Manned/Unmanned and Recurring/Non-recurring

2) Principle program elements and phase information such as; major systems of a Lunar Base-Habitat Module, Lunar Rover, etc.

3) Subsystem information such as; major components of a Lunar Rover - Navigation, Power, etc.

7.1.2. SYSTEM SUPPORTED DATA

System supported input data is defined as that information previously entered and stored into the Advanced Missions Cost Model by model developers and/or users.

7.1.2.1. STANDARDIZED WORK BREAKDOWN STRUCTURE

A WBS consisting of approximately 150 elements, with detail to the fifth level. The concept behind the standardized WBS is given a historical program, in it's original accounting format, translate that data to fit into the standardized WBS format. This will avoid comparing two historical programs with different WBS, year dollars, weight statements, etc. The main priority of the standardized WBS is to be generic, thus giving it the ability to describe various cross cultural systems. Examples of systems described by the standardized WBS include, but are not limited to:

1) Unmanned satellite systems (Viking, ATS, HST)
2) Manned space systems (Gemini, Lunar Module)
3) Aeronautical systems (X-15)

7.1.2.2. PRODUCT CATEGORY

An equipment or services descriptor, that allows very different historical programs to be classified into a similar format. There are approximately 100 product categories.

7.1.2.3. MISSION CATEGORY IDENTIFIER CODE

A mission identifier that describes the type of mission performed. It is a three digit code with the format XYZ

where X can be, but is not limited to:
1 => Historical, Manned
2 => Historical, Unmanned
3 => Conceptual, Manned
4 => Conceptual, Unmanned

where Y can be, but is not limited to:
1 => Low Earth Orbit
2 => High Earth Orbit/ Geosynchronous Orbit
3 => Lunar
4 => Planetary

and where Z can be, but is not limited to:
1 => Feasibility Demonstration
2 => Science
3 => Applications
4 => Exploration
5 => Habitation
6 => Exploitation

7.1.2.4. ESCALATION TABLE

A NASA-provided escalation table, containing escalation factors between any two years, starting in 1959 and ending in 1999.

7.1.2.5. HISTORICAL PROGRAM DATABASES

7.1.2.5.1. ORIGINAL FORMAT

Historical program data, in original source document(s) hard-copy format, containing information of cost, programmatic and technical characteristics, to provide a traceable source of raw data. The original historical program database will be implemented by Advanced Missions Cost Model developers, and maintained to keep historical program information as dynamic as possible. The original historical program database will be the major input data source for the standardized historical program database.

7.1.2.5.2. STANDARDIZED FORMAT

Single source of standardized WBS correlated cost, programmatic and technical data to supply direct historical information to the Advanced Missions Cost Model. The standardized historical program database will be implemented by Advanced Missions Cost Model developers, and maintained to keep historical data as dynamic as possible. Typical data stored in the standardized historical program database includes, but is not limited to:

1) Dry weight of system
2) Initial operational capability of

system
3) Number of prototypes units produced
4) Number of production units produced
5) Product categories represented within that standardized WBS
6) Non-recurring, recurring and operations cost of system

7.1.2.6. NASA BUDGET DATABASE

7.1.2.6.1. HISTORICAL FORMAT

A historical NASA budget database

7.1.2.6.2. FORECAST FORMAT

A current NASA budget for new and continuing projects

7.1.2.7. DESIGNER'S CATALOG DATABASE

A database, containing technical information in tabular and graph form, from advanced mission studies commissioned by NASA, DoD, ESA, and similar agencies. The Designer's Catalog may also include detailed historical technical data.

7.1.2.8. CROSS CULTURAL ANALYSIS

The results of the off-line Cross Cultural study will be integrated into the Advanced Missions Cost Model.

7.1.2.9. BASELINE ESTIMATING RELATIONSHIPS

7.1.2.9.1. COST ESTIMATING RELATIONSHIPS

A set of cost estimating relationships will be stored in a user-accessible library. The information stored for each baseline estimating relationship may include, but is not limited to:

1) Equation form
2) Baseline equation coefficients
3) Equation predictor variables
4) Standardized WBS element associated with base-line equation

The Cost Estimating Relationship equations are grouped by cost type to be generated and system type of program. For example, baseline Cost Estimating Relationships may be grouped by:

1) Total costs for Unmanned programs
2) Recurring costs for Manned programs
3) Non-recurring costs for Manned and Unmanned programs

7.1.2.9.2. TIME ESTIMATING RELATIONSHIPS

A set of time estimating relationships that will be stored in a user-accessible library.

7.2. MODEL OUTPUTS

The Advanced Mission Cost Model output will be in several formats. The output format includes, but is not limited to:

1) Predefined tabular format
2) User-defined tabular format
3) Predefined graph format
4) User-defined graph format

7.2.1. NEW MISSION PHASE COST OVER TIME

Display predefined screens of tabular and graphical data of the calculated development, production and operations costs. The user will be allowed to produce a hard-copy of any screen, and through a series of menus and help screens, produce user-defined tabular and graphical displays.

7.2.1.1. LIFE CYCLE COST SUMMATION, SPREAD AND DISCOUNTING

7.2.1.1.1. SCHEDULE ESTIMATES

Tabular and graphical displays of estimated schedules of development, production and operations cost. The user will be allowed to produce hard-copies of any display.

7.2.1.1.2. RDT&E COST ESTIMATES

Tabular and graphical displays of estimated costs of RDT&E. The user will be allowed to produce a hard-copy of any display.

7.2.1.1.3. PRODUCTION COST ESTIMATES

Tabular and graphical displays of estimated costs of Production. The user will be allowed to produce hard-copies of any display.

7.2.1.1.4. OPERATIONS COST ESTIMATES

Tabular and graphical displays of estimated costs of Operations. The user will be allowed to produce hard-copies of any display.

7.2.1.2. COST UNCERTAINTY

Tabular and graphical displays of estimated cost versus probability of achievement. The user will be allowed to produce hard-copies of any display.

7.2.2. NEW MISSION FUNDING WEDGE

Tabular and graphical display of estimated advanced mission funding requirement over a planning horizon, as well as the funding "slice" integrated into the current NASA forecast budget. The user will be allowed to produce hard-copies of any display.

7.3. MODEL FUNCTION

7.3.1. MAINTAIN HISTORICAL PROGRAM DATA

The user, or a designated database manager, will have the ability to modify or add information stored in the historical program databases.

7.3.1.1. ORIGINAL FORMAT

Given the fact that source documentation for a given project may vary, not only in level of detail, but in accuracy as well, it will be necessary to analyze the source document(s) to find the "best" information to date. Because of this iterative processing of data, the data manager will be allowed to modify existing historical program data, or add new historical program data.

7.3.1.2. STANDARDIZED FORMAT

To obtain standardized historical program data, the Advanced Missions Cost Model developers derived a methodology to take original format historical program data and "fit" it into a standardized format. This methodology development is on-going, therefore standardizations made previously may need to be updated. Also, new original historical programs will continually be added to the database, requiring standardization of the new data and eventually of addition of new data to the standardized historical program database. Therefore, the data manager will be allowed to update existing standardized data as well as add new data to the standardized historical program database.

7.3.2. ESTIMATING RELATIONSHIPS

7.3.2.1. DEVELOPMENT OF NEW COST ESTIMATING RELATIONSHIP EQUATION

If the user does not wish to use an existing cost estimating relationship from the Advanced Missions Cost Model, the model will allow the user to generate a new estimating relationship.

7.3.2.2. RECALCULATION OF EXISTING COST ESTIMATING RELATIONSHIP

The Advanced Missions Cost Model will contain a set of baseline estimating relationships. The user will be allowed to view this set of estimating relationships, and decide whether or not one of the existing estimating relationships will fit the user's need for an estimate. If not, the Advanced Missions Cost Model will allow the user to calculate a cost estimating relationship, using one of the baseline estimating relationships and a new set of historical program data points.

7.3.3. MAINTAIN DESIGNER'S CATALOG

The Designer's Catalog Data Manager will periodically update the Designer's Catalog as new information is obtained. The Designer's Catalog Data Manager may enter data in various ways. These methods include, but are not limited to:

1) Keyboard input of text data 2) Optical Character Reader for text, graphs and technical drawings

3) Keyboard design to re-create graphs and technical drawings

4) Electronic data transfer from another computer

7.3.4. BUILD NEW MISSION DEFINITION

The Advanced Mission Cost Model user will select/input the required variables that define a new advanced mission. The Advanced Mission Cost Model will assist the user in various ways, such as:

1) List available standardized WBS numbers that may be assigned to the new mission

2) List available product category numbers that may be assigned to the new mission

3) Display analogous technical and text data from the Designer's Catalog

4) List available mission category identification codes that may be assigned to the new mission.

5) Check for valid upper and lower boundary values on inputted data.

7.3.5. ESTIMATE NEW MISSION PHASE COST OVER TIME

Given the new mission definition data provided by the user and the system supported data the Advanced Mission Cost Model will estimate Development, Production, Operations and Support cost and schedules for the new mission. The user will be allowed to perform "what ifs" on the cost/ schedule estimate, such as:

1) Incorporating the affects of the cross-cultural study

2) Modifying the new mission definition data 3) Incorporating cost uncertainty profiles

The user may store the estimate for reference and comparison to future estimates.

7.3.6. ESTIMATE AFFECT ON NASA FUNDING

Given the cost/schedule estimate for the new mission and the NASA Budget requirements the Advanced Missions Cost Model will project the funding requirements over a planning horizon and compare the available budget "slice" for various NASA agency-level ceilings.

7.3.7. MISSION SCHEDULING

The Advanced Mission Cost Model will attempt to find the "best mix" of Advanced Missions schedules. This may be accomplished by modifying new mission definition inputs such as, but not limited to:

1) slipping schedule requirements
2) modifying system requirements

The NASA forecast budget requirements may also be modified to allow larger or smaller funding "wedges".

The cost/schedule estimates can then be run again until the user is satisfied with the results.

7.3.8. ADVANCED MISSION COST MODEL SECURITY

7.3.8.1. DATA SECURITY

The Advanced Mission Cost Model will provide access security for ALL SYSTEM SUPPORTED data. This will be done by, but not limited to:

1) Password protection for historical database managers

2) Password protection on proprietary historical data.

3) Password protection on baseline estimating relationship equations

4) Password protection on Work Breakdown Structure, Product Categories, Escalation Table, and other predefined system data.

7.3.8.2. MODEL SECURITY

Level of access to the Advanced Missions Cost Model will be protected by a user-required password. The levels of access include, but are not limited to:

1) Top level - the user may only input a new mission definition and run a cost/schedule estimate

2) Analyst level - same functions as Top level in addition to; recalculating estimating relationships,...

3) *Model Manager level - same* functions as above in addition to ability to modifying system supported data.

7.4. SYSTEM SOFTWARE

Currently, the software required to run the Advanced Mission Cost Model includes, but not limited to:

1) Mainstay - a database and programming environment
2) An IBM compatible operating system (i.e.. DOS)
3) *a high level programming* language (i.e. Fortran)

8. IMPLEMENTATION

Up 'til now the philosophers have only interpreted the world in various ways. The point, though, is to change it!
- Karl Marx

REQUIREMENT: Develop a plan for changing the environment in which exploration programs are developed in order to make them more affordable.

NASA culture has changed over the years with the tightening of budget; however, more changes can be made in the future to improve the efficiency of program development. Implementing these changes may require significant change in the structure of NASA. Alternative institutional strategies may be needed to cause real change in the cost of major programs.

8.1. NASA CULTURE HAS CHANGED

Since its inception, NASA has gone through two major phases of cultural change. The first was the growth period of the early 1960's leading up to the Apollo lunar missions. The second phase was a period of retrenchment following the Apollo program as the civilian space budget was dramatically reduced. The effect of budget considerations on program approaches can be seen in Figure 8.1 on page 8-2.

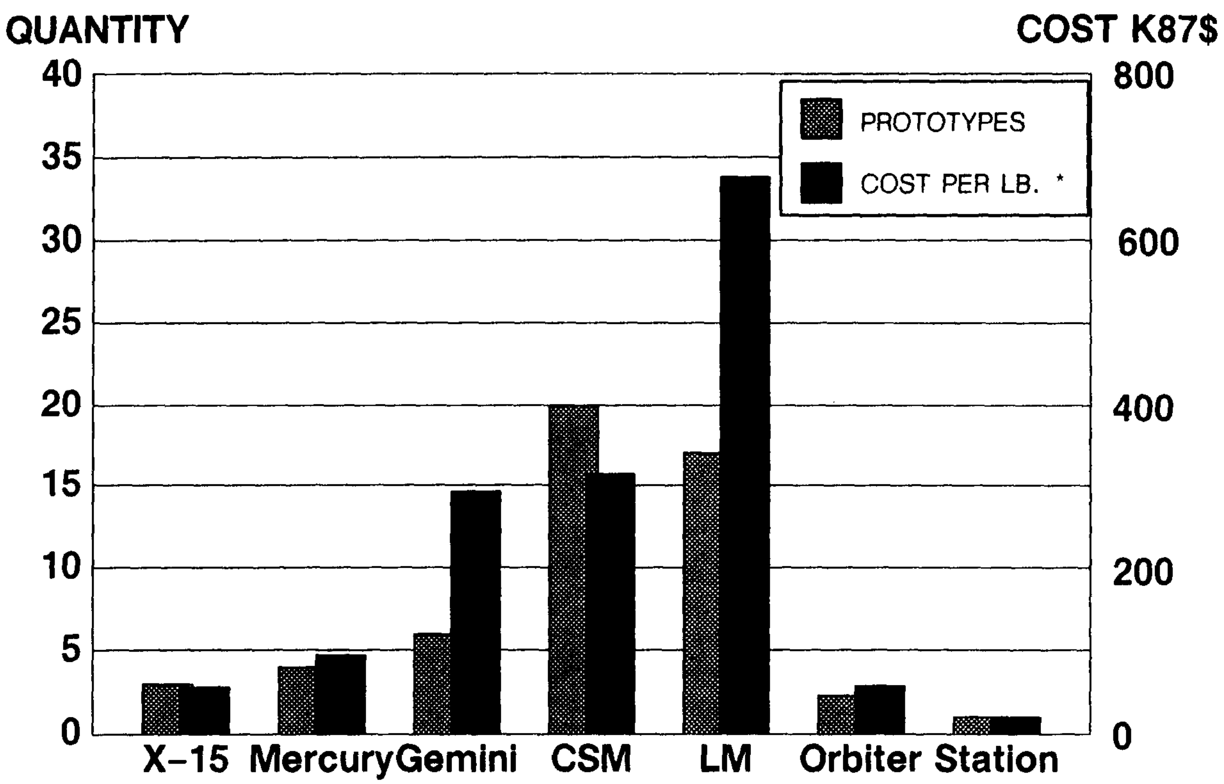
Prototyping and testing strategies, and their corresponding non-recurring cost, can be seen to grow and then decline with the shifting sands of budget policy. Reduced prototyping and testing significantly affects the cost of development. The trend towards larger spacecraft also leads to economies of scale in design and development. Computer aided engineering and design tools have improved the productivity of industry workers. Also, lessons learned from previous programs have been applied to new programs to improve management efficiency.

8.2. MORE CHANGES CAN BE MADE

The Office of Exploration is making a serious effort to minimize the cost of its programs while retaining the high level of mission success that is expected of NASA. A blue ribbon panel was recently commissioned by the President to study Department of Defense acquisition programs and there were eight characteristics the "Packard Commission" found existed in successful system acquisition programs.

In addition to the "Packard Commission" recommendations, several other recommendations are being considered. It should be noted that some of these recommendations may actually increase the front end cost of a program. These characteristics should be used as groundrules for the implementation of new civilian space initiatives. They are as follows:

8.1 NASA CULTURE HAS CHANGED



* K87$ PRIME CONTRACT NON-RECURRING COST PER POUND OF DRY MASS

8.2.1. ACQUISITION COST REALISM ALONG WITH UNIT PRODUCTION COST AS A SIGNIFICANT DESIGN REQUIREMENT.

In order to achieve program stability, it is essential that a program realistically budget for development and production costs at the start of full scale developmen. The development program manager must address the risk associated with development of the system. In this way, it is possible to determine if the program is affordable. The affordable cost level can then be used to establish a unit production cost design objective for the system which essentially becomes the "design-to-cost" objective.

8.2.2. DEVELOPING PROTOTYPES (FOR BOTH COST AND PERFORMANCE) AND EARLY, EXTENSIVE TESTING.

Prototyping has historically been done; but, its primary objective was proving that something is possible. Traditionally, the system was then almost totally redesigned during the development phase; such that the unit production cost of the system rose dramatically. The prototyping should be done with a system that is close enough to the ultimate production design such that one is able to make a good estimate of the cost and the performance of the ultimate system.

After moving out of the prototype phase, history has shown that one of the ways programs have saved money is to reduce the number of test units and the amount of test time. This is another example of short-sighted attempts to save development dollars at the expense of what ultimately becomes a stretched out and overrun program. Clearly, if you can't afford to do adequate testing early on, then the program is doomed to problems later.

8.2.3. PLANNED PRODUCT IMPROVEMENTS AND MAXIMUM USE OF PROVEN COMPONENTS AND SUBSYSTEMS (ESPECIALLY COMMERCIAL ITEMS).

It has been found that when existing systems are modified, rather than new systems started from scratch, the time and cost for development are both dramatically reduced. The concept here is not to start off a new program assuming that it requires a new set of avionics, a new engine, et cetera; but, to independently develop each of these subsystems with standardized interface specifications so they can be plugged in when they are proven and to insert these upgrades at an appropriate block point in the production cycle.

Consistent with the idea of using proven systems and subsystems is the concept of systems making maximum use of commercial subsystems and components. Both the Packard Commission and a recent Defense Science Board Task Force emphasized the dramatic benefits, particularly with electronics, that could be achieved through greater use of commercial components. The Defense Science Board study found that systems built from commercial components would have costs that were between two and eight times cheaper overall, with comparable or better reliability; and, that these systems could be acquired between two and five times more rapidly as a result of using off-the-shelf, proven, commercial components.

8.2.4. PRESENCE OF A CONTINUOUS ALTERNATIVE.

What makes a market economy operate effectively is the continuous presence of an alternative for the buyer; such that, if a supplier reduces his quality or raises his price, the buyer can go elsewhere. However, the typical program has no such alternative present. Rather, there is usually a fierce competition for the initiation of a development program and this is followed by sole-source environment throughout the many years of the development and production phases of the program.

Occasionally, the presence of continuous competition in the development and/or the production phase has been tried and the results have been very impressive. For those programs that had "dual sourcing" during the development phase of a set of Army programs, the R&D costs were better controlled; however - most importantly - the production costs were dramatically reduced as a result of the competitive development phase; thus, far more than justifying the increased development cost for the second source. Equally significant, it was found that, on the average, the performance was much higher on those programs that had been dual sourced.

8.2.5. SHORT AND STABLE SCHEDULES FOR DEVELOPMENT AND PRODUCTION.

All of the successful programs studied began by using previously demonstrated technology and by realistically estimating their program costs. They then fully funded the necessary dollars and maintained the program's initial requirements throughout the program's development. This combination - of demonstrated technology, cost realism, and stable budgets and requirements - allowed them to achieve extremely short development and production schedules. Thus, they realized maximum economic efficiency and got the new systems fully deployed in the fastest possible time.

8.2.6. EXPERIENCED, SMALL STAFFS, WITH CLEAR COMMAND CHANNELS AND LIMITED REPORTING.

In a typical US Government program, the senior managers frequently are quite inexperienced and often rotate a number of times during the development phase. Often, this inexperience is compensated for by having a relatively large staff of people, all responsible for small pieces of the overall activity. This problem is further compounded by having a very large number of layers above the program office, through whom all decisions must be passed. One estimate was that on a seven year development program, over three and a half years of the time was taken up with decision making and the rest with actual development. Finally, it is estimated that something like 20 percent of a typical DOD development program's cost is devoted to reporting on the program. By contrast, in those programs that were successfully run, the primary reports required were deviation reports in which thresholds were established for cost, schedule, and performance and, as long as the program stayed within these limits, very little reporting was required.

8.2.7. EFFECTIVE COMMUNICATION WITH USERS FOR COST/PERFORMANCE TRADEOFFS.

There is a myth that exists within the acquisition world that there is an initial "requirement" established for a system, and then this is turned over to the development community to pursue. By contrast, on a successful program, it is recognized that there must be a continuous trade-off made between the user and the developer in terms of the impact of varying requirements on development and production oosts and schedules.

8.2.8. EARLY DEVELOPMENT PHASE FUNDING FOR PRODUCTION AND SUPPORT CONSIDERATIONS.

Traditionally, the development phase of a new system focuses almost exclusively on that phase. Then, later, we find out how much it will cost to produce and maintain it. However, this is inefficient, in both time and dollars, particularly with current trends towards new, computer integrated manufacturing technologies.

If funds are available up-front to include production considerations as part of the original design job, then one can make the transition from computer-aided design through computer aided manufacturing and into computer-aided logistics in a smooth and continuous process. This requires an engineering/production/support team in the early design phases that is more than simply the lip service that has traditionally been given to this area. It requires that the design be continuously modified in order to take producibility and maintainability directly into account.

8.2.9. MASS PRODUCTION TECHNIQUES SHOULD BE USED AS MUCH AS POSSIBLE AT EVERY LEVEL OF HARDWARE (AND SOFTWARE).

The use of mass production techniques to reduce cost is so widely known that it need not be elaborated on here. This lesson has apparently been lost on NASA, however, since the agency still insists on building virtually every spacecraft from scratch. Even in relatively small quantities, mass production can save money by spreading design, prototype, test, and tooling cost. Further economies can be gained by the production learning that occurs with repetitive tasks. Although some small loss in performance may be the price of commonalty, this could be offset by the increased confidence in performance from a proven design.

8.2.10. TECHNOLOGY SHOULD BE PUSHED FORWARD ONLY AT REASONABLE RATES AS DETERMINED BY THE RECOGNIZED TECHNOLOGY MANAGER.

Part of the NASA mission is to develop new technology; but, when a program depends on the successful development of a technology that cannot realistically be accomplished, the result is cost overruns and schedule delays. In order to minimize program risk, programs should depend on technology that is already demonstrated, or that can be proven well within the required schedule.

8.2.11. MINIMIZE FUNCTIONAL COMPLEXITY OF INDIVIDUAL HARDWARE ELEMENTS.

8.2.12. DRIVE FOR COMMONALITY AMONG HARDWARE ELEMENTS.

8.2.13. DESIGN HARDWARE ELEMENTS WITH SUBSTANTIAL PERFORMANCE MARGINS.

8.3. IMPLEMENTING CHANGES

Any attempt to make major reductions in the cost of future space programs will require major cultural change in the NASA institution. Any cultural change, whether it is good or bad, will be perceived as a threat and will meet major resistance.

Any NASA cultural change will either be very slow or will require a forceful, dramatic intervention. A slow cultural change will not produce results soon enough to influence exploration planning and coercive methods of change are not likely to work; therefore, institutional strategies, other than changing NASA from within, must be examined.

8.4. ALTERNATIVE INSTITUTIONAL STRATEGIES

The number of institutional options that could be utilized for space exploration is virtually unlimited. The following list is by no means comprehensive; but, is intended to represent a full spectrum of possible options.

8.4.1. **Retain the current NASA institutional structure. Implement NASA cultural changes within existing institution.**

8.4.2. **Restructure the NASA organization.**

8.4.3. **Spin off operational programs from NASA. Focus NASA on research and development.**

8.4.4. **Establish a new organization with alternative financing methods.**

8.4.5. **Create an incentive-subsidy for private sector space exploration. NASA develops technology only.**

9. NEAR TERM COST ESTIMATES

REQUIREMENT: Provide an interim capability to do relative cost and schedule estimates for exploration case studies beginning in FY 1989.

9.1. RECOMMENDATIONS

In order to meet the requirements for performing cost estimates in fiscal year 1989, several optional approaches have been assessed. The first option is to accelerate development of the Advanced Missions Cost Model currently under development by a contract with ECON. A second option would be to develop an interim cost model based on, but much simpler than the ECON model. The third option is to use existing cost models.

Based on an evaluation of costing needs during the next fiscal year (FY89), the following recommendations are made.

RECOMMENDATION: Use existing cost models and implement with local cost personnel at integration agent centers for early FY89 needs.

RECOMMENDATION: Accelerate application of funding to ECON model in order to support Cycle 2+ (FY 1989) cost estimating.

RECOMMENDATION: Implement a standardized nomenclature for hardware elements along the lines of the Air Force equipment designation scheme (e.g., AIM-9P)

RECOMMENDATION: Implement an automated, centralized data base for scenario and hardware element data ASAP.

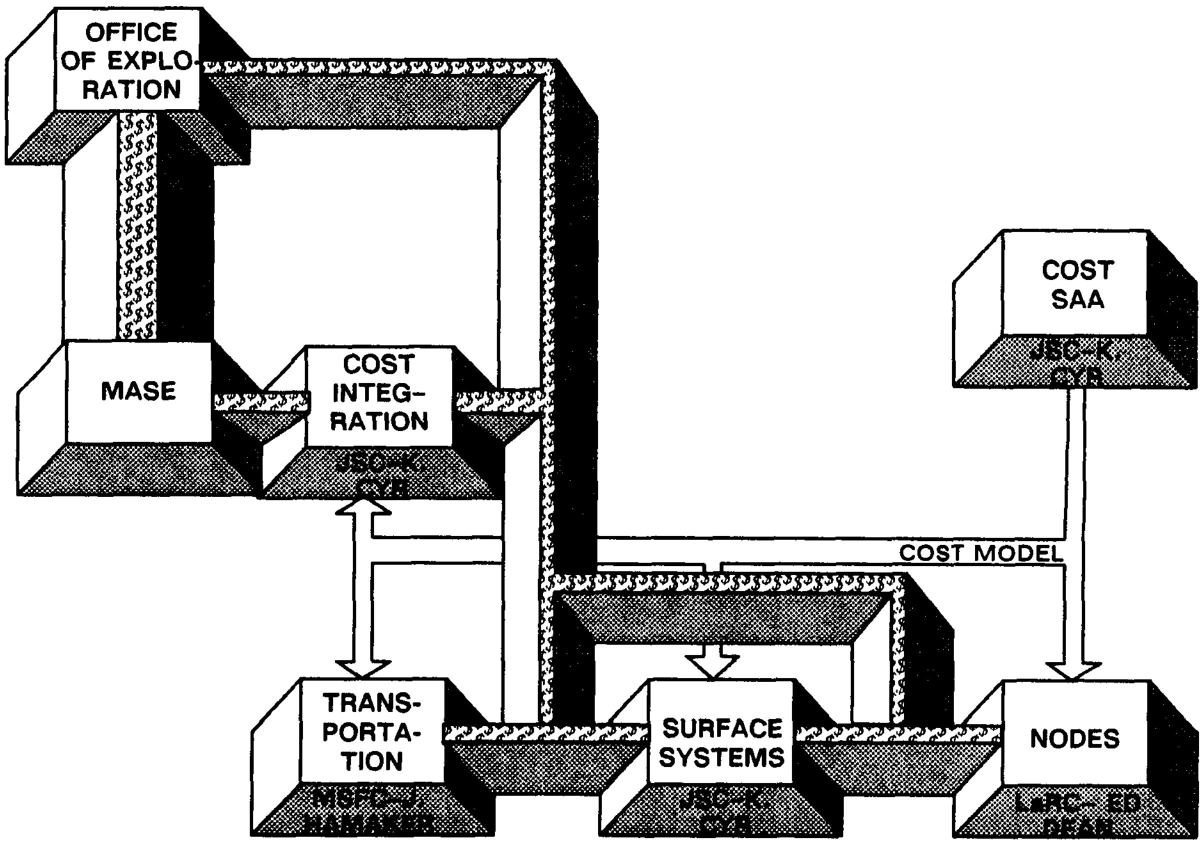
9.2. COST ORGANIZATION

An informal cost organization has been set up for the cost task. The function of this organization is to guide the development of cost models, provide expert cost knowledge, and implement the resulting methods. In order to provide cost expertise in all the areas needed and to make maximum use of existing resources, the cost organization is distributed among several NASA Centers. The distribution of cost functions corresponds to the allocation of technical functions among the Centers by the Office of Exploration.

9.2.1. SENIOR ADVISORS FOR COST UNDERSTANDING AND MODELING

The Senior Advisors for Cost Understanding and Modeling (SACUM) group is made up of senior cost analysts and managers from each of the NASA centers. The purpose of this group is to provide oversight and guidance to the cost activities. The SACUM will meet to establish guidelines for cost estimating, preside over Advanced Mission Cost Model progress reviews, and direct an activity to develop affordability approaches for exploration programs. The SACUM reports directly to the Associate Administrator for Exploration.

9.1 COST ORGANIZATION



9.2.2. COST UNDERSTANDING SPECIAL ASSESSMENT AGENT

The Johnson Space Center (JSC) has technical responsibility for mission analysis and synthesis. This is the primary integration function. As such, the overall responsibility for cost understanding is collocated at JSC under the Special Assessment Agent for Cost Understanding. Organizationally, this function is within the Cost Analysis Office of the Space Station Project Control Office. This is a matrix organization from the Administrative Directorate.

The cost understanding function is reportable directly to the Office of Exploration as a Special Assessment Agent (SAA). Interfaces with the Integration Agents and the Center cost leads are also required to ensure that model development meets the needs of the performing organizations. The SAA reports to the Associate Administrator for Exploration at the Code Z Progress Review on a monthly basis. The SAA also submits input to the Office of Exploration Annual Report.

The SAA serves as Technical Monitor for the Advanced Missions Cost Model Contract (AMCM). The AMCM contractor reports formally to the SAA and the SACUM on a quarterly basis. The AMCM is primarily responsible for cost model and data base development.

The cost understanding task also requires the acquisition of historical cost data on various programs. Data will be obtained from the Redstar data base at the Marshal Space Flight Center under a contract with Planning Research Corporation of Huntsville.

9.2.3. MISSION ANALYSIS AND SYNTHESIS COSTING

The JSC cost organization is also responsible for supporting the integrated cost estimating function. This is a function of the Mission Analysis and Synthesis Agent (MASE) which is in the Lunar Mars Exploration Office at JSC. The cost integration activity includes defining data base requirements, establishing cost guidelines, performing trade studies, and integrating cost estimates from the Integration Agents.

9.2.4. PLANETARY SURFACE SYSTEMS COSTING

The final cost responsibility at JSC is to support cost estimates by the Planetary Surface Systems Integration Agent. This will include developing special costing expert knowledge for surface systems, collecting cost input data, performing cost estimates and trade studies, and presenting results to higher management levels.

9.2.5. SPACE TRANSFER VEHICLE COSTING

The Marshal Space Flight Center has the Integration Agent responsibility for space transfer vehicles. Cost estimating for these systems will be the responsibility of the Engineering Cost Group in the Program Planning Office of Program Development. The center cost agent will be responsible for collecting cost input data, performing cost estimates and trade studies, and presenting results to higher management levels for space transfer systems.

9.2.6. ORBITAL NODE COSTING

The Langley Research Center has the Integration Agent responsibility for orbital nodes. Cost estimating for these systems will be the responsibility of the Cost Estimating Office in the Systems Engineering Division of the Systems Engineering and Operations Directorate. The center cost agent will be responsible for collecting cost input data, performing cost estimates and trade studies, and presenting results to higher management levels for space nodes.

Workshop Report:

Lunar Base Precursor Strategies

A Workshop Held on 8 April 1988

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Compiler's Note

This report reflects as closely as possible the proceedings and products of the Workshop on Lunar Base Precursor Strategies, held in Houston on 8 April 1988. A working version was produced from notes written furiously during exchanges that were maintained at higher pace for a longer period of time than some might believe possible. To minimize oversights and errors, that version was sent to all of the attendees for their inputs. (Fourteen of the nineteen participants responded within 11 days of the mailing, which may be taken as an indication of the interest in the subject as well as of the importance which the participants view the subject of lunar bases.) Those comments and corrections which served to clarify a point or correct a statement were included when appropriate. Those which might have changed the meaning, intent, or spirit of a conclusion or statement made by the participants as a group, however, were not included in this final version. Obviously, a half-day time limit for such a workshop places stringent requirements on efficiency and expediency. When the attendees had time to reflect on the report and the table contained therein, most of them found items which they wished had been stated otherwise -- if at all. This is only natural, and it is hoped that those who might use this report will understand that it represents an initial effort made under an imposing timeline.

That having been said, I would like to thank John Frassanito and his associates in the Beta Building for their gracious efforts in hosting the workshop, and the participants for their time and cooperation in taking a very good initial crack at a very complex problem.

Any errors or misrepresentations contained in this document, unintentional though they may be, remain my responsibility.

Mark J. Cintala
Johnson Space Center

WORKSHOP REPORT
Lunar Base Precursor Strategies
8 April 1988

A half-day workshop was held in order to begin the process of addressing the data requirements for establishment of a base on the lunar surface. The meeting took place in the Beta Building in Houston, just outside the Johnson Space Center. The twenty workshop participants represented a wide variety of disciplines, and are listed in Appendix A.

This workshop was requested by M.B. Duke, Director for Science in the Office of Exploration (Code Z) at NASA Headquarters. He began the proceedings with a presentation describing the need for such a meeting, the scenarios being considered by Code Z, and the types of questions requiring answers (Appendix B). The Director of the Office of Exploration has requested a definition of precursor missions that would be necessary in preparation for the establishment of inhabited lunar and martian bases; such missions would provide information necessary for the location, emplacement, design, and construction of those bases, among other things. The objective of this workshop was not to design those precursor lunar missions, but to define requirements for those which would provide essential data in support of the lunar-base initiative. The scientific data acquired by such precursor missions would be intended primarily to support the operational aspects of lunar-base activities, and secondarily to enhance scientific knowledge of the Moon and its environs. In this light, data analysis could be considered, but the development of technology as such would be beyond the purview of these precursor missions. The "bottom line" objective of the workshop was summed up in the question: *"What should we learn before the first lunar base is established?"*

As discussions began at a very vigorous pace, it was obvious that this topic touched upon subjects that were near to the hearts of all of the participants. Perhaps the most pervasive questions that arose dealt with the lunar base itself:

- o What will be its purpose, and what will be the information required by those who will design and build it? In other words, what are the engineering requirements?
- o Will it be permanent or temporary?
- o If temporary, would the base itself then be considered a "precursor mission" if it were used to obtain the requisite data for use in establishing a permanent base?

With these questions in the forefront, the initial round of discussions was directed more at the utility of, justification for, and characteristics of the base itself. The participants were reminded that the utilization of the base -- not whether it will be temporary or permanent -- will be the driver for the precursor missions.

The question of engineering requirements will be determined by the purpose of the base, which smacks of truism, but is nevertheless an important item to keep in mind. The initial base will be relatively modest, but complex:

(1) Reasonable mobility will be required/available.

(2) Subsurface operations will be likely in order to emplace habitation modules, etc.

(3) Instrumentation will be necessary at and probably below the surface.

These requirements were considered inadequate by some of the participants if decisions were to be made regarding methods of data acquisition. Another phase of the discussion then followed, in which an attempt was made to determine whether two sets of requirements could be distinguished: one set that would be independent of the objectives of the base, and another that would be driven *by* its projected goals. During this discussion, two different philosophies became apparent. In short, they can be summarized by the following:

- o Try to be anticipate the fiscal climate (which will likely be less than ideal) and minimize costs by choosing an Apollo landing site, for which we already have good data, as the location for the first base. In this vein, the Apollo 17 site was invoked with some regularity, principally because of its known, relatively high concentrations of titanium-bearing materials.

- o Conduct a careful, global evaluation of the Moon and its environment in order to expand our information base in terms of scientific and resource options, thus increasing the chances of establishing an optimum site for the base.

This was judged to be an important difference in approach, since the question of acceptability *vs.* optimization is strongly related to the timing and, consequently, to the overall cost of the of the project. In defending the Apollo-site option, it was pointed out that the funds for anything more ambitious might not even exist in the foreseeable future, and that such a site would be the best chance for a base. The optimum-site school suggested that long-range costs might well be saved by a global survey which might locate, for example, concentrations of valuable resources. It was agreed, however, that even a return to an Apollo site would require more local information.

An attempt was then made by the group to construct a table of information, centering on the types of data which would be required -- whether such data exist or not -- before lunar-base design and construction could begin in earnest (Table 1.). Because the requirements for data resolution, the type of measurement, and the purpose for acquiring the data will be different depending on the scale of the features to be examined, the data types were divided into three separate groups on the basis of scale: *global*, *regional* (on the order of a few hundred kilometers in radius about the point of interest), and *local* (10-km radius). Key words or phrases are the entries in the table, and explanations are given following the table.

A number of separate items were suggested and discussed. Among them were the following:

- o There are fundamental pieces of information that will be necessary regardless of the role of the base. Examples of these include:

  - o Can a spacecraft land at a given site? Will the hazards -- such as boulder density at the surface, bearing strength of the soil, and topographic hazards during the approach -- permit it?

- o Will the regolith be thick enough for the various uses in which it will be employed at and around the base?

- o In reference to potential large-scale "earth"-moving requirements, what is the relation between the abundance of buried boulders and their abundance on the surface? It was suggested that a small bulldozer-like vehicle be part of the precursor strategy, with the objective of digging trenches into the regolith. A detailed study of the regolith to a significant and substantial depth would then be possible. The results would be extremely valuable for the engineers charged with construction of the base.

- o Accurate topographic maps will be necessary at least for candidate areas, as will remote-sensing information in order to locate and map potential resources. The quantity and resolution of each will be determined by the type of site desired for the base (*vis à vis* the Apollo *vs.* the optimum site).

- o The cartographic control-net for the Moon is sadly lacking, particularly on the far side. A considerable improvement will be necessary should farside operations be considered seriously.

- o Deep drilling will likely be one of the most important activities from both scientific and engineering standpoints. How much lead time will be necessary to develop this and other comparable technological advances before exploration for a site and emplacement of the base begin?

- o A compelling reason for placing astronomical facilities on the Moon is the stability of the surface from a seismic standpoint. This stability is not well known quantitatively for most of the Moon, and would be a very desirable -- if not necessary -- parameter to know in advance of site selection for such observatories.

The participants adjourned following completion of the table, which represented the bulk of the time spent in the workshop. Although it was agreed at the beginning of the proceedings that a date for a follow-on meeting would be set at the end of the morning's activities, it was felt generally that some time would be needed to contemplate this report and ponder the next step.

Table 1. Data requirements for activities related to the definition, design, and emplacement of a lunar base. Three scales of study are listed: *Global*, which refers to phenomena or features that are greater than roughly a thousand kilometers in size; *Regional*, which denotes features or areas that are a few hundred kilometers in radius; and *Local*, which indicates sites of interest with radii of 10 kilometers or less. *Minimum* in the *Precursor Data Needs* column refers to an Apollo site (such as Apollo 17), while *Preferred* represents some potential site for the lunar base which has not yet been visited. An "x" in either of these columns indicates that available data are probably sufficient to meet that particular need.

	Global	Regional	Local	Precursor Data Needs	
				Minimum	Preferred
Topography				x	1-2 m
Gravity	Orbital systems	Mascons, Internal structure	Geophysical Boundary Conditions, Prospecting	x	"LGO"
Morphology			Boulders, Mobility	Site: 0.5 m	Global: 100 m
Composition	500 m digital base	500 m digital base	50 m digital base	x	500 m (Resource assessment), Sample return
Magnetics	Field strength and direction		Field strength and direction	x	x
Subsurface Structure	Lava tubes	Layering: 1 km depth, Regolith thickness	Layering: 100 m depth, Grain-size distribution, Boulder abundance, Regolith thickness	Geophysics, Trench	Geophysics, Trench, Search for lava tubes
Soil Properties	Maturity	Maturity	Maturity, Electrical, Thermal, Dust Assessment	x	Maturity
Atmosphere	Composition, Density, Dynamics		Outgassing	x	x
Volatiles	Reconnaissance		Measurement	x	x
Cartographic Control	100 m	Surface navigation	Navigation	x	100 m
Seismicity	Measurement	Monitoring	Monitoring	x	x
Micrometeoroids	Flux, Velocity distribution		Defense	x	x
Radiation	Warning		Warning, Shielding	x	x
Water	Reconnaissance		Accessibility, Prospecting	x	Assessment
Soil Mechanics		Excavatability, Bearing strength	Slope stability, Adsorptivity, Permeability, Trafficability	Trench	Trench

Notes and Explanations

The following is a brief explanation of the entries in Table 1, organized in order of appearance in the table.

Topography

Preferred -- Knowledge of the topography at the site of the lunar base would be necessary to an accuracy of at least 1-2 meters principally for engineering purposes.

Gravity

Global -- Knowledge of variations in the Moon's gravity field will be necessary to permit repeatable, high-accuracy landings. Note that high-quality gravity observations of the far side will require a tracking network, likely by satellite, on the far side of the Moon.

Regional -- The majority of the global gravity variations are caused by regional mascons, which are probably related to internal structures usually associated with the larger impact basins. Depending on the location of the base, irregularities in the gravity fields generated by the mascons on a smaller scale could also be important.

Local -- Gravity anomalies on a local level could be due to mineral (or ore) deposits. Boundary conditions can also be placed on geophysical measurements with high-resolution gravity data.

Preferred -- The group suggests strongly that the Lunar Geoscience Observer in whatever manifestation it is flown ("LGO") provide the capability to obtain high-quality data on the Moon's gravity field, at least on a regional scale.

Morphology

Local -- Knowledge of small-scale surface features at the potential site of a base will be required in order to determine hazards to landing, trafficability of surface vehicles, etc.

Minimum -- The candidate site must be photographed at a resolution of at least 0.5 meters to permit unambiguous identification of potential hazards.

Preferred -- Global photographic coverage at a resolution of at least 100 meters would be advantageous in selecting potential sites for the lunar base, and in closing the selenodetic net to 1-km accuracy. Once a site were selected, very high-resolution imagery would be required. (See *Minimum* above.)

Composition

Global -- The prospect for success of a search for potential resources will be enhanced greatly by global mapping spectrometer coverage at a resolution of at least 500 m, supported by γ- and X-ray spectrometer coverage at somewhat lower resolutions. Features smaller than that, while interesting scientifically, would likely not be economically attractive.

Regional -- See *Global* above.

Local -- Compositional data at higher resolution will be necessary for establishing the layout of the base.

Preferred -- Once the moderate-resolution data were available, sample collection would be undertaken to evaluate the economic suitability of the site in question.

Magnetics

Global -- This requirement is similar to that for the local gravity data, in that both geophysical modeling and potential resource identification would profit.

Local -- Communications, astronomical observations, and other electronic/radio activities could be affected by local magnetic fields.

Subsurface Structure

Global -- Suggestions to employ lava tubes as storage volumes or actual sites for the base have been made. Identification of such features would then be necessary.

Regional -- Narrowing down the possible sites will require information on the depth to the bedrock and the overall thickness of the regolith in a given region.

Local -- The availability of regolith for mining, insulation, etc., will be a prime consideration for selecting a site. Other factors such as the distribution of buried boulders will also be important in engineering operations. As the subsurface structure of the local area would be known at a lower resolution to a depth of 1 km from the regional survey, this would constitute a study of the local area at higher resolution.

Minimum -- Much detailed information could be extracted from the regolith by digging a trench to a depth of at least a few meters or, ideally, to the regolith-"bedrock" interface. Geophysical measurements, such as high-resolution gravity surveys and electromagnetic soundings, would complement the trench data.

Preferred -- In addition to the trench and geophysical data, the engineering requirements might dictate the existence of a lava tube for subsurface uses.

Soil Properties

The maturity of a lunar soil is correlated with other factors, such as regolith thickness, free-iron content, abundance of implanted gases, etc. It thus is a principal parameter to be measured at all scales.

Local -- Characterization of the regolith at the site of the base will be of prime importance in determining its insulating characteristics (*e.g.*,, dielectric constant and conductivity), electrical properties (for over-the-horizon communications, radio astronomy, etc.), fraction of very fine-grained component (dust hazards), and other qualities of extreme relevance to planning the base.

Atmosphere

A concern voiced by many at the workshop was the certainty that human activity associated with the establishment and activities at a lunar base would produce gaseous species that would virtually overwhelm the indigenous lunar atmosphere. It is strongly suggested that a sample of the pristine (or nearly so) lunar atmosphere be preserved -- either on the Moon or elsewhere -- as a "posterity sample", and that an effort be made to characterize the pre-base atmosphere as completely as possible before it is contaminated.

Global -- The composition, density, and dynamics (time-variable qualities) of the meager lunar atmosphere should be quantified. Not only will the extensive activities required to establish the base change the existing atmosphere drastically, but some gaseous species could interfere with astronomical observations.

Local -- One of the potential time-variable influences on the lunar atmosphere would be outgassing from the subsurface. Such outgassing, were it to occur near certain types of astronomical observatories, could be disastrous from a scientific standpoint.

Volatiles

Global -- Trapped volatiles, were they to exist in sufficient quantities, could be of extreme use at the site. Discovery of such "deposits" would influence decision-making in determining a base location.

Local -- An assessment of volatile resources in the vicinity of the base would be required in order to exploit them, should they exist.

Cartographic Control

Global -- An accurate control net is mandatory for the whole Moon if activities on the farside or parts of the nearside not under Apollo groundtracks are to be contemplated. Because of insufficient photography at the requisite levels of accuracy, the current control net (used in producing maps) is inadequate at best. A Moon-wide control net with 1-km accuracy will require careful photography at ~100-m resolution.

Regional -- Navigation on the surface during extended traverses will only be as accurate as the maps used by the planners and crews.

Local -- High-resolution mapping will be necessary for planning the base as well as for navigating in its vicinity.

Preferred -- Accuracy to within 100 meters relative to the regional setting will be necessary at the site of the base for both planning and navigational purposes, as well as for the interpretation of high-resolution gravity data.

<u>Seismicity</u>

Global -- Measurement of the Moon's seismicity would be a requirement for locating astronomical observatories, physics experiments (*e.g.*, relativity research, optical investigations, etc.), and other efforts requiring stable platforms. Although measured moonquakes have been weak by terrestrial standards, the chosen site of a lunar base would likely be removed from any center of seismic activity.

Regional -- Seismic activity must be monitored for both predictive purposes (in terms of hazards and periodicity which might affect experiments) and to correct any measurements that might be taken during such activity.

Local -- See *Regional* above.

<u>Micrometeoroids</u>

Global -- There are large-scale processes which can result in concentrations of meteoroid impacts in certain parts of the Moon relative to others, particularly in the leading edge of the Moon as it moves through space. Knowledge of the flux and velocity distribution of such particles would be valuable in assessing any hazards related to impacts.

Local -- Knowledge of the flux and velocity distribution of meteoroids at the site of the base would permit design of an appropriate defense against such hazards.

<u>Radiation</u>

Global -- A warning system on a global scale would be necessary to alert all personnel on and above the surface of impending radiation danger.

Local -- The warning system would be complemented by sufficient shielding, determined by earlier studies of the regolith and other materials as potential shielding candidates.

Water

Global -- A search for water ice or hydrated minerals, if successful, could be the most valuable study undertaken as a precursor venture. The very possibility of the existence of water on the Moon in significant quanities and concentrations should mandate such a search.

Local -- Should "water deposits" be identified in any form, the base would almost certainly be built at or very near their location. Their accessibility and a determination of the extent of the deposits would be required for planning the base.

Preferred -- An evaluation of the existence of water at potential sites would be one of the mandatory activities in supporting the decision-making process.

Soil Mechanics

Regional -- Resource extraction from the regolith could extend to some distance from the base itself. With this in mind, it would be advantageous to characterize the regolith in such terms as its bearing strength, angle of internal friction, ability to be excavated, etc.

Local -- A complete mechanical description of the regolith (or of the different regoliths) in the vicinity of the base would complement the characterization of the other properties described above. Included would be factors such as permeability, adsorptivity, slope stability, trafficability, excavatability, density, and porosity. Much of this information could be collected by digging and studying a trench, as described earlier.

APPENDIX A

Lunar-Base Precursor Strategy Workshop
8 April 1988

Attendees

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APPENDIX B

Introduction and Charge to the Workshop

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PRECURSOR SCIENCE MISSIONS AND STUDIES REQUIRED FOR A LUNAR BASE PROGRAM

April 8, 1988

PRECURSOR SCIENCE MISSIONS AND STUDIES

REQUIRED FOR A LUNAR BASE PROGRAM

Approach:

o Many of us have thought about the problem for a long time.

o Time at this workshop is short.

o In order to shorten the process, a draft set of requirements has been prepared.

o We would like the workshop to:

a. Critique and improve the draft

b. Suggest additional requirements

c. Suggest alternative modes of presentation

d. Take ownership, or concur in, the final draft

PRECURSOR SCIENCE MISSIONS AND STUDIES

REQUIRED FOR A LUNAR BASE PROGRAM

ASSUMPTION: The U.S. plans to establish a base on the Moon, with the first element to be emplaced in 2001.

QUESTION: What are the requirements for information and experience that can be satisfied by precursor science missions and studies?

PRECURSOR SCIENCE MISSIONS AND STUDIES

REQUIRED FOR A LUNAR BASE PROGRAM

LUNAR BASE STRATEGY TRADE SPACE

BASE UTILIZATION	TEMPORARY	PERMANENT
RELATION TO GELOLGICAL FEATURES	IMPORTANT	UNIMPORTANT
ACCESS TO RESOURCES	IMPORTANT	UNIMPORTANT
SPECIAL SITE REQMTS.	ABSENT	PRESENT
CONSTRUCTION REQUIREMENTS	SURFACE	SUBSURFACE
OTHERS ???	???	???

PRECURSOR SCIENCE MISSIONS AND STUDIES

REQUIRED FOR A LUNAR BASE PROGRAM

DEFINITION OF TRADE SPACE

BASE UTILIZATION

TEMPORARY: Base serves as a basecamp, temporarily occupied for science or resource extraction. Limited buildup of base infrastructure. Repeated landings at the site are required.

PERMANENT: Base will evolve by addition of equipment from Earth or from internal expansion capability. Permanently occupied.

RELATION TO GEOLOGICAL FEATURES

IMPORTANT: Utilization of the base is dependent on access to particular geological features, for example, regions of particular geochemistry, or special associations of geological units.

UNIMPORTANT: Base objectives are not constrained by geology. A base which focuses on long term human habitation or radioastronomy may fit in this category.

PRECURSOR SCIENCE MISSIONS AND STUDIES

REQUIRED FOR A LUNAR BASE PROGRAM

DEFINITION OF TRADE SPACE(cont.)

ACCESS TO RESOURCES

IMPORTANT: Resource extraction is an objective of the base and effectiveness depends on magnitude and type of resources available (eg. polar water, concentrations of other elements)

UNIMPORTANT: Resource extraction is not an objective, or can be accomplished without detailed knowledge of special resources.

SPECIAL SITE REQUIREMENTS

PRESENT: Base installation or specific experiments impose special requirements for parameters such as flatness, absence of boulders, depth of regolith, etc.

ABSENT: No special site characteristics are required.

CONSTRUCTION REQUIREMENTS

SURFACE: Facilities will be placed on the surface. Radiation shielding will not be required or will be provided by covering surface facilities with regolith.

SUBSURFACE: Facilities will be emplaced below the lunar surface, eg. by trenching or tunneling.

PRECURSOR SCIENCE MISSIONS AND STUDIES

REQUIRED FOR A LUNAR BASE PROGRAM

DECISIONS WHERE PRECURSOR INFORMATION IS APPLICABLE:

SITE SELECTION - Engineering data (safety, trafficability, surface soil mechanics, etc.)

SITE SELECTION - Science objectives

SITE SELECTION - Resource Availability

BASE CONSTRUCTION - Engineering data

BASE OPERATIONS - Engineering data

PRECURSOR SCIENCE MISSIONS AND STUDIES

REQUIRED FOR A LUNAR BASE PROGRAM

INITIAL EVALUATION OF REQUIREMENTS

PREVIOUS LANDING SITES

Apollo visited six sites and rather thoroughly investigated three. For a surface-emplaced base with science requirements that can be met at one of the previously visited sites, or a base that is independent of geological or resource constraints, establishment at an Apollo landing site can be done with no new information. If subsurface construction is required, detailed subsurface knowledge of the chosen site is required.

IMAGING

Some regions of the Moon which may be desirable sites for lunar bases have not been imaged to the detail necessary to support base location and safe installation. Particularly at polar longitudes, there are vast areas with unsatisfactory resolution and/or sun angle. For currently unmapped regions, resolution at the 30 meter level is required.

Additional information at a resolution of about 1 meter is needed for engineering and science reasons for any base that is to be established other than at an Apollo landing site. This information is needed for landing safety and base layout considerations.

RESOURCES

It is unknown whether special concentrations of any element, other than aluminum and titanium, exist in mineable quantities. This includes the potential of water in polar cold traps. It is required that a global map of available resources be available for any base that seeks to optimize production from indigenous resources.

It is possible that a lunar resource mapping mission may discover previously unknown resources. If such information is revealed, and the resource is a factor in base siting, additional data will be required to establish the extent and quality of the resource. This may include both insitu investigations and the return of sample materials for analysis.

Processes that can effectively extract useable materials from lunar resources must be developed. These include mining, beneficiation, extraction, and manufacturing from natural materials. These may be primarily technolgical in their scope; however, processing techniques derived from the lunar equivalent of the study of microgravity materials processing can have a large effect on the choice of scenarios and products to be sought at the lunar base. A suite of potential processes must be developed.

SURFACE PHYSICAL PROPERTIES

For any base that must be constructed partly or wholly underground, more must be known about the subsurface boulder distribution and regolith physical properties than is known for any place on the Moon, including Apollo landing sites. Properties such as density, bearing strength, and cohesion are required to support any structure that places significant forces on the surface in the vicinity of recent impact craters. Depth to bedrock may be required for certain classes of experimental facilities, such as construction of the most sensitive seismic monitoring stations. These data are required in the immediate vicinity of the projected base site.

It is unlikely that trafficability, i.e. the distribution of meter-sized boulders or areas of soft regolith will be an issue for any long term lunar base. However, in special cases, local site surveys may be required for emplacement of special equipment or optimum placement of base elements.

For bases that are complex, information is required on the soil mechanics of insitu materials that can allow the definition of appropriate surface preparation procedures (smoothing, compaction, etc.) in association with base construction.

For bases that are temporary, do not require complex construction, and can be emplaced at the surface, it is possible that construction can be undertaken with no additional surface physical property information, or that such information can be gathered by the first human landing mission.

ENGINEERING PRECURSOR TESTS

Apollo landed safely six times on the Moon, one of which times was targeted to the location of a previous Surveyor spacecraft. It is anticipated that the emplacement of surface navigation aids will significantly simplify the problem of repeated landings at the same lunar site. Demonstrations of automated landing, including hazard avoidance, will be necessary to allow unmanned lunar landings of cargo, propellant, etc. at the base. Automated rendezvous in lunar orbit should also be demonstrated as a prelude to an operational surface to lunar orbit capability.

GEOLOGICAL AND GEOPHYSICAL FIELD INVESTIGATIONS FROM A LUNAR BASE AT MARE SMYTHII

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ABSTRACT

Mare Smythii, located on the equator and east limb of the Moon, has a great variety of scientific and economic uses as the site for a permanent lunar base. Here a complex could be established that would combine the advantages of a near side base (for ease of communications with Earth and normal operations) with those of a far side base (for shielding a radio astronomical observatory from the electromagnetic noise of Earth). The Mare Smythii region displays virtually the entire known range of geological processes and materials found on the Moon; from this site, a series of field traverses and investigations could be conducted that would provide data on and answers to fundamental questions in lunar geoscience. This endowment of geological materials also makes the Smythii region attractive for the mining of resources for use both on the Moon and in Earth-Moon space. We suggest that the main base complex be located at 0, 90 E, within the mare basalts of the Smythii basin; two additional outposts would be required, one at 0, 81 E to maintain constant communications with Earth, and the other, at 0, 101 E on the lunar far side, to serve as a radio astronomical observatory. The bulk of lunar surface activities could be conducted by robotic teleoperations under the direct control of the human inhabitants of the base.

INTRODUCTION

Several advance planning studies are currently underway to identify strategies for the establishment of a permanent base on the Moon (Mendell, 1985). Depending upon the ultimate emphasis placed on lunar base operations, several considerations enter into the planning process, one of which includes the selection of the base site. Any lunar base site will offer something for various users. Duke et al. (1985) identified three separate scenarios for development of a lunar base, each having a different emphasis for ultimate base use: lunar science, resource utilization, and lunar settlement. These different thrusts are not mutually exclusive, but each could have slightly different criteria for base site selection. In fact, it is highly probable that a lunar base program will have elements of each emphasis; indeed, one of the attractions of a lunar base program is that it has so much to offer to many different users.

Although it may be premature at this stage to design detailed, site-dependent operational strategies, it is not too soon to begin considering what types of lunar base sites offer the most benefits to the most potential users. In this spirit, we here present a study of the Mare Smythii region, on the east limb of the Moon, and suggest that this location presents many advantages to all the currently identified potential base users.

ADVANTAGES OF A BASE SITE ON THE LUNAR LIMB

A consequence of the Moon's synchronous periods of rotation and revolution is that the Earth is always visible at the same location in the sky on the near side and always invisible from the far side. This presents both opportunities and problems; for normal lunar base operations, it may be desirable to maintain constant communication with the Earth, a condition satisfied by any near side site. However, one of the prime advantages of the Moon as an astronomical observing platform is that the lunar far side is the only known place in the Solar System that is permanently shielded from the extensive radio noise produced by our home planet. These two requirements are mutually incompatible, short of designing and operating two separate lunar base sites.

Because the Moon orbits the Earth in an elliptical path and the plane of the lunar orbit is not quite perpendicular to its rotation axis, the Moon slightly wobbles, or librates, in both latitude and longitude. Thus, the lunar limb (the great circle defined by the poles and the 90 degree meridians) is the only place on the Moon where the Earth is sometimes visible and sometimes occulted. It is in this region that a base could be established that may potentially satisfy both paradoxical requirements: that of radio access to the Earth and shielding from the Earth's radio noise. We emphasize at the outset that no single site accomplishes these goals at all times, but rather, several outposts or "sub-bases" in close proximity are required to make use of the lunar libration effect.

Several studies have advocated base sites at the lunar poles (e.g., Burke, 1985) either because of the availability of continuous solar power or because the continuous darkness of crater floors may have trapped volatiles (including water) over geologic time (Arnold, 1979). However, from an astronomical viewpoint, a major drawback to a polar site is that only half of the sky is ever visible. Moreover, the unique lighting conditions of the poles, where the sun is constantly near the horizon and the surface is either jet black or blazing white, would make both surface operations and geological exploration difficult.

For these reasons, we believe that a limb site located on the equator has many advantages over a polar site. First, the entire sky is visible from the lunar equator over the course of a month. Second, equatorial sites on the Moon are easily and constantly accessible in minimum energy trajectories from the LEO space station, the probable staging location for base establishment. The Mare Smythii site that we endorse as a lunar base site is not only on the limb, at the equator, but it is in a region containing evidence of a great diversity of geological processes as well as a variety of materials that occur in reasonably close proximity. This region can satisfy all potential lunar base users -- geoscientists, astronomers, miners, and colonists.

ADVANTAGES OF THE MARE SMYTHII SITE

Mare Smythii is a dark, lowland on the east limb of the Moon (Fig. 1). The region is well covered by orbital remote-sensing data; analysis of these data suggests that the region is probably one of the most diverse on the Moon (Figs. 2, 3; Table 1). (For a concise summary of our current understanding of lunar geoscience, see Lunar Geoscience Working Group, 1986). In the following paragraphs, we briefly discuss the advantages of the Mare Smythii region from the perspectives of several potential lunar base users.

Geological considerations. The Mare Smythii region displays the two principal geological units found on the Moon: maria (the dark, smooth plains) and terrae (the rugged, heavily cratered highlands). Mare Smythii consists of dark lava flows that partly fill a much older, multiringed basin. The Smythii basin is one of the oldest lunar basins that retain recognizable ring structure; it is composed of three rings 370, 540, and 740 km in diameter. Basins were formed by the impact of asteroid-sized bodies on the Moon before about 4 billion years ago; the study of the mechanics of their formation and their geological effects on crustal materials is one of the primary tasks of lunar geoscience.

The dark, smooth maria are known from Apollo results to consist of basaltic lava flows; the ages of mare basalt samples returned by Apollo range from 3.9 to about 3.1 billion years. The ages of mare lava flows not visited by Apollo may be estimated by examining the density of superposed impact craters. Results of this exercise for Mare Smythii are shown in Fig. 4; the astonishing result is that the lava flows of the Smythii basin are among the youngest on the Moon. The position of the crater-frequency curve of Mare Smythii relaive to that of dated Apollo-site lava flows indicates that the Smythii basalts are probably 1 to 2 billion years old. (A more precise estimate is impossible because the cratering history of the Moon over the last 3 billion years is only approximately known.) Although very old by terrestrial standards, these are the youngest lunar volcanic products; their study will greatly aid the reconstruction of the volcanic and thermal history of the Moon.

In addition to the lava flows within the basin, several localities exhibit dark mantling deposits (Fig. 2a-c). We know from the Apollo results that lunar dark mantles are composed of volcanic, pyroclastic glasses, such as the Apollo 17 orange and black glass beads. Lunar pyroclastic glasses form in Hawaiian-type "fire fountaining" eruptions; moreover, the composition of these glasses indicates that they undergo little chemical modification during their ascent from their source regions in the lunar mantle. Thus, study of pyroclastics is important to understand lunar volcanism and the composition of the lunar mantle.

In addition to lava flows and pyroclastics, several craters inside the Smythii basin appear to modified by internal processes (Figs. 2a-c, 3a). These features, floor-fractured craters, are not uncommon on the Moon and many are associated with the margins of the maria and other sites of volcanic activity. One hypothesis for their origin is that the subfloor zones of impact craters become sites of magmatic intrusions; the continuing injection of magma has uplifted the crater floor in a doming action that fractured them (Schultz, 1976). The Smythii basin, containing at least eight of these features in different states of development, is an ideal area in which to study the process of internal modification of impact craters.

The lunar terrae or highlands make up the vast bulk of the lunar crust. The crust appears to be composed largely of rocks rich in plagioclase (a silicate mineral rich in aluminum and calcium); such rocks are called anorthosites. One of the early ideas concerning the origin of the lunar crust was that the amount of plagioclase in the crust far exceeds what could be reasonably expected to be produced by partial melting and that a former "ocean" of magma existed on the Moon. The terrae surrounding the Smythii basin contains aluminum-rich terrains that are probably largely composed of anorthosites, heavily brecciated (shattered and reassembled) by impact cratering. These highlands offer an opportunity to study the rock types that make up the lunar crust as well as the effects of impact bombardment on the highlands.

One interesting and important lunar rock type contains high concentrations of potassium, rare earths, and other incompatible elements (those that do not fit well into the crystal stucture of common rock-forming minerals). This rock type, called KREEP, may represent the final stages of the crystallization of the original global magma system. KREEP is not uniformly distributed around the Moon, but in the Mare Smythii region, both Balmer and Mare Marginis contain KREEP-rich rocks (Figs. 2e, 3e; Table 1). At Balmer, KREEP basalts apparently underlie a thin covering of highlands debris, while at Marginis, the high KREEP appears to be associated with the mare lava flows. Thus, the Mare Smythii region offers the opportunity to study the occurrence and nature of KREEP-rich rocks in two entirely different geologic settings.

In short, the Smythii basin offers a wide variety of geologic units and processes for detailed investigation from a lunar base. From a centrally located base site, a series of traverses can be designed to explore this diverse terrain, as will be discussed below.

Geophysical considerations. Structurally, the Smythii basin is of geophysical interest because (1) it is similar in size and depth to the younger Orientale basin (Wilhelms, 1987) and (2) it formed on a crust thought to be thicker (about 60 to 80 km thick) than that beneath the Apollo 12 and 14 landing sites (45-60 km thick). This thicker crust is inferred from the relative absence of strain-induced grabens in the Smythii region, which implies that subsidence was minimized by a thicker ancient lithosphere (Solomon and Head, 1980). Because the lithosphere and the more plastic aesthenosphere were probably identical with the differentiated crust and mantle at the time of basin formation, a thicker crust is expected. A regional seismic network near a base in Mare Smythii would test this expectation directly and provide a determination of subsurface wave-velocity structure in a thick crustal zone to complement the Apollo results.

The Smythii and Orientale basins are also structurally similar in that they contain relatively thin mare basalt flows and strong gravity anomalies that imply the existence of subsurface mass concentrations or "mascons". The mascons are caused partly by the surficial mare basalt flows and partly by impact excavation of less dense crustal material, followed by rising of the denser mantle to compensate isostatically for the excavated crustal material. The existence of a relatively thick ancient lithosphere beneath Smythii is believed to have assisted in the preservation of a mascon beneath this basin despite its relatively thin mare fill. Geophysical characterization of the subsurface density structure under the Smythii basin (inferred from seismic and gravity surveys) will therefore provide a general test of models for the structure of mascon basins.

Mare Smythii is adjacent to a large group of swirl-like albedo markings north and east of Mare Marginis (Fig. 2d). Although the origin of these swirls is poorly understood, they are similar to markings found elsewhere on the Moon and are closely associated with strong magnetic anomalies detected from lunar orbit (Fig. 5). A base in Mare Smythii would therefore afford an opportunity to investigate the magnetic anomaly sources. In addition to establishing the nature of the swirls, such an investigation would further constrain the origin of lunar paleomagnetism, an enigma raised by the Apollo data (see Lunar Geoscience Working Group, 1986). The swirls have been suggested to be either surface residues of relatively recent cometary impacts or zones of the lunar surface that have been shielded from the ion bombardment of the solar wind by the associated strong magnetic fields. In the latter model, solar wind hydrogen is considered a necessary part of the process that results in darkening with time (or "optical maturation") of lunar surface materials (Hood and Williams, 1988). To complicate matters further, it is likely that transient plasmas produced during hypervelocity impact are responsible for generating short-lived magnetic fields that magnetized some lunar surface materials. Geologic and magnetic investigations at a lunar base may verify this process in detail for the benefit of future paleomagnetic investigations of the Moon and similar bodies in the solar system.

At a lunar base, it will be important to obtain new heat flow measurements in situ to supplement the two Apollo measurements. A determination of the global mean heat flow is important for constraining not only the thermal state of the interior but also the bulk lunar composition (through the inferred abundance of heat-generating radioactive elements such as uranium and thorium). A major deficiency of the Apollo 15 and 17 heat flow determinations is that both were obtained near mare-terra boundaries, transitions between a surface with a thick, insulating megaregolith layer (the highlands) and a surface with a very thin insulating layer (the maria). Because heat flow at such boundaries is expected to be anomalously large, the Apollo measurements may not be representative of the Moon as a whole (Warren and Rasmussen, 1987). Although indirect orbital measurements of lateral variations in heat flow may be made prior to the establishment of a lunar base, direct Apollo-type heat flow determinations at additional sites around the Moon almost certainly will be required to establish the absolute magnitude of global lunar heat flow. Measurements at sites in and around Mare Smythii (or any other circular mare) would allow an evaluation of heat flow as a function of megaregolith thickness. Mare Smythii is also known to be higher in radioactivity than the surrounding highlands (Fig. 3e); heat flow measurements in the Mare Smythii region, combined with orbital measurements of lateral heat flow and of surface abundance of radioactive elements, would therefore contribute ground truth for a more accurate evaluation of mean global heat flow.

Astronomical considerations. The uses of a permanent lunar base for astronomical observations have been described in detail (Burns and Mendell, 1988; Smith, 1988). Here we will note the advantages offered by the Mare Smythii site for an astronomical observatory.

As discussed above, it is highly desirable to establish a radio astronomical observatory somewhere on the lunar far side, out of view of our electromagnetically noisy home planet. Perhaps the greatest advantage of the Smythii site is that the radio observatory could be near the main base, but out of Earth radio range. Moreover, the equatorial location of a Smythii site would ensure that the entire sky would be visible over the course of each lunar day (about 28 terrestrial days). The potential for astronomy at wavelengths other than radio is as exciting in this region as at any site on the Moon.

From a geocentric viewpoint, the Moon experiences about 8 degrees of longitudinal libration: any point beyond 98 degrees longitude is never in radio sight of the Earth. However, diffraction effects for very low frequency radio waves (Taylor, 1988) require that the radio observatory be located an additional 75 km east of this longitude (a degree of lunar longitude at the equator is about 30 km). Thus, the prime location for a lunar radio observatory is on the equator at any longitude greater than 100.5 degrees east or west; at these locations, Earth radio noise does not exist. We suggest that the radio observatory for the Smythii base be an outpost, largely automated, located at 0, 101 E (Fig. 6). For routine maintenance of the observatory, a road could be bulldozed and the observatory serviced by tracked or wheeled vehicles. The observatory would be about 330 km from the main base; on a prepared road, routine speeds of at least 30 km/hr could be achieved, making the transit time to the observatory about 11 hours. Transport and servicing could be largely automated or teleoperated, thus greatly minimizing surface exposure risks to the human inhabitants of the lunar base.

The Mare Smythii site offers all the advantages of lunar-based astronomical observation. Its equatorial location would make the whole sky visible and the lunar far side is in close proximity. Other properties intrinsic to the Moon (e.g., hard vacuum, stable platform; see Smith, 1988) are as applicable here as at any lunar base site. In conjunction with its numerous other virtues, the Smythii site easily satisfies the criteria of lunar base astronomical users.

Lunar Resource considerations. A wide variety of potential uses for the indigenous resources of the Moon has been identified (see chapters 6,7, and 8 in Mendell, 1985). Although these proposed uses differ widely by process and required feedstock materials, several lunar resources appear to be common to many different schemes. In general order of decreasing usefulness, these resources are bulk regolith, ilmenite (a Ti-rich mineral), volatiles, Al-rich highlands soils, and KREEP-rich material. Each of these materials is abundant at the Mare Smythii site.

The entire surface of the Moon is covered by a mass of fragmental material, ground up by impact bombardment, called regolith. The most important identified use of bulk regolith will be to shield surface habitats from the harsh radiation environment on the Moon (e.g., Haskin, 1985). This is the easiest recognized use of lunar materials; loose soil can be bulldozed to cover prefabricated living modules. Moreover, expanding human presence in Earth-Moon space will require shielding at space outposts where humans will live (e.g., in geosynchronous orbit or at the Lagrangian points). Thus, bulk lunar regolith may become one of the first economical lunar exports. Both mare and highlands surrounding the Smythii site are suitable for mining bulk regolith; the old age of the highlands suggests that the regolith is extremely thick in these areas, possibly as thick as 30 meters.

The mineral ilmenite is of particular importance in schemes for utilizing lunar resources. Not only would ilmenite be useful in the production of oxygen on the Moon by a reduction process (e.g., Gibson and Knudsen, 1985), but ilmenite-rich soils contain high concentrations of ^{3}He, implanted on the grains by the solar wind over geologic time. This ^{3}He, used in terrestrial nuclear fusion reactors, could become the most profitable lunar export resource (Wittenberg et al., 1986). Ilmenite is abundant in the mare basalts from the Apollo 11 and 17 landing sites; remote-sensing data show that these high-Ti basalts are widespread within the maria, including Mare Smythii (Fig. 3g; Table 1). Photogeological evidence suggests that the mare basalts here are relatively thin, but a significant amount of the observed soil chemistry of Mare Smythii is contributed by underlying highlands debris, added to the soils by vertical impact mixing (e.g., Rhodes, 1977). This observation and the observed relatively high Th content of Smythii soils (Table 1) suggest that the basalts of Smythii are similar in composition to the Apollo 11 high-K subgroup of high-Ti basalts. These basalts contain about 20 percent by volume ilmenite and about 7 wt.% Ti (BVSP, 1981). Thus, the Mare Smythii basalts are prime candidates for any mining process that requires large amounts of ilmenite.

Volatile elements appear to be rare on the Moon. However, notable concentrations, including zinc, sulfur, and lead, are found on the surfaces of lunar pyroclastic glasses. Not only are these materials important for what they tell us about the indigenous lunar volatile content, but they also constitute a potential resource. As noted above, dark mantle pyroclastics are abundant in the Mare Smythii region (Figs. 2, 3a) and are present in minable quantities on the basin floor. Because they are of small extent, we cannot be certain of their composition; however, pyroclastics found at the Apollo sites appear to be broadly similar in composition to their associated mare basalts. Thus, the Smythii pyroclastics are probably also of the high-Ti variety.

The highlands surrounding Mare Smythii display some of the highest Al concentrations seen in the Apollo orbital data (Fig. 3c). This suggests the presence of nearly pure anorthositic soils, an Al-rich material that is readily usable for construction on the lunar surface and in Earth-Moon space. One proposed process, which requires such soils, involves fluorination of anorthite (the major mineral in anorthosites) to produce both oxygen and aluminum (Burt, 1988).

KREEP, a material rich in trace-elements, is also available at the Smythii site, and it may ultimately be needed for phosphorus to support lunar agriculture. Although its extraction is probably an element of the advanced lunar base, it is fortunate that significant KREEP deposits are near the proposed base site.

Virtually every use of lunar resources that has been thus far proposed can be accomplished at a base site within Mare Smythii. Thus, the geological diversity that makes the Smythii site such an attractive candidate for geoscience exploration also makes it a prime candidate for lunar resource exploration and utilization.

Summary. The Mare Smythii region has several attractive attributes for the siting of a permanent lunar base. Its location on the equator and limb combines the best of the near side and far side base advantages and permits easy access to the lunar surface from the supporting LEO spaceport. The geological diversity of the region, which contains mare basalts, pyroclastics, KREEP-rich rocks, and aluminous highland soils, permits a wide variety of surface scientific exploration and resource utilization. The regional context of the Smythii site is significantly different from the Apollo sites, thus enabling detailed comparative geophysical studies. A base established in Mare Smythii has the potential to service the various scientific and engineering users of such a base from one central location.

GEOLOGICAL FIELD INVESTIGATIONS AT SMYTHII BASE

Having detailed the numerous merits of the Smythii region for a lunar base, we now briefly discuss some model studies that could be carried out from such a base. For the purposes of this discussion, we will tentatively place the main base site at exactly 0, 90 E (Fig. 6); this site is on the high-Ti basalts of Mare Smythii and is a good location to utilize the ilmenite resources of the mare floor for both oxygen production and possible mining of ^{3}He. Moreover, the smooth, flat surface of Mare Smythii will also be conducive to the ultimate construction of a mass driver, thus making the export of lunar material cheap and reliable. Because the Earth will be out of radio view from this site on some occasions during the libration cycle, we show a communications outpost on the west rim of the Smythii basin at about 0, 81 E (Fig. 6). Here, a radio installation will have a permanent view of the Earth for base communications purposes; communication from this outpost to the main base ultimately will be established by direct optical link. For base start-up operations, either a surface relay network or a temporary comsat will provide a continuous radio link with the Earth.

We describe below three separate strategies for geological exploration based on the distance from the main base to features of interest at ranges of < 100 km, 100 - 500 km, and > 500 km from the base (Fig. 6). For extensive traverses beyond 100 km, we envision that most geological exploration will be by teleoperated robots (Spudis and Taylor, 1988) which would be directly controlled by geologists who remain at the main base site. These robots have many advantages over human field workers and could effectively conduct most of the exploration advocated here. Follow-up visits by human geologists are assumed; these visits would largely consist of quick sorties to minimize extensive and complex life-support systems and risks from radiation. Although planning for detailed traverses and field work cannot be done until a base site is selected, the following exploration plans are offered as examples that could be undertaken from a base in Mare Smythii.

Near-base activities (< 100 km radius). In the early stages of base establishment, most geological work will probably be done near the base site; also, several scientific problems lend themselves well to near-base work even after the longer traverses begin. Thus, field work near the base will start at the time of base emplacement and continue for the indefinite future.

The base's location on the basalt flows of Mare Smythii will provide the opportunity to study both regolith formation and lava stratigraphy. To determine the complete history of regolith formation and evolution, it will be most useful to bulldoze a pit down to bedrock (at this site, probably no more than a couple of meters, because of the young age of the Smythii lavas). The early stages of regolith growth are still almost completely unknown; within this pit, we can study the bedrock interface and address questions of grain-size evolution and soil maturity as study proceeds upsection. The sequence of lava flows and possible changes in magma composition with time can also be studied at the base site; this study of the regional bedrock unit can be done either directly (by shallow drilling and coring of the basalt flows) or indirectly (through the sampling of the ejecta from small craters in the mare to reconstruct possible subsurface layering).

Small particles of highlands rocks are found in the mare soils of all Apollo landing sites, and the aluminous composition of Mare Smythii (Table 1) suggests that this site is no exception. Most of these particles are derived from directly beneath the surface flows by vertical impact mixing of sublava highlands terrain. Thus, even though the base will be located on the mare flows, samples of the terra basin-floor materials will be available within the mare soils. Determining the composition and history of the highlands surrounding the Smythii base site will be one of the prime long-range tasks for the base geologists.

Farther afield, both the extensive dark-mantle pyroclastics and the internally modified floor-fractured craters are within 100 km of the main base site (Figs. 2a-c, 3a, 6 (I)). The pyroclastics should be sampled to determine their place in the general volcanic history of the Smythii region, their possible compositional affinities to the mare basalt flows, the nature of their mantle source regions beneath the crust in this area, and their potential as minable resources. The floor-fractured crater Purkyne U (Fig. 2c) lies about 60 km east-southeast of the base site; this crater has an uplifted, fractured floor, partial fill by mare basalt (erupted from the crater interior, as demonstrated by its unbreached rim), and a partial covering of dark pyroclastic material. Detailed field study of this crater could elucidate the processes of internal modification of lunar craters and contribute to our understanding of the volcanic history of the basin.

In addition to these primary studies, several smaller scale ones will be conducted in the near-base area. These will include study of a large population of small (< 1 km diameter) craters to understand their formational mechanics and the regional cratering history, study of the lateral variations in both the lava flows and the subfloor basement, and the nature of crater rays. (This area is covered by rays from distant craters and it is important to establish the exact amounts of crater primary ejecta contained in ray material). These studies alone are of significant importance and complexity to provide the base geologists with challenging exploration opportunities.

Short-range traverse activities (100-500 km radius). The middle range of exploration traverses is illustrated by the 500 km circle of Figure 6. In this range, almost all of the diverse geological features of the Smythii region are available for study. Model traverse route II (Fig. 6) could be followed by a teleoperated robot investigating the materials and processes described below. Undoubtedly, significant discoveries made along the way will perturb the actual route, but route II as shown encompasses most of the currently identified field geology goals.

In the first leg of the traverse, the lateral heterogeneity of the young Smythii lava flows north of the base will be investigated (Fig. 6). The north rim of the basin will be sampled to determine its relation to materials of the basin floor, collected near the base site (see above). Next, the traverse will continue north into the lava flows of the Mare Marginis basin (Fig. 2d). These lavas also appear to be relatively young (about 2 billion years old); moreover, they are enriched in Th (up to 3.4 ppm; Fig. 3e). This suggests that they are a variety of KREEP-rich mare basalt, rare in the Apollo collections, and their study could shed light on the process of igneous assimilation of KREEP into mare basalt magmas.

The traverse will continue north to sample and investigate the mysterious swirl materials of northern Marginis (Fig. 2d). As described above, these swirls are associated with large surface magnetic anomalies (Fig. 5) and field studies of their composition and local environment are required to fully understand their origin. It would also be of interest to visit the crater Goddard A, as it has been proposed that this crater may be related to the Marginis swirls.

The traverse will now turn south, across Mare Marginis to determine its lateral variations, cross the mare-filled crater Neper, and return to the Smythii basin. One goal of this leg is to examine the lateral variations of the highland deposits making up the Smythii basin rim. The trip will continue south into the basin to examine and explore the floor-fractured craters Schubert C (Fig. 2b), Haldane, and Kiess. These craters display a range of modification states, and comparative studies between them and the previously studied Purkyne U (see above) will enable a resolution of the problem of their origin. In addition, this leg of the traverse covers the most abundant dark mantle deposits and local volcanic vents of the region (Figs. 2b, 3a). Field study of these features will aid in a detailed reconstruction of the volcanic history of the Smythii basin.

In the final leg of this traverse, we will study the highlands of the Smythii basin's south and west rims (Fig. 6). When this leg is completed, we will have a fairly complete knowledge of the lateral variations in basin rim deposits. We may even find evidence for large-scale compositional zoning within the basin ejecta deposits, a feature long postulated for basin geology based on incomplete and inadequate remote-sensing data, but as yet unproven on the Moon. This geological traverse provides a variety of features and processes for direct study, all within a fairly short traverse radius.

Long-range traverse activities (> 500 km radius). Beyond the 500 km limit, virtually the entire Moon beckons for detailed exploration. Indeed, one of the advantages of the teleoperated robot system is that it turns a single-site base into a "global base" by providing access to any point on the Moon (Spudis and Taylor, 1988). For the purpose of brevity, we here restrict our attention to a likely early long-range traverse, a mission to explore and sample the intriguing Balmer basin (Fig. 6, III).

As noted previously, Balmer is an old multiring basin apparently filled with light plains materials of Imbrian age (Fig. 2e, 3a). This otherwise unremarkable basin is worth investigating for two reasons: (1) the light plains that fill Balmer display dark-halo craters (Fig. 2e), indicating the presence of a subsurface basalt unit at least 3.9 billion years old; and (2) orbital gamma-ray data suggest that this area is rich in KREEP (Fig. 3e), having a local Th concentration of 4 ppm. Moreover, this Th enrichment is coincident with the plains displaying dark halo craters, suggesting that the KREEP component is associated with the underlying, ancient lava flows. In combination, these observations suggest the presence of ancient KREEP basalt flows; flows of this composition have long been postulated in the lunar literature, but thus far we have identified only one example, the planar Apennine Bench Formation near the Apollo 15 landing site. Because the concept of KREEP volcanism is so important to models of lunar evolution and because of the controversy over its existence, we have specifically planned this traverse to examine and characterize the volcanic fill of the Balmer basin.

The traverse begins by exploring the southwestern floor and rim of the Smythii basin, previously unvisited, to determine more completely the nature of the highlands around Mare Smythii and to provide comparative data for the previous traverses. This route includes a complete traverse of the crater Ansgarius; not only can we investigate the geology of this large, complex crater of Imbrian age, but this location also demarcates the crest of the outermost ring of the Smythii impact basin. The internal structure of this basin ring may be exposed within the walls of Ansgarius, thus making the detailed geologic structure of the ring available for study.

The traverse next proceeds to the plains of the Balmer basin. The goals in this area include characterization of the Imbrian age light plains to determine their provenance and study of the dark-halo craters to understand their internal structure and ejecta. It is within the ejecta of these craters that we hope to find the long-sought KREEP basalts; through study of the ejecta volumes and their distribution around the craters, we can estimate the thickness of the overlying highlands debris mantle and, possibly, the thickness of the buried ancient basalts. Another important goal at this stop is study of regolith developed on the ejecta blankets of the dark halo craters to understand how they form the strong photometric contrast seen in orbital photographs. These tasks involve intensive field work; an advantage of using robots here rather than human field geologists is that as much time as is required can be spent in the field area to completely understand and solve these problems.

On the return trip to base, we will investigate the west basin rim and the light plains fill of the craters Gilbert and Kastner G (Fig. 6). At these two craters, an important question is the possible relation of their plains fill to that in the Balmer basin. If these light plains are related to the Crisium basin to the north (Fig. 1), these stops will test the concept of lateral variation in basin debris blankets and could also address the vexing question of primary basin ejecta versus locally reworked material in highland plains materials. On the final leg, we will continue previously started field studies of the Smythii basin floor material, dark-mantle deposits and vents, and a previously unvisited floor fractured crater, Runge (Figs. 3a, 6).

Summary. These three strategies of geological exploration demonstrate the amazing variety of geological units and processes that are available for direct exploration at the Smythii base site. The units represent the range of lunar geologic processes and absolute ages, from the ancient brecciated highlands crust to the youngest, rayed craters. Many additional traverses could be described; moreover, after a short time of base operations, many significant new discoveries will undoubtedly be made, thus altering the order of exploration priorities and planning of the actual routes. The total potential of a lunar base for geologic study is of such magnitude that it is impossible to predict the exact schedule and order of surface operations.

GEOPHYSICAL FIELD INVESTIGATIONS AT SMYTHII BASE

Following the order outlined above for geological exploration, we divide the discussion of geophysical exploration into categories depending upon the maximum radial traverse distance from the base.

Near-base activities (< 100 km radius). After base establishment, the first priority for geophysical studies should be the deployment of an Apollo-type geophysical station containing such instruments as a seismometer, heat flow probe, magnetometer, and solar-wind spectrometer. These instruments should be emplaced near enough to the base to allow easy access for maintenance and recalibration but far enough so that base activities do not add an undue amount of artifical noise to the measurements. The structure of near-surface seismic wave velocities can be determined using active sources, perhaps in conjunction with construction or mining activities. To deduce the structure at greater depths, using a single-station seismometer, will require active energy sources of increasing magnitude, comparable to those produced by the planned crashes of LM ascent modules and S-IVB stages during the Apollo program. Measurements from a single heat flow probe should be monitored for at least a year to establish the thermal properties of the surrounding regolith, which are needed for heat flow determination. The final value, if obtained away from the periphery of the mare, will provide a valuable benchmark for comparison with the Apollo values. A single magnetometer and solar wind spectrometer will define the local crustal magnetic strength at the base site and determine the extent of deflection by this magnetic field of ions in the solar wind.

The next order of priority after establishing the base geophysical station is to conduct field geophysical measurements during the surface geological traverses. A local-area network of seismic stations should be emplaced to allow passive seismic studies using meteoroid impacts and shallow moonquake sources. Active seismic sounding using artificial sources will also be very effective using this local array. Heat flow probes can be deployed at a series of sites on different megaregolith thicknesses to obtain a first determination of the dependence of lunar heat flow on this quantity. During exploratory traverses, it will be desirable to obtain direct surface gravity and elevation measurements at specific points along the route to constrain later modeling studies of subsurface density structure. These measurements will provide a ground-truth supplement to Apollo and LGO orbital gravity and topography data. Also, magnetic field and solar wind flux measurements along the traverse will provide the first direct measurements of solar wind deflection as a function of surface magnetic field intensity and direction. The surface magnetic field measurements, combined with orbital magnetic data, will also facilitate modeling of the bulk magnetization properties of large-scale geologic units (e.g., mare basalt flows) in order to constrain the nature and origin of lunar paleomagnetism.

Short-range traverse activities (100-500 km radius). Several important geophysical investigations can be done during the medium-range traverse discussed above (II in Fig. 6). This traverse will enable the deployment of one or more geophysical stations that will become part of a regional network designed to determine the subsurface structure and thermal state of the Smythii region. The primary instruments to be deployed at these stations will be seismometers and heat flow probes. In order to resolve basin structure, individual seismic stations should be no more than about 150 km (5 degrees) apart, requiring at least 8 regional stations in addition to the base station. Active seismic sounding near at least one of the highland stations will allow the first direct crustal thickness determination at a highland site on the Moon. As noted previously, the crust is expected to be substantially thicker in this region than at the Apollo sites. Crustal thickness peripheral to the basin will be larger still because of the expected isostatic raising of the crust-mantle boundary beneath the basin center. Following establishment of both mare and terra seismic velocity and thickness benchmarks using active methods, the passive network will be capable of a first-order determination of the velocity structure beneath the entire basin. In combination with gravity and topography surveys, the two-dimensional velocity model will provide strong constraints on the subsurface composition and density structure of this mascon basin. Heat flow measurements will likewise establish the lateral variation of surface heat flow and probable subsurface thermal state as a function of radial distance from the basin center.

A traverse to the Mare Marginis swirl belt will make possible direct surface magnetometer and solar wind spectrometer measurements at the site of one of the largest magnetic anomalies on the Moon (Fig. 5). As stated above, simultaneous geologic investigation and sampling of the swirls should establish their origin. As a by-product of these investigations, solar wind spectrometer measurements will determine the lateral variation of the implantation rate of solar wind gases (mostly hydrogen and helium) into the uppermost regolith. For example, the strongest lunar magnetic anomalies are probably capable of completely deflecting bombardment by ions of the solar wind (Hood and Williams, 1988). This process will lead to zones of relatively low implantation rates near the centers of large surface anomalies and zones of high implantation rates in complex, curvilinear areas peripheral to the same anomalies. Measurement of these fluxes will be helpful for evaluating the volatile resource potential (i.e., the extraction efficiency of trapped solar wind gases; Haskin, 1985) of different source regions. Furthermore, the strongest magnetic anomalies are characterized by surface field amplitudes that probably exceed several hundredths of a Gauss (for comparison, the Earth's field near the equator is about 0.3 G). Depending upon their horizontal scale, these relatively strong crustal fields may be capable of significantly deflecting a part of the solar cosmic ray flux during flare events. If such deflection is beneficial in reducing the hard radiation environment for human activities, then it may even be desirable to locate outposts or bases within the shelter of strong magnetic anomalies.

Long-range traverse activities (> 500 km radius). When the robotic field explorations are extended to greater distances, identified geologic targets can be characterized using geophysical methods. For example, along the suggested route III of Fig. 6, small-scale seismic sounding and surface gravity measurements may be useful in delineating the thickness of subsurface basalt units in the Balmer basin and in identifying the crest of the outermost ring of the Smythii basin. More generally, remote geophysical stations may be deployed to allow seismic and electromagnetic sounding of the deeper lunar interior. These stations would be part of a global-scale network that should be established in the course of continuing field investigations. Among the major objectives of large-scale seismic and electromagnetic sounding network studies are determinations of the seismic-velocity profile of the lunar mantle and of the existence and size of a possible metallic core. Although core detection may be achieved earlier through alternative approaches, a detailed characterization of the size, mass, and physical properties of the core will probably require long-term seismic measurements using a large number of stations. Similarly, a more accurate appraisal of mantle structure and thermal state will need both long-term and large-scale seismic, electromagnetic, and heat flow measurements. Thus, the geophysical stations deployed in the course of lunar base activities and traverses will contribute significantly to an eventual accurate determination of the structure, composition, and thermal state of the deep lunar interior. Because the bulk composition of the Moon (including core size and mass) is a basic constraint of lunar origin models, such a determination will lead to a much improved understanding of the origin of the Moon.

CONCLUSIONS

We have demonstrated that the Mare Smythii region holds great promise as a lunar base site from a scientific, operational, and resource utilization viewpoint. This site enables enough flexibility to satisfy any potential lunar base user. Among its attributes are the following:

(1) Its location on the lunar limb permits the establishment of a base complex that combines the benefits of a near side base (for ease of initial and routine base operations) and a far side base (to shield the radio astronomical observatory from the electromagnetic interference produced by the Earth).

(2) Its equatorial location allows for easy base access from the LEO space station and also permits a clear view of the entire sky for astronomical observations.

(3) The Smythii region abounds in a diversity of both geologic features and natural resources. This diversity permits a wide range of geological and geophysical investigations to be performed and it also provides almost the entire known range of potential lunar resources to be mined, processed, and used on the Moon and in Earth-Moon space.

Nearly all of the identified lunar geoscience problems can be addressed at a base located in Mare Smythii. Some of these problems are the origin and evolution of the lunar crust and mantle, the cratering history of the Moon, the formational mechanics of large craters and basins, the nature and evolution of the lunar regolith, the origins of lunar paleomagnetism, and lunar volcanic history. Geologic and geophysical field studies conducted from the Smythii base will provide data applicable to all of these problems.

Based on our study of the Smythii region, we have tentatively identified the following operational requirements for base establishment and initial operations:

(1) We propose that the main lunar base be located at 0, 90 E, in Mare Smythii. This location will provide high-Ti mare regolith as a feedstock for oxygen production and possibly ^{3}He mining, and it allows for easy access to a variety of important geological and geophysical exploration targets.

(2) At least two installations will be required in addition to the main base. The first is a communications outpost on the west rim of the basin at about 0, 81 E. This site is in constant radio view of the Earth; the outpost will be needed as a relay station when the main base is out of contact with Earth during minimum libration cycles. The outpost will be connected to the main base by an optical link cable emplaced during base start-up; interim Earth-Moon communications can be provided by a temporary lunar comsat.

(3) The second outpost could be a lunar very low frequency radio astronomy observatory. It should be located on the equator east of 100.5 E; we suggest an intercrater area at 0, 101 E, where the observatory will be permanently shielded from radio noise from the Earth. The suggested site is about 330 km from the main base; a road can be constructed to allow easy rover access to service the outpost.

(1) As we envision base operations, most geological field work and emplacement of geophysical instruments can be done by teleoperated robots. Some visits by humans to sites distant from the base will be required.

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FIGURE CAPTIONS

Figure 1. Global view of the eastern limb of the Moon showing Mare Smythii and its relation to the longitudes of lunar libration. Some regional features and two Apollo landing sites are also shown. AS12-55-8226.

Figure 2. Geological features of and near the Smythii basin.

a) Regional view of the Smythii basin. Dark smooth area (M) is Mare Smythii, consisting of high-Ti mare basalts. Smythii basin rim (B) is 370 km in diameter and is composed of anorthositic rocks. Dark mantle (arrows) is pyroclastic ash deposits produced by fire fountain eruptions of basaltic magmas. Many floor-fractured craters (F) are visible on the basin floor. Large, mare-filled crater at upper left is Neper (N), 137 km in diameter. AS15-95-12991.

b) Western part of the Smythii basin floor. The prominent floor-fractured crater (F) is Schubert C (31 km dia.). Mare basalt flows (M) fill the highlands terrain (H) of the basin. Dark mantle deposits are associated with irregular volcanic vents (arrows) in this area. LO I-5 M.

c) Eastern part of the Smythii basin floor, showing lava flows (M) of Mare Smythii, highlands basin rim (H), and floor-fractured crater Purkyne U (F; 51 km dia.). Young rayed crater (arrow) overlies lava fill of Purkyne U. LO I-19M.

d) Regional view of terrain northeast of Mare Smythii (S). Basalts of Mare Marginis (M) are relatively young (about 2-3 billion years old) and rich in KREEP. Swirls within Marginis (big arrow) are associated with large surface magnetic anomalies. Dark-halo impact craters (small arrows) are associated with buried ancient mare basalts, common in this area. Large rayed crater at top (G) is Giordano Bruno (22 km dia.), possibly the youngest large crater on the Moon. Portion of AS16-3021.

e) Regional view of terrain southwest of Mare Smythii (S). The mottled light plains (P) of the Balmer basin display dark-halo craters (arrow) and are KREEP-rich. Large crater near bottom center is Humboldt (207 km dia.). Portion of AS17-152-23293.

Figure 3. Maps of geologic, topographic, and chemical remote-sensing data for the Smythii region. All geochemical data were obtained from the orbiting Apollo 15 and 16 spacecraft and, except for the thorium data, are from La Jolla Consortium (1977); chemical composition of major geologic units is summarized in Table 1. All maps are Mercator projection.

a) Geology, modified from Wilhelms and El-Baz (1977). Relative ages indicated by capital letters: E-Eratosthenian, I-Imbrian, N-Nectarian, pN-pre-Nectarian. Units: Em- basalts of Mare Smythii; Im- other mare basalts; EId- pyroclastic dark mantle deposits; Ip- smooth plains, some displaying dark halo craters; Icf- floor-fractured craters; pNbr- Smythii basin rim material; pNbf- Smythii basin floor material; Ic, Nc, pNc- impact crater materials; NpNt and pNt- undivided terra (highlands) material. Dashed lines indicate basin rings. Smaller circular features are impact craters; lines with ticks indicate crater rims.

b) Regional topography from Apollo metric photographs by U. S. Geological Survey (unpublished, 1982). Elevations based on global datum of a spherical Moon 1738 km in radius.

c) Aluminum concentration data (in weight percent).

d) Magnesium concentration data (in weight percent).

e) Thorium concentration data (in parts per million; from Haines et al., 1978).

f) Iron concentration data (in weight percent).

g) Titanium concentration data (in weight percent).

Figure 4. Crater-frequency distribution for the mare basalt flows of Mare Smythii (squares), shown in comparison to those of the Apollo 11 and 12 landing sites and the crater Copernicus (from BVSP, 1981). Position of the Smythii curve indicates that these lava

flows are significantly younger than those of the Apollo 12 site (the youngest sampled lunar lavas); the age of the Smythii flows is probably about 1 to 2 billion years.

Figure 5. Amplitude of near-surface magnetic fields in the region of Maria Smythii and Marginis as deduced from the reflection of low-energy electrons (from Hood and Williams, 1988).

Figure 6. Relief map of the Smythii region of the Moon, showing location of the main lunar base, the far side radio astronomical observatory, and permanent communications outpost. Circles indicate a 100 km and 500 km radius of action from the main base. Three model geological traverses (I, II, and II) are described in the text. Base is portion of LOC-2, original scale 1:2,750,000.

TABLE 1. COMPOSITIONAL PROPERTIES OF SELECTED GEOLOGICAL UNITS IN THE MARE SMYTHII REGION

Material	Age*	Al wt.%	Mg wt.%	Ti wt.%	Fe wt.%	Th ppm	Comments
Mare Smythii	E/ ~1.2	8.5-10	5- >7.4	2.5-3.5	5.6-7.4	2.4	High-Ti mare basalts thin, with admixed highlands debris
dark mantle	EI/3.5-1	10-11.5	5.8->7.4	2.5-3.5	4.4-5.6	1.2-2.4	Mafic, pyroclastic glasses, probably high-Ti
Basin rim	pN/ ~4.0	11.5->14	<5 -5.8	0.6-1.9	<1.5-4.4	0.5-0.7	Anorthositic debris, breccias
Balmer plains	IN/ 3.9	10	6.2-7.4	1.4-1.9	7.4-9.6	4.0	Thin mantle of terra debris overlying KREEP basalts
Terra, west of basin	NpN/ ~4	11.5-13	<5 -5.8	1-1.9	5-7.4	0.7	Anorthositic norite breccias
Terra, east of basin	pN/ >4	13->14	<5	<0.3-1.4	<1.5-3	0.5	Anorthositic to pure anorthosite breccias

* Relative ages: E- Eratosthenian, I-Imbrian, N-Nectarian, pN- pre-Nectarian. Absolute ages (in billion years) are rough estimates.

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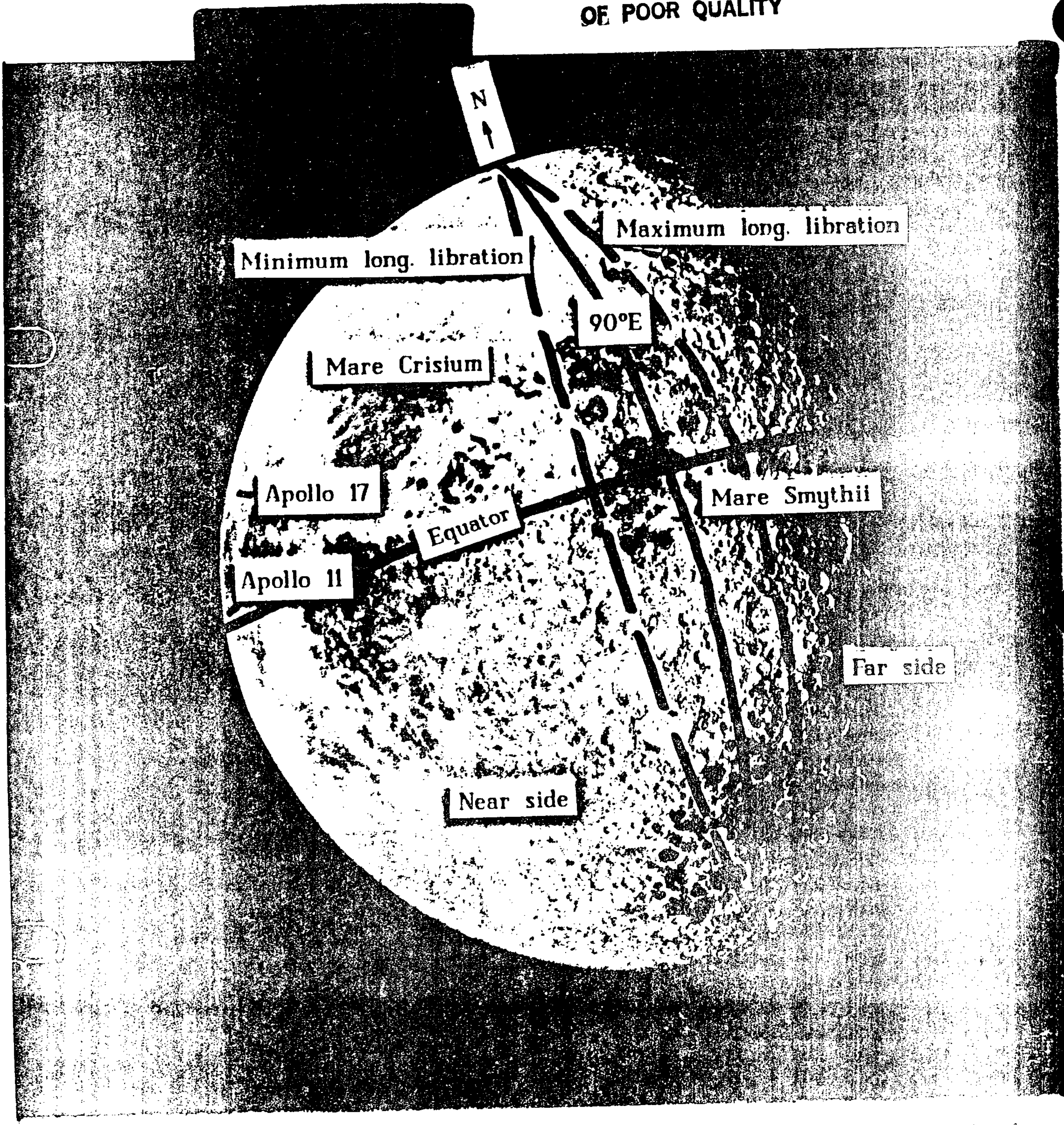

Spudis and Head

Fig. 1

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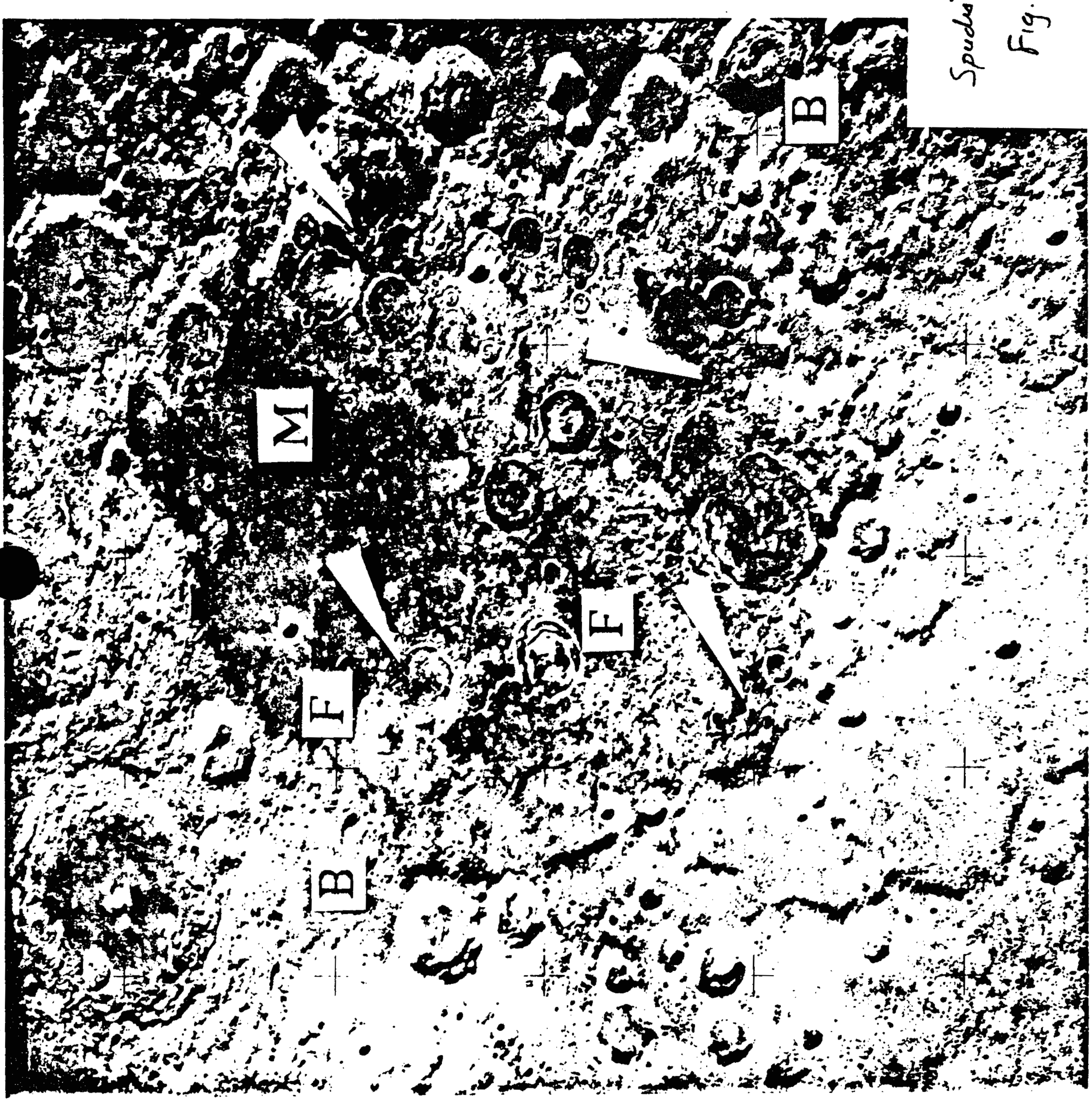

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Spudis and Hood

Fig. 26

Spudis & Hood
Fig. 2c

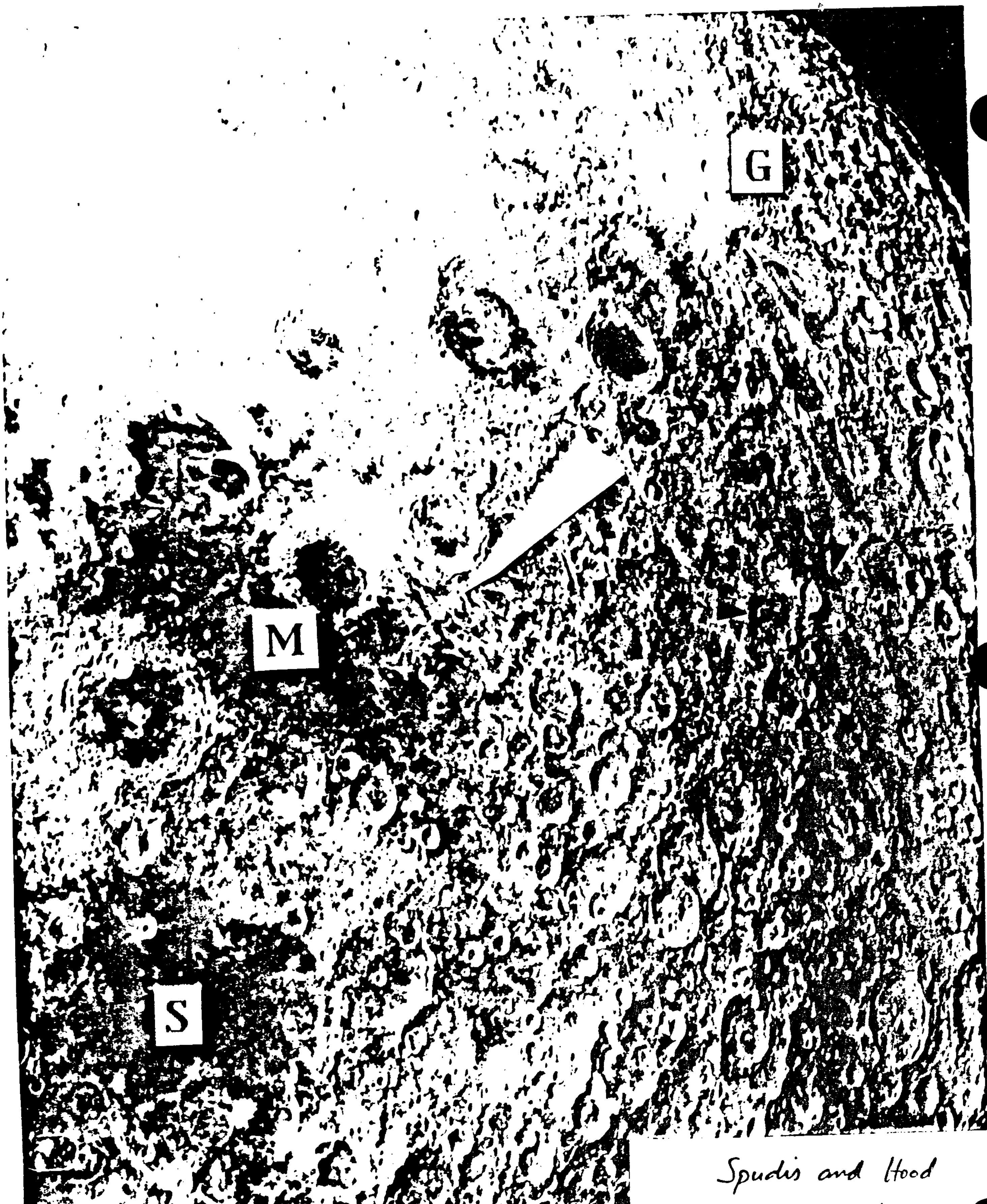

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Fig. 2d

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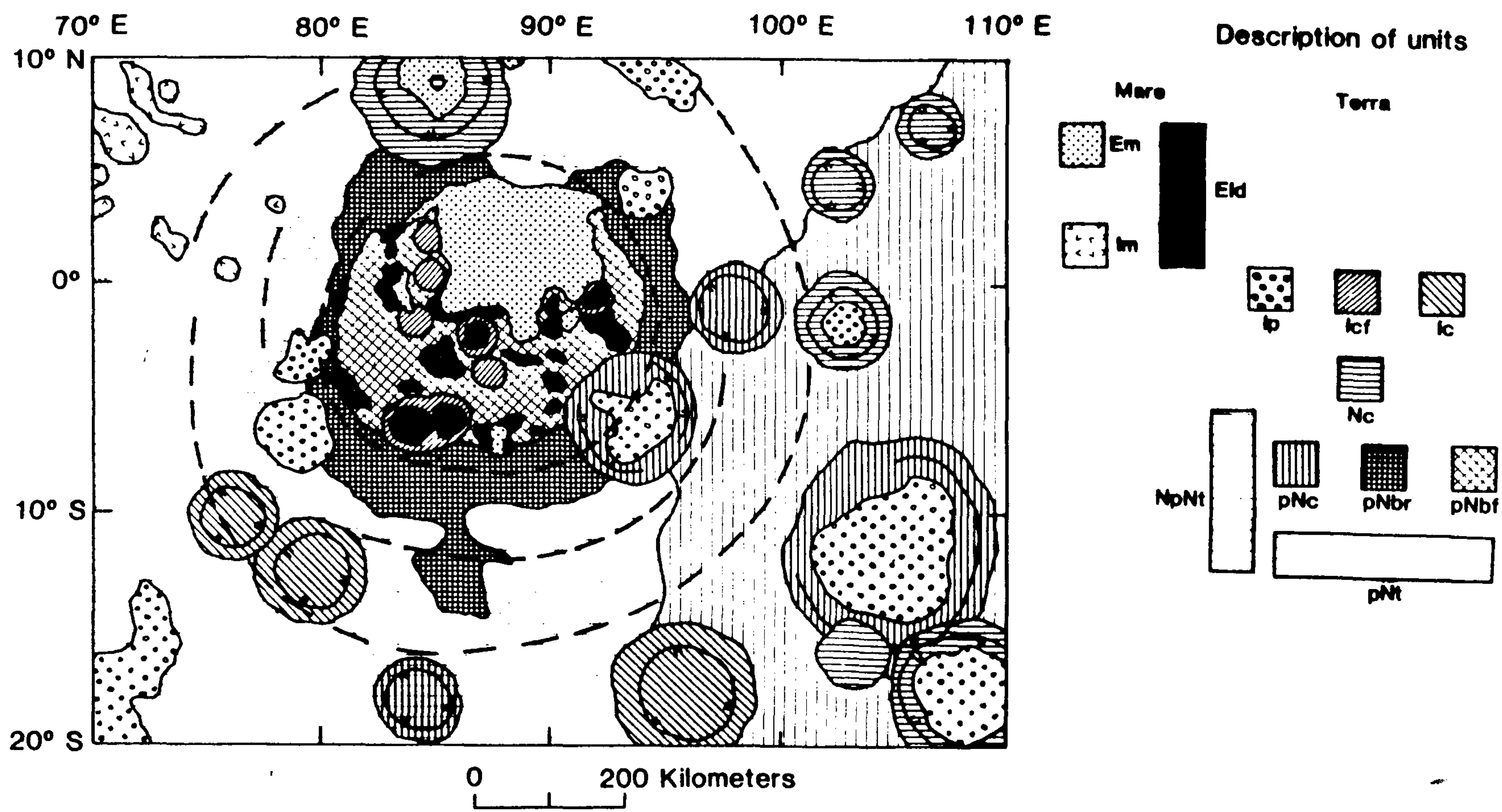
Fig. 2e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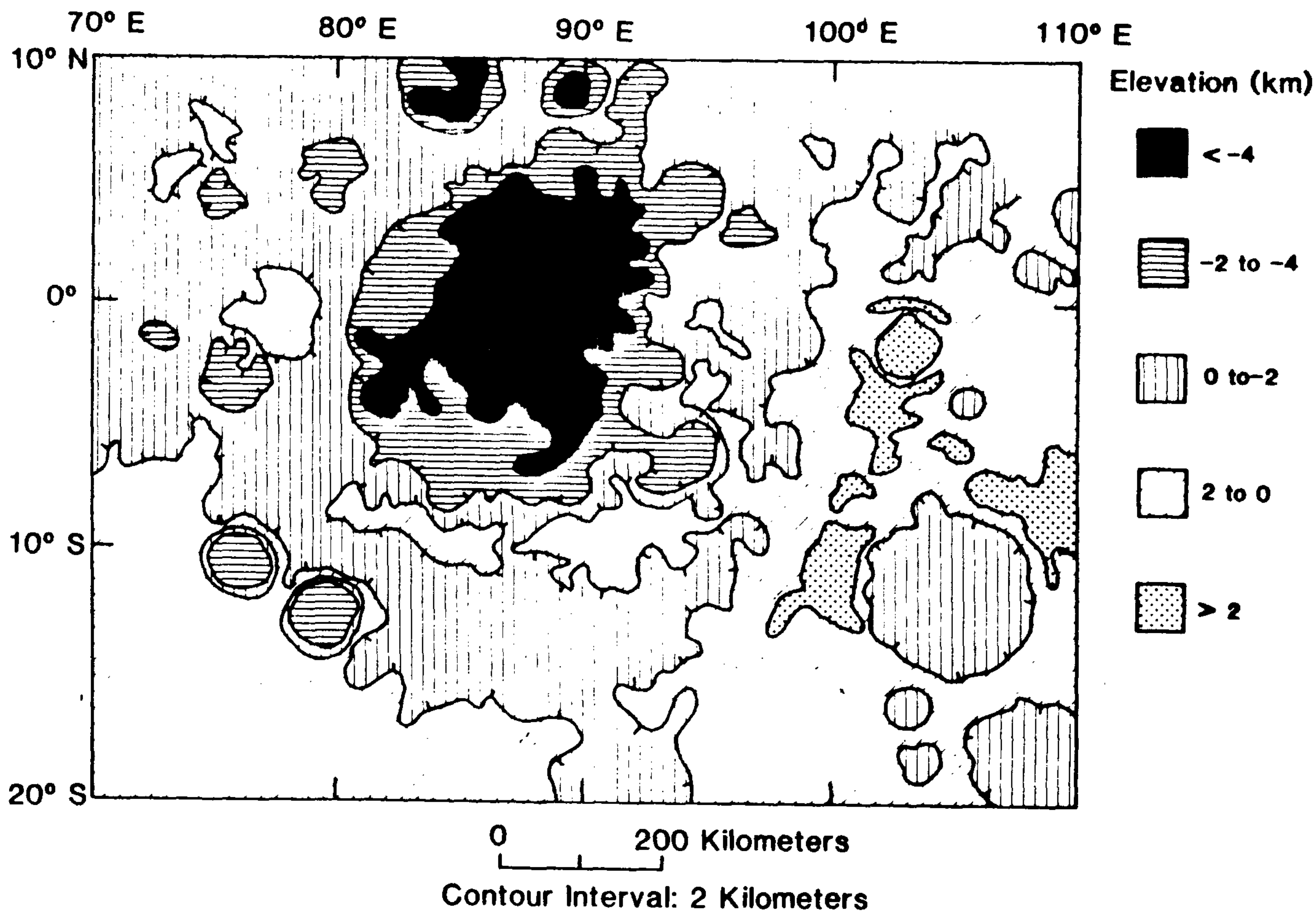
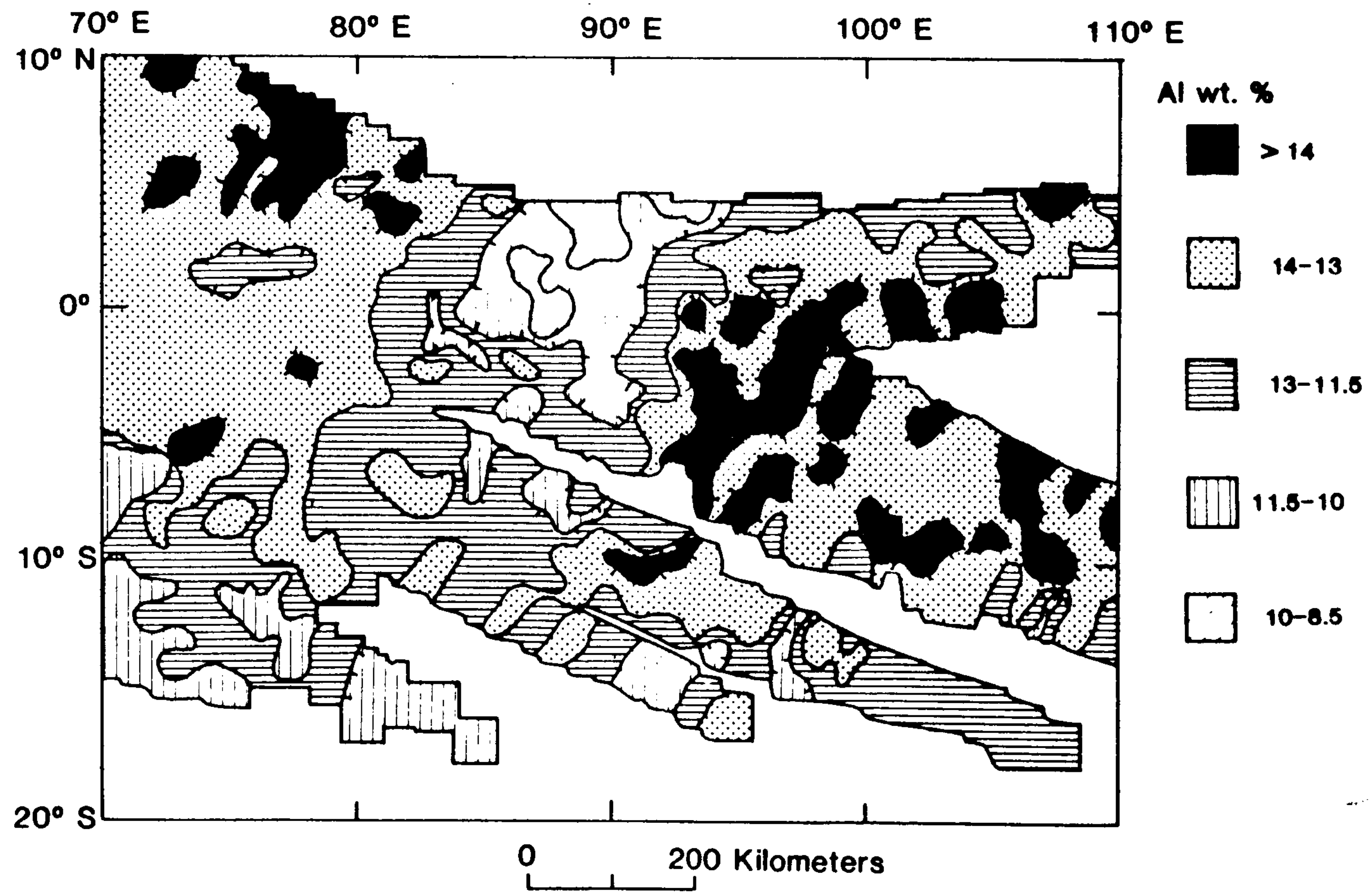
Spudis & Hood



Spudis and Head
Fig. 2b



Spudis & Hood
Fig 3c

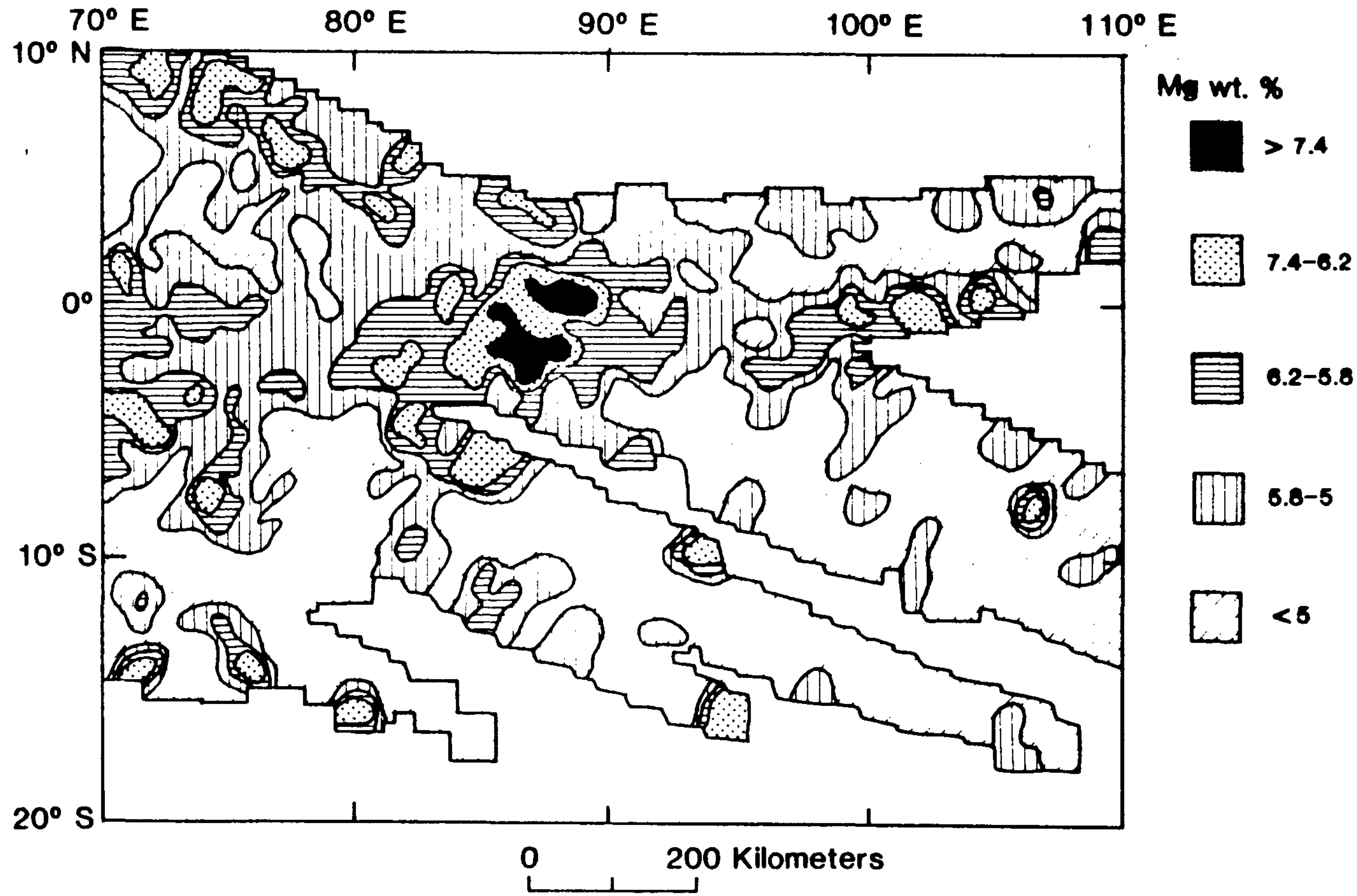


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Fig. 3d

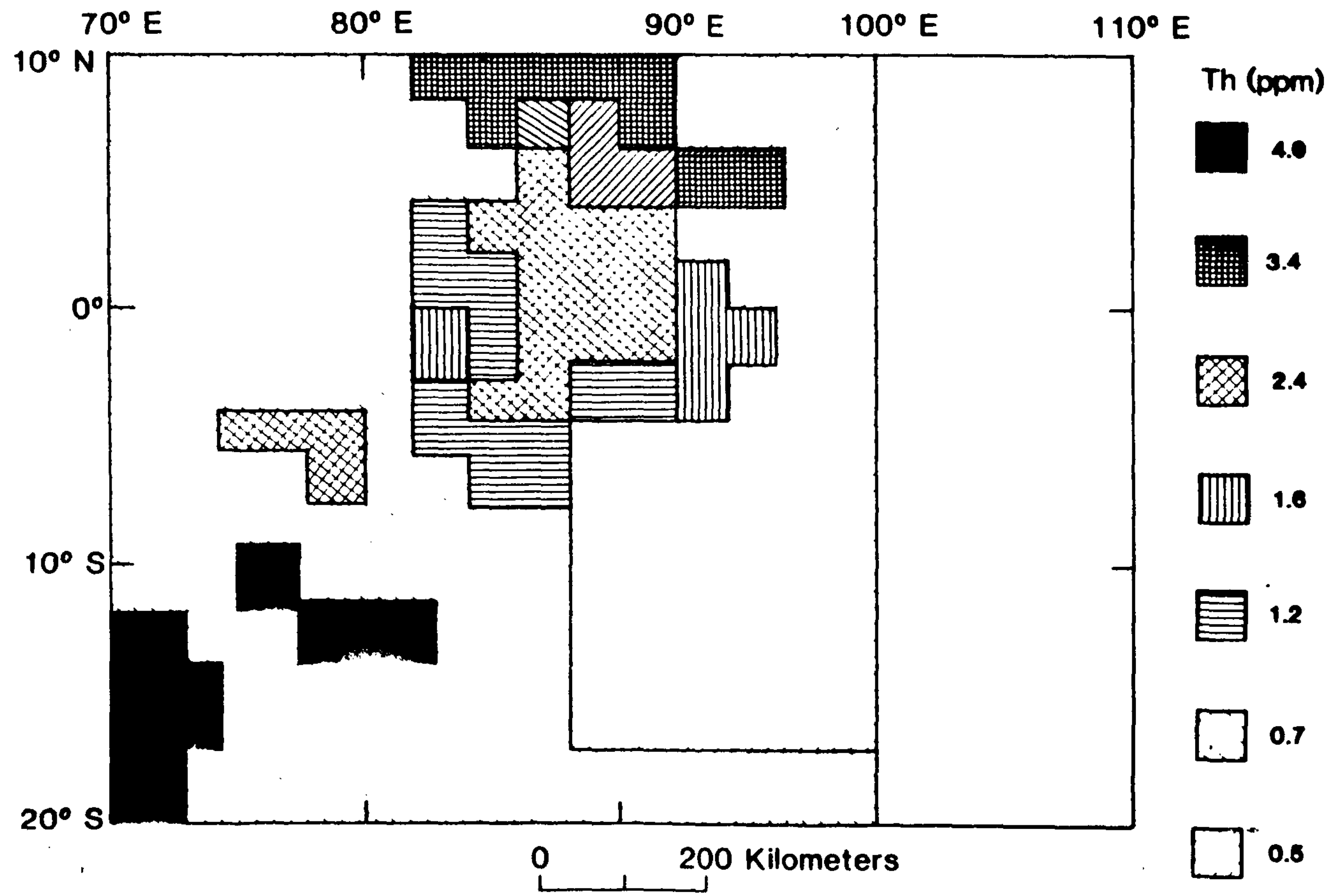

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Fig. 3e

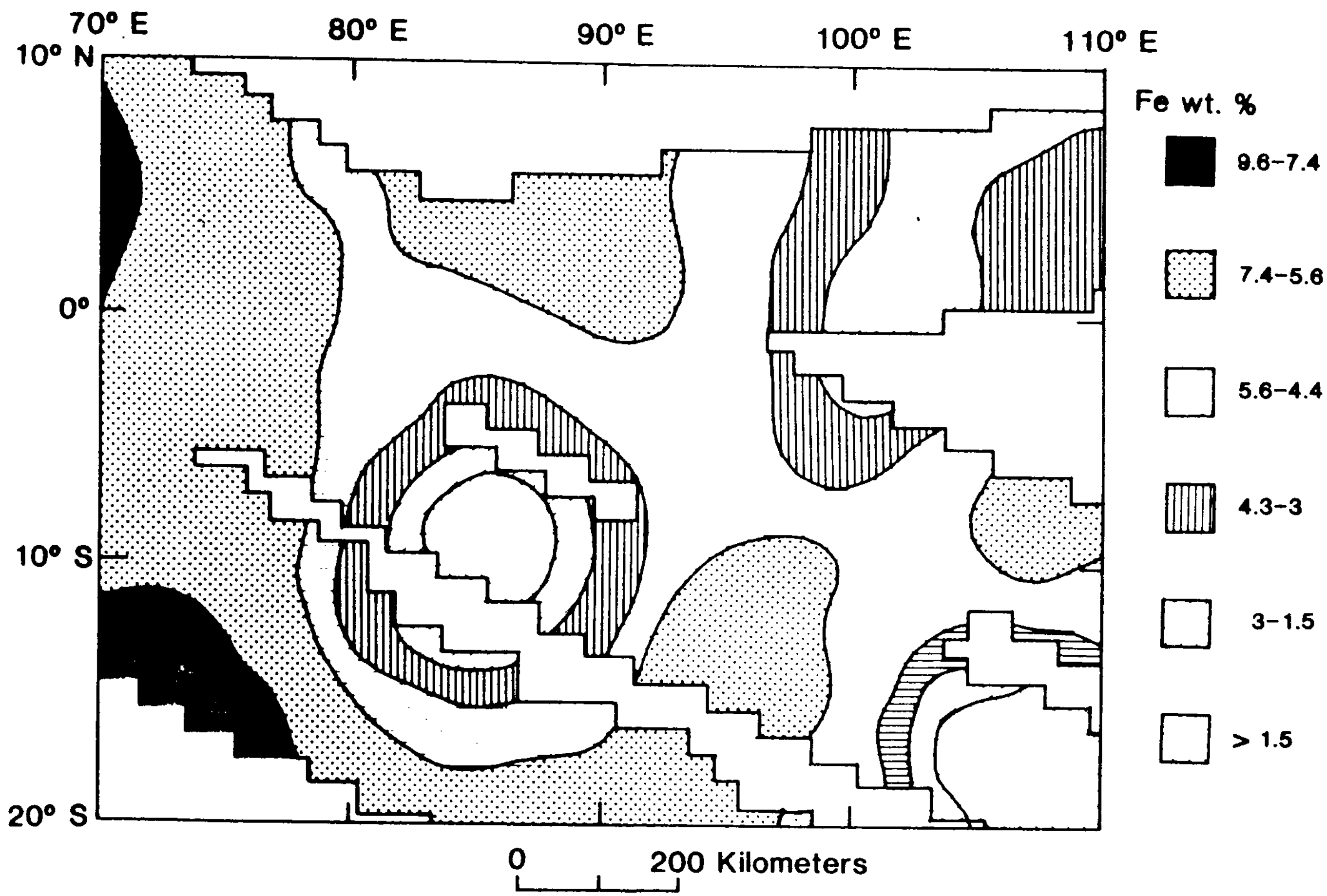

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Fig. 3f

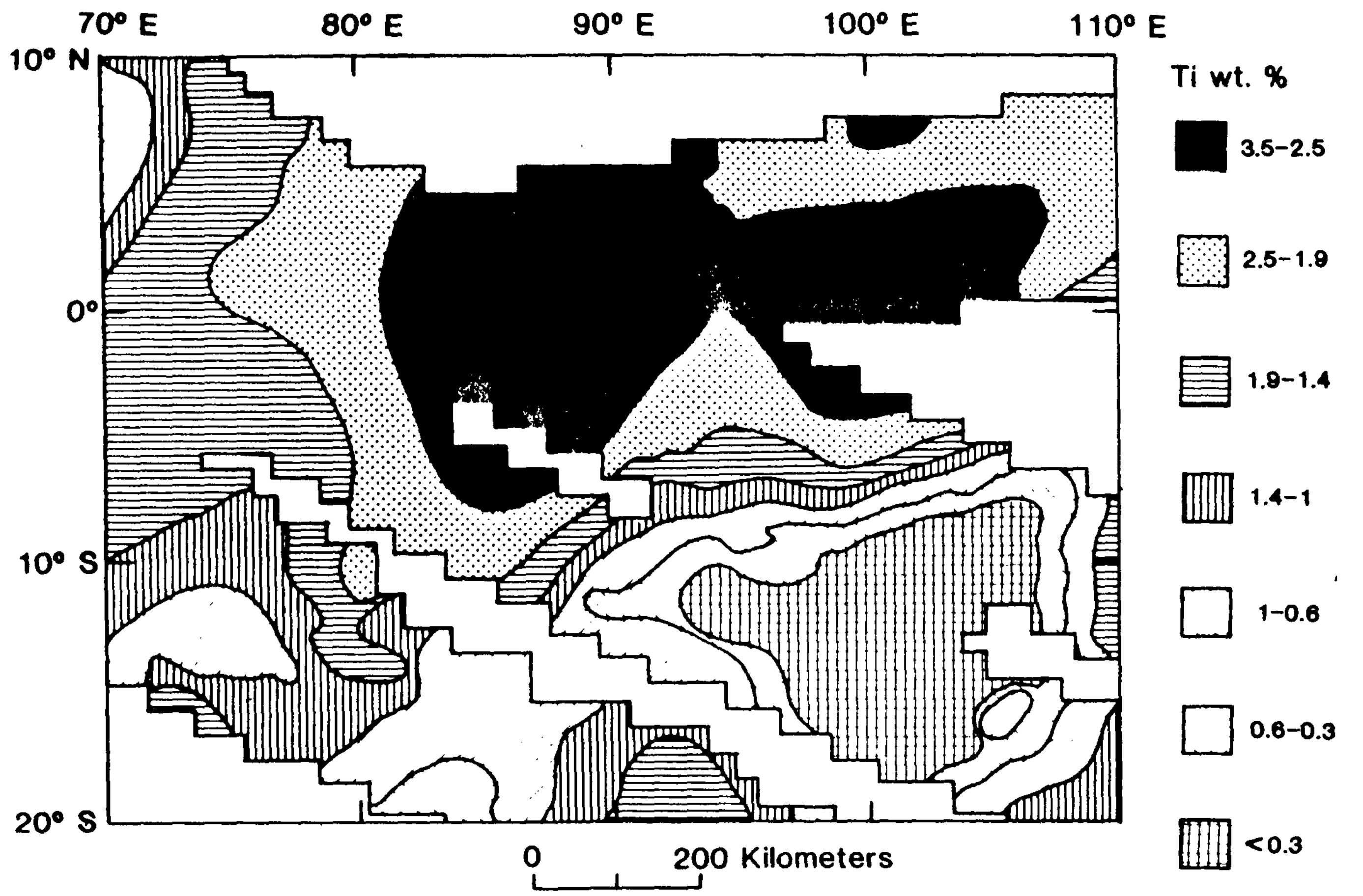

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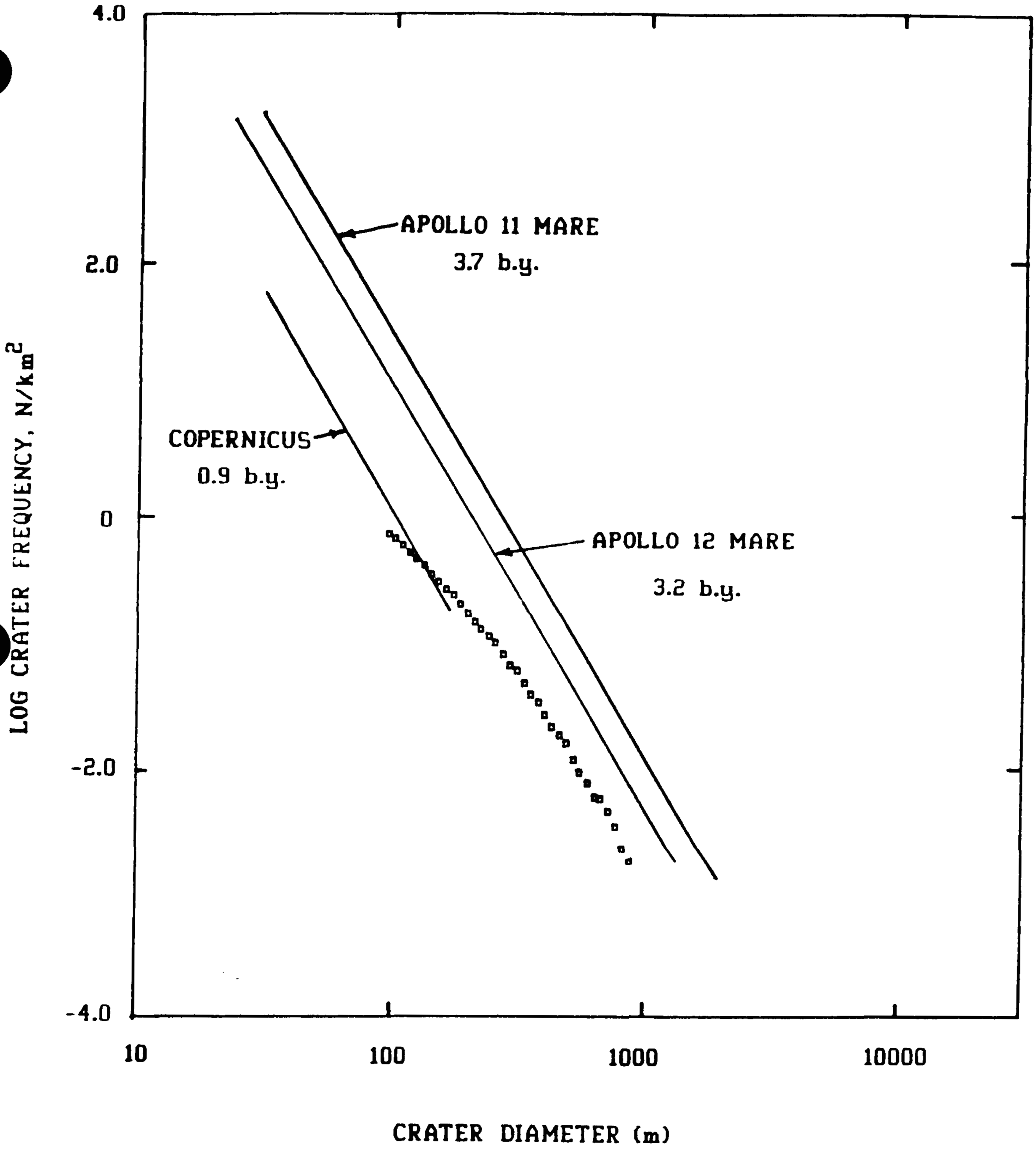
Fig. 3g



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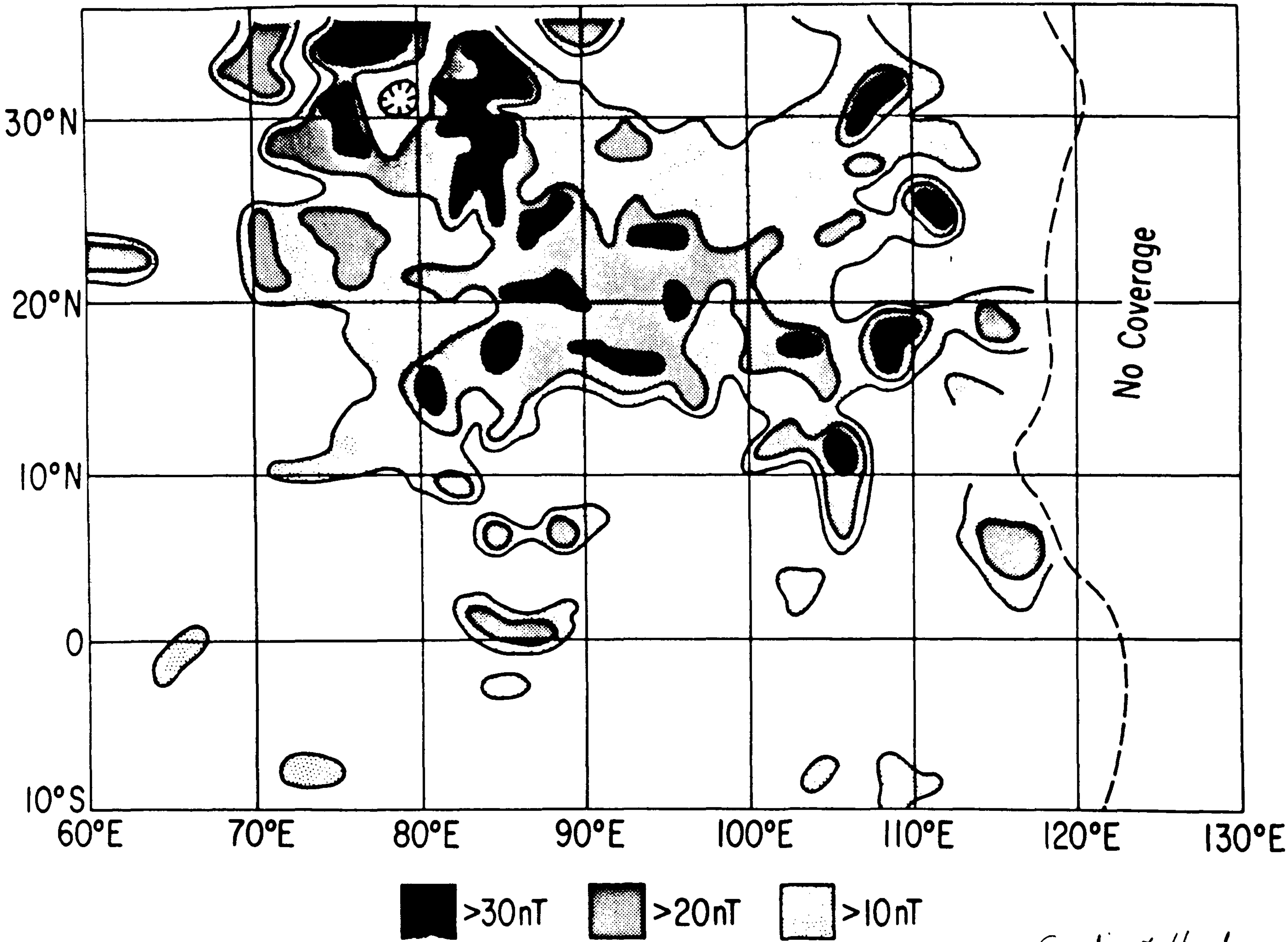

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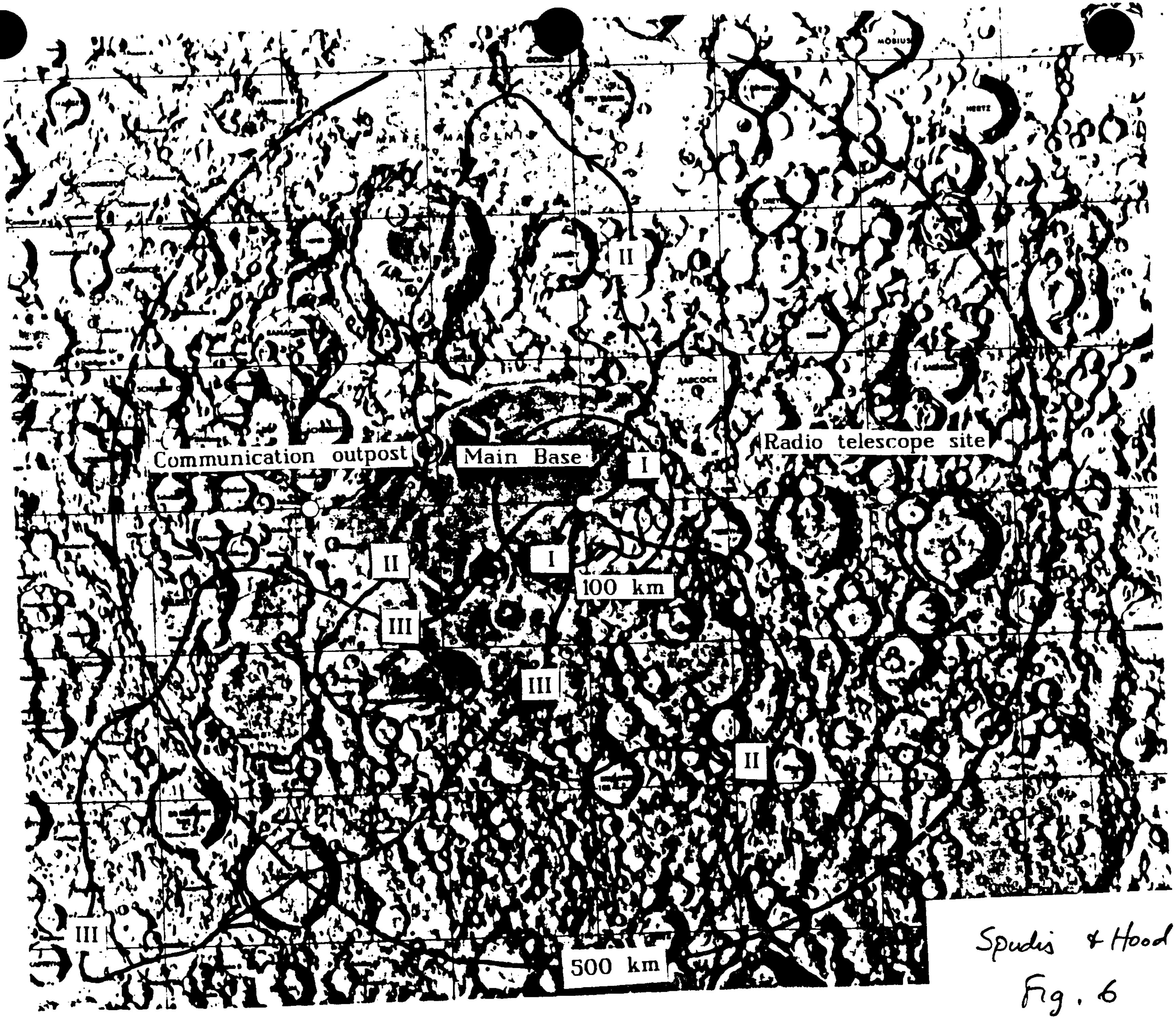

Spudis & Hood
Fig. 6

Proceedings of the Workshop on

A LUNAR FAR-SIDE VERY LOW FREQUENCY ARRAY

Held at: The BDM Corporation
Albuquerque, NM

Dates: February 18 - 19, 1988

Sponsored by: The National Aeronautics & Space Administration
The University of New Mexico
The BDM Corporation

Editors: Jack O. Burns
Nebojsa Duric
Stewart Johnson
G. Jeffrey Taylor

CONTENTS

PROLOGUE

On February 18th and 19th, 1988, a workshop was convened to discuss the scientific goals and preliminary designs of a potential Very Low Frequency Array (VLFA) to be constructed on the Lunar Far-Side. This two day meeting was conducted in the attractive, informal atmosphere of the BDM Corporation in Albuquerque. An attempt was made to gather together a small but representative group of astronomers who have participated in the construction of and observations with low frequency radio telescopes. The primary motivation behind the workshop was to seek guidance from our panel of experts on how we might plan for a VLFA on the Moon. We attempted to build upon previous foundations that were laid at other Lunar Base conferences and in other discussions of space-based very low frequency astronomy. We rolled our sleeves up, and asked some tough, detailed questions about the scientific justification for a lunar VLFA, the location and deployment of the array in the harsh environment of the lunar far-side, and the possible configuration of the antennas and their electronics. This report describes the results of our deliberations.

For the purposes of this workshop, we have defined very low radio frequencies to be < 30 MHz (> 10 meters wavelength). This is a practical definition for astronomical observations as discussed in this report. We do note, however, some inconsistency with radio frequency engineering that usually refers to very low frequencies as < 1 kHz.

The specific goals of the workshop were two-fold as follows:

(1) Define the Scientific Objectives of the Lunar Far-Side VLFA. We attempted to consider potential observations of the Sun, the magnetospheres of planets, the interstellar medium of the Galaxy, compact stellar objects, and active galaxies and quasars with the VLFA. Our efforts were restricted to defining in general what types of observations one might pursue for the purpose of guiding the design of the VLFA.

(2) Develop a preliminary design of the VLFA for further study. Among the areas of discussion were the frequencies for observation, the mode of operation (scan versus aperture synthesis), receiver and dipole design, computer requirements, data transmission, and antenna pattern and deployment. We reached a general consensus on these topics and have proposals that address these issues in this report.

We adjourned feeling satisfied that we accomplished our primary mission which involved the first serious discussions of a lunar VLFA. However, this workshop was only the beginning of what we hope will become a permanent working group on the VLFA. Many technical questions were posed that will require further study. A series of recommendations for future work are offered in the last section of this report. As with many issues involving a manned lunar base, pursuit of answers to these questions will require funding of the research scientists and engineers so that adequate time can be devoted to this work.

I would like to thank Mike Duke and Barney Roberts from the NASA Johnson Space Center for their on-going support of our studies of specific astronomical observatories on the Moon. A special thanks goes to Wendell Mendell for his participation in the workshop and for providing important motivation for a lunar base over the last five years. Finally, I would like to acknowledge Dr. Stewart Johnson and the BDM Corporation for providing us with marvelous facilities that helped to make this workshop a success.

Jack O. Burns
The University of New Mexico
June, 1988

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AGENDA

February 18, 1988

8:30 - 9:00	Overview & Introductory Remarks	J. Burns
9:00 - 9:30	Review of Lunar Environment	J. Taylor

Review of Low Frequency Observatories

9:30 - 10:00	Clarke Lake Telescope	W. Erikson
10:00 - 10:30	Texas Array	J. Douglas
10:30 - 10:45	Coffee Break	
10:45 - 11:15	Iowa State VLB Program	J. Basart
11:15 - 11:45	RAE Satellite Program	M. Kaiser
11:45 - 12:15	NRAO 75 MHz Array	R. Perley
12:15 - 2:00	No Host Lunch	

Science With a Lunar VLF Array

2:00 - 3:00	Effects of Interplanetary & Interstellar Scintillations (with discussion)	B. Dennison and/or W. Erikson
3:00 - 4:00	Round-Table Discussions on Science	N. Duric
4:00 - 4:15	Coffee Break	
4:15 - 5:30	Continuation of Round-Table Discussions	N. Duric
6:00 - 8:30	Dinner at Rio Grande Yacht Club	
8:30 -	After-dinner drinks & conversation at Duric's House	

Agenda (continued)

February 19, 1988

Lunar VLFA Design Considerations

8:30 - 9:30	Precursor Missions	J. Burns & J. Taylor
9:30 - 9:45	Coffee Break	
9:45 - 10:15	Strawman Proposal for Lunar VLFA	J. Burns & J. Basart
10:15 - 10:45	Potential Lunar Sites	J. Taylor
10:45 - 12:00	Round-Table Discussions on Designs	S. Johnson
12:00 - 1:30	No Host Lunch	
1:30 - 2:30	Further Discussions of Design	S. Johnson
2:30 - 3:00	Summary	J. Douglas

INTRODUCTION

Jack O. Burns
The University of New Mexico

In response to the recent report by Sally Ride (1987), NASA has begun to move aggressively into long-range planning beyond the Space Shuttle and the Space Station. A new Office of Exploration headed by John Aaron was recently added at NASA Headquarters to consider, among other things, how the U.S. might establish a permanent presence on the Moon and on Mars. With regards to a lunar base, a four phase scenario is being considered. This will hopefully serve as a generic template for implementation definition and for a schedule required for analysis. The four phases are as follows:

Phase I - Exploration and Base Site Selection

Phase II - Scientific Outpost

Phase III - Permanently Inhabited Base

Phase IV - Self-supported Base.

Some of the goals for each phase of exploration are shown schematically in Figure 1. More detailed facilities and a possible timeline for deployment are shown in Figure 2. These figures are courtesy of Mike Duke and Barney Roberts.

For the purposes of this report, we note that a far-side observatory is scheduled for Phase II, in the later half of the first decade of the 21st century. This would presumably include a VLFA that would likely be deployed with a minimum of human presence. At first glance, this timeline appears both optimistic and also a distant prospect for the future. Why should astronomers be concerned with such a futuristic observatory when we have yet to see the launch of the first of the Great Observatory Series, the Hubble Space Telescope, and have yet to secure funding for AXAF and SIRTF? The answer to this question lies in the long timescales required to develop support in the astronomical community and in Congress, and to develop good scientific and technical proposals for such space-based projects. The analogy to the Space Telescope may be particularly appropriate. Lyman Spitzer's first proposal for an Earth-orbiting large optical telescope appeared in print in the late 1940's, nearly 40 years ago. The first NASA-sponsored meetings began in 1962. Some 27 years later (1989), the Hubble Space Telescope is anticipated to be launched by the Shuttle. It is interesting to note that 27 years from the date of this workshop is 2015. Thus, it is not too early to begin to explore lunar observatories such as the VLFA on the far-side.

The idea of observatories on the Moon can be traced back to at least the mid-1960's (NASA Summer Studies on Lunar Exploration and Science, 1965, 1967). The idea has laid dormant since the end of the Apollo program in the early 1970's. It was revived in the early to mid-1980's as shown in Table 1.

Table 1. Some Recent History of Lunar Observatories Discussions

EVENT	YEAR
Field Committee Report	1982
Los Alamos Workshop	1984
Lunar Bases Symposium I	1984
Workshop on Future Astronomical Observatories on the Moon	1986
National Commission on Space Report	1986
Ride Commission Report	1987
Lunar Bases Symposium II	1988

In the Field Committee Report on Astronomy and Astrophysics for the 1980's, the last section discusses astronomical observatories on the Moon. It was noted that "The Moon offers certain decisive advantages as a base for astronomical observations. In particular, the far side of the Moon provides protection from radio interference from sources on or near the Earth and therefore has great potential value for radio astronomy". The report went on to recommend that the U.S. government "begin planning in the near future for the establishment of lunar observatories early in the next century".

This recommendation was followed several years later by a specific proposal by Douglas and Smith (1985) for the establishment of a very low frequency radio astronomy observatory on the Moon. This paper was presented at the first Lunar Bases and Space Activities of the 21st Century Symposium held at the National Academy of Sciences in Washington, DC. In this paper, Douglas and Smith propose "an extremely simple, low-cost Very Low Frequency radio telescope, consisting of a large (approximately 15 x 30 km) array of short wires laid on the lunar surface, each equipped with an amplifier and digitizer, and connected to a common computer. The telescope could do simultaneous multifrequency observations of much of the visible sky with high resolution in the 10- to 100-m wavelength range, and with lower resolution in the 100 toward 1000 m range." To a large extent, this paper provided the motivation and beginning design for the VLFA discussed in the present report.

The idea of a very low frequency array was further discussed by Jim Douglas at the workshop on Future Astronomical Observatories on the Moon held in Houston in January, 1986 (Burns and Mendell, 1988). It became clear at this workshop, that a VLFA on the lunar far side was one of the more interesting and important proposals for lunar observatories.

The National Commission on Space Report and the recently completed second Lunar Bases Symposium have further emphasized that the Moon is possibly the best location within the inner solar system from which to conduct astronomy, especially meter to kilometer wavelength radio astronomy. The Moon has several natural advantages over the Earth for low frequency radio astronomy. These include:

(1) The Moon absorbs Earth radio noise for a far side observatory. The Earth's environment is very loud at low frequencies. First, man-made interference from radio, television, and communications dominates on the surface of the Earth and leaks through the ionosphere at significant levels (Erickson, 1988). Thus, both on the Earth's surface and in Earth orbit, one is faced with high levels of interference. Second, the Earth's magnetosphere is a strong source of kilometric auroral radiation, especially below 1 MHz. This was first discovered in the early 1970's by the Radio Astronomy Explorer (RAE) satellite (see paper by Kaiser in Part II of this report). The Moon is a natural filter for these interfering signals.

(2) The Moon has very little ionosphere. On the Earth's surface, the ionosphere does not transmit radio frequency radiation below about 5 MHz. In addition, scintillation and scattering of radio waves by the ionosphere and troposphere limit both position accuracy and resolution of radio observations below about 30 MHz. The exact value of the global average electron density in the Moon's ionosphere is uncertain, but is believed to be low (< 100 electrons/cm^3). This corresponds to a plasma frequency of < 90 kHz which would be the lower bound on observations from the Moon. Other astronomical considerations discussed in this report would likely drive the lower bound on the frequency of the VLFA above 500 kHz. However, Vondrak (1988) has recently pointed out that a 100-m layer of negatively charged electrons and ions may hug the surface of the Moon on the day side. The electron levitation is believed to be due to the net positive charge of the surface produced by interaction with the solar wind. The density may be as high as 10^4 electrons/cm^3 corresponding to a plasma frequency of 1 MHz. Clearly, this must be investigated further since it would impact on the design of the VLFA.

(3) The Moon is a stable platform. This is an advantage over Earth orbiting spacecraft. On the Moon, the dipole array can be deployed over large areas with accurate relative positions for phase coherence. In orbit, gravitational strains constantly move the relative positions of array components, thus varying one's ability to observe all sources in the same fashion (i.e. with the same u-v coverage for aperture synthesis) or monitor source variability.

(4) The maintenance of a lunar VLFA will be very low. Once deployed, there will be no erosion of the cables and components by wind and rain. Although the temperature gradient from day to night is large (280 K), this is not expected to significantly alter the characteristics of the simple dipole antennas.

There are, however, some concerns that one has about placing the VLFA on the lunar far side. These include:

(1) The possible damage of integrated circuitry and computer chips in the receiver, correlator, and computers by cosmic rays. Since the Moon has no significant magnetic field, the surface is not shielded from the damaging cosmic radiation. However, radiation-hardened electronics are commonly produced for spacecraft that fly above the Earth's van Allen radiation belts.

(2) Human environmental hazards. The Moon is a very inhospitable place to live and work. Lack of oxygen and water, and radiation from the Sun and the Galaxy are major concerns. The deployment of the VLFA over tens of kilometers on the far side could potentially be very time intensive. Such concerns would appear to demand that the array be deployed by intelligent robots. This increases the technical complexity of an otherwise simple instrument.

(3) Remote basing on the far side. The likely location of lunar colonies for the first decade or two will be on the near side. Operations on the far side must be remotely controlled and semi-autonomous. In addition to the deployment problem noted above, one must also deal with the operation of the VLFA in a completely remote mode. Repair and upgrades of components, and initial processing and transmission of data must all be handled without human presence.

(4) Cost. Establishment of a lunar base is expensive. Sellers and Keaton (1985) estimate the cost will be about $80 billion in 1984 dollars. However, spread over 20 plus years, the yearly cost could be less than the Apollo program. Clearly, astronomy cannot bear this burden. This must be a national or international effort that is driven by political as well as scientific motivations. Once a permanent manned presence is established on the Moon, the cost of a VLFA is relatively low because of the simplicity and low mass of the components.

<u>Astronomy and Astrophysics for the 1980s</u>. Report of the Astronomy Survey Committee, Vol. 1, National Academy Press, 1982, pp. 165 - 166.

Burns, J. O. and Mendell, W. W. 1988, editors, <u>Future Astronomical Observatories on the Moon</u>, NASA Conference Publication #2489.

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<u>NASA Summer Conference on Lunar Exploration and Science</u>, 1965, NASA Report, pp. 369-391.

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Vondrak, R. R. 1988, in Lunar Bases and Space Activities of the 21st Century, Vol. 2, in press.

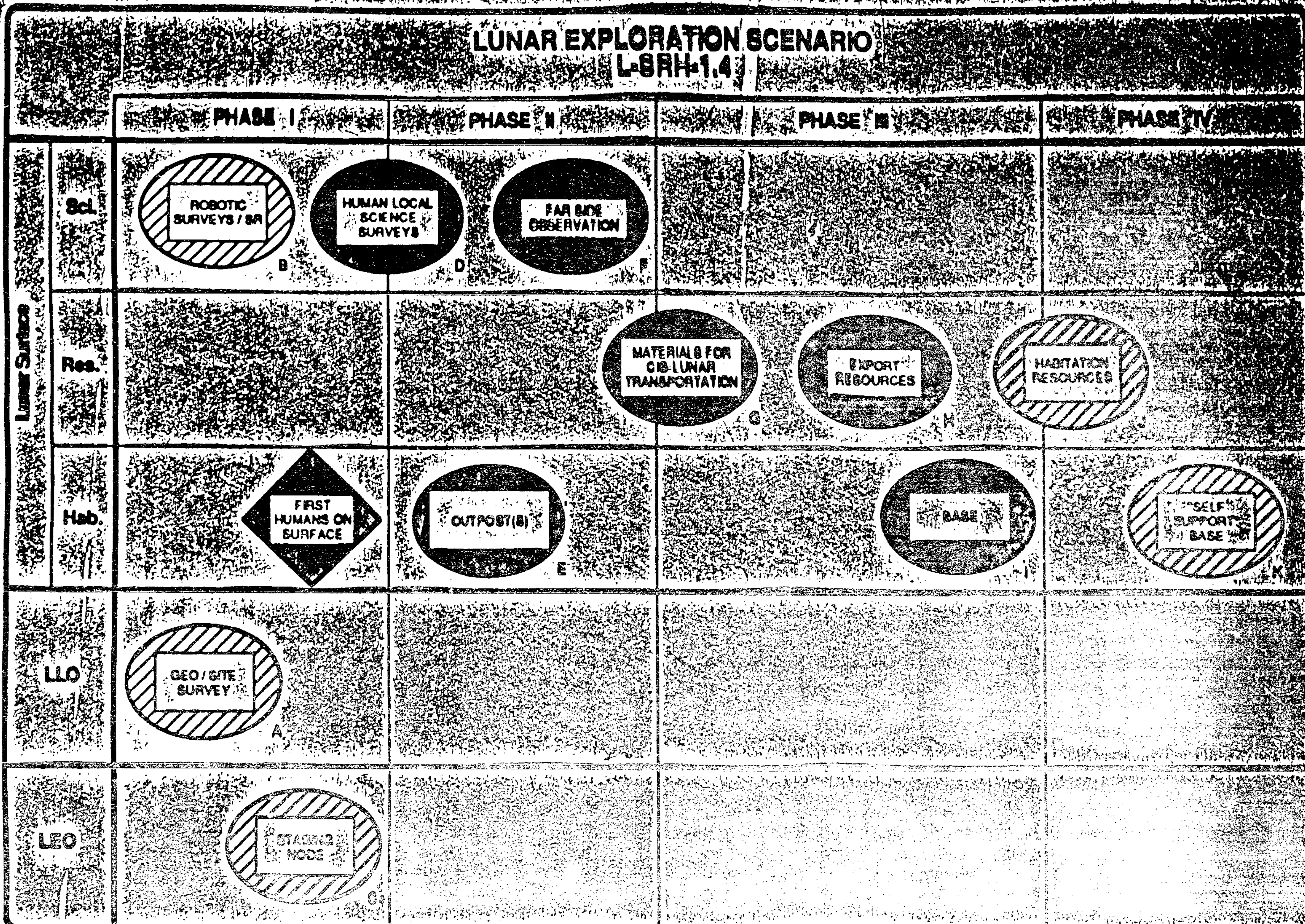

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Figure 1

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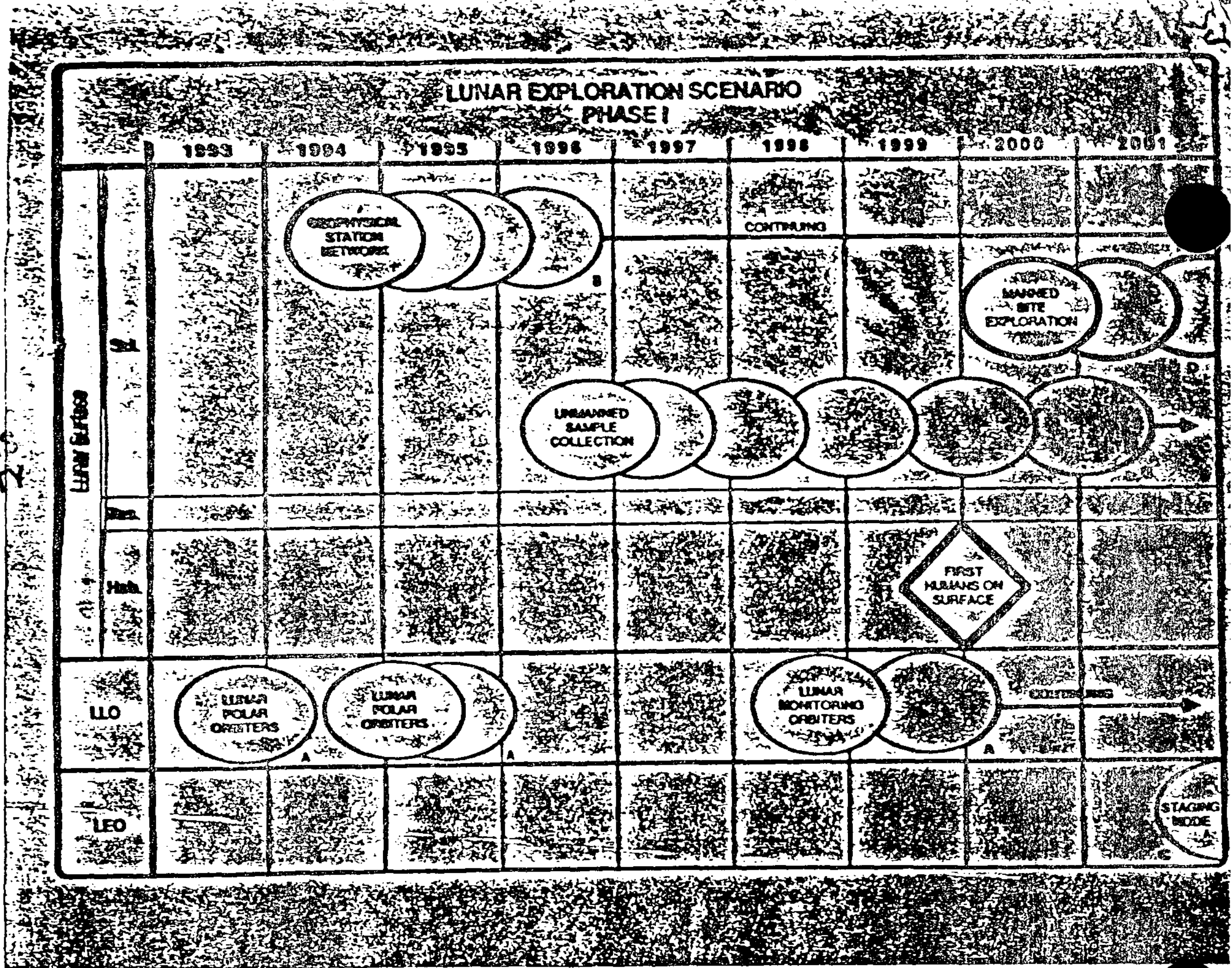

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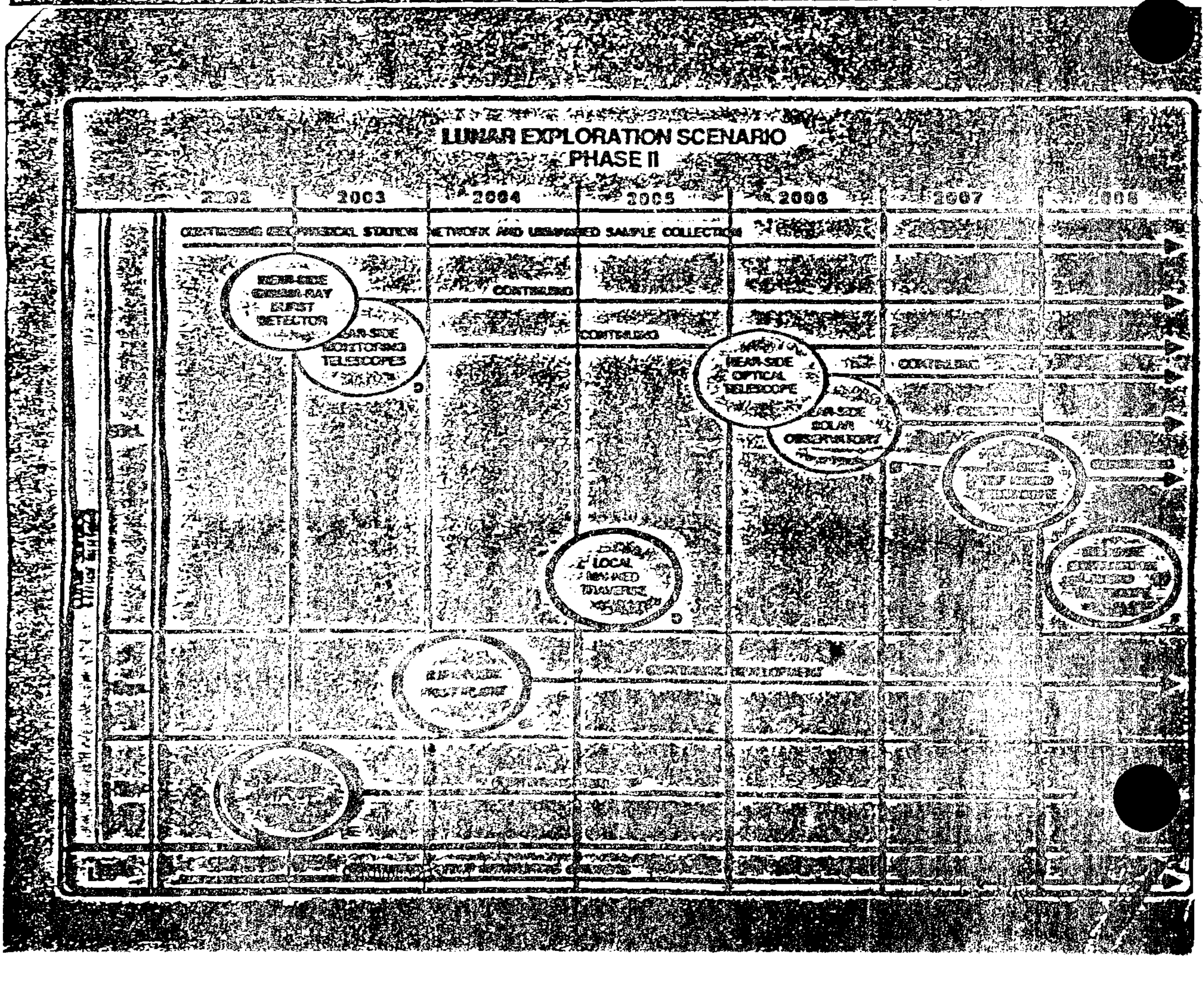
Figure 2

b)



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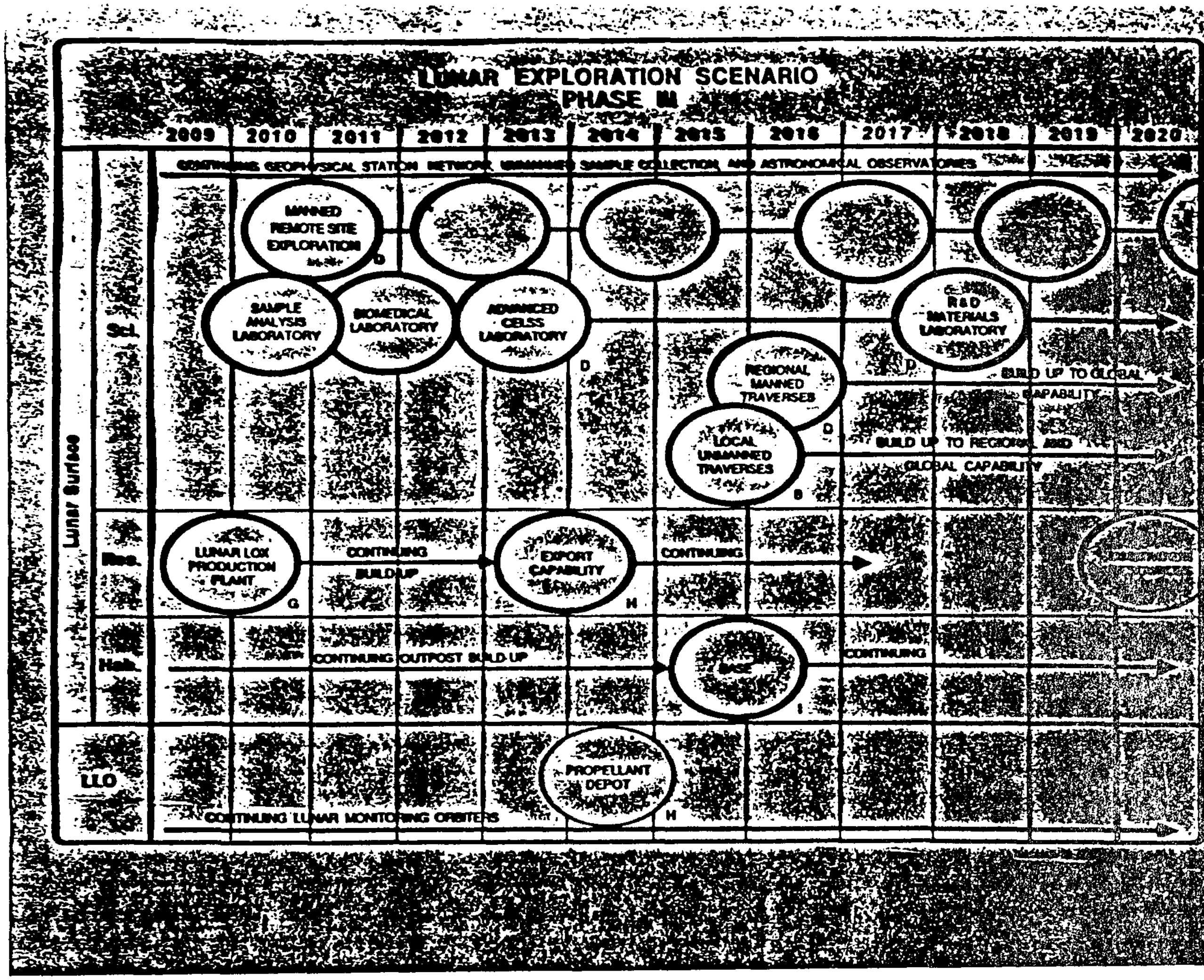

Figure 2 (CONTINUED)

PART I - THE ENVIRONMENT AT THE LUNAR SURFACE

THE ENVIRONMENT AT THE LUNAR SURFACE

G. Jeffrey Taylor

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Abstract

The Moon's geologic environment features: 1) A gravity field one-sixth that of Earth. 2) A sidereal rotation period of 27.3 days. 3) A surface with greater curvature than Earth's surface; a chord along a 60-km baseline would have a bulge of 260 meters. 4) A seismically and tidally stable platform on which to build structures and transportation systems; total seismic energy released is 10^8 times less than on Earth, and most moonquakes have magnitudes of 1 to 2, within the Earth's seismic noise. 5) A tenuous atmosphere (the total mass at night is only 10^4 kg) that does not cause wind-induced stresses and vibrations on structures. 6) A large diurnal temperature variation (100 to 385 K in equatorial regions), which facilities must be designed to withstand. 7) A weak magnetic field, ranging from 3 to 330×10^{-5} oersted compared to 0.3 oersted on Earth at the Equator. 8) A surface exposed to radiation, the most dangerous of which are high-energy (1-100 Mev) particles resulting from solar flares. 9) A high flux of micrometeorites which are not slowed down from their cosmic velocities because of the lack of air; data indicate that microcraters 10 m across will form at the rate of $3000/m^2/yr$. 10) A regolith 2 to 30 meters thick which blankets the entire lunar surface; this layer is fine-grained (average grain sizes range from 40 to 268 m), has a low density (0.8 to 1.0 g/cm^3 in the upper few millimeters, rising to 1.5 to 1.8 g/cm^3 at depths of

10-20 cm) is porous (35-45%), cohesive (0.1 to 1.0 kN/m^2), and has low thermal diffusivity (0.7 to 1.0 x 10^{-4} cm^2/sec) and electrical conductivity (10^{-14} $ohm^{-1}m^{-1}$). 11) A rubbly upper several hundred meters in which intack bedrock is uncommon, especially in the lunar highlands. 12) Craters with diameter-to-depth ratios of 5 if fresh and 15 km across; larger and eroded craters have diameter-to-depth ratios 5.

The Moon's geologic environment is dramatically different from Earth's and presents fascinating challenges to engineers designing facilities on the lunar surface. This paper summarizes the geologic nature of the stark lunar surface and its tenuous atmosphere.

GENERAL CHARACTERISTICS

The strength of the Moon's gravitational field is about one-sixth that at Earth's surface; the surface gravity is 162 cm/sec^2 and the escape velocity is 2.37 km/sec. The lower gravity allows use of materials of lower strength than on Earth for structures of equivalent size. Alternatively, much larger structures can be built on the Moon. The Moon has a slow sidereal (the time it takes to complete one revolution) rotation period of 27.3 Earth days, so days and nights last almost two weeks. Consequently, solar energy systems require some way to store energy and plants will need artificial light during the long lunar night. Finally, because of the Moon's smaller radius, its surface has a larger curvature than does the Earth's surface. For example, a chord along a 10-km baseline would have a bulge along it of 7.2 meters; a 60-km baseline would have a bulge of 260 meters.

STABLE PLATFORM

The Moon provides a stable platform on which to build. Seismic properties are summarized in Table 1, which is adapted from Goins et al. (1981). There are two main categories of lunar seismic signals, based on the depth at which they originate. Almost all occur deep within the Moon at depths of 700 to 1100 km; on average, about 500 deep events were recorded during the eight years that the Apollo seismic network operated. These deep moonquakes are related to tidal forces inside the Moon.

Moonquakes also occur at much shallower depths (200 km), but apparently below the crust (Nakamura et al., 1979). They occur much less frequently than do deep moonquakes, only about 5/year. Shallow moonquakes do not appear to be related to tidal flexing of the Moon or to surface features. For comparison, most earthquakes occur at depths of 50 to 200 km.

Lunar seismic activity is drastically less than terrestrial seismicity (Table 1). Lunar seismographs detected only 500 quakes per year. In contrast, 10,000 detectable earthquakes occur each year on Earth. Note that the magnitudes of detectable quakes is different on Earth and the Moon, due mostly to greater seismic noise on Earth. In fact, most moonquakes are in the magnitude 1 to 2 range, which is in the Earth's seismic background.

The total seismic energy released in the Moon is about 10^8 times less than in Earth. The magnitudes of the largest events on the Moon are also much less than the largest events on Earth (Table 1). The most energetic lunar events are the shallow ones, the largest recorded one being only 4.8 magnitude, corresponding to an energy of 2×10^{17} ergs. The largest recorded earthquake measured 9.5 magnitude on the Richter scale, corresponding to an energy of 10^{26} ergs.

Seismic waves are intensely scattered near the lunar surface. This causes the energy of the waves arriving at a given point to be spread out, so the damaging effects of a moonquake would be less than those of an earthquake of the same magnitude. (In fact, values of seismic energy and magnitudes reported for the Moon by Goins et al. (1981) are greater than those reported by Lammlein et al. (1974) because the latter authors had not accounted for scattering of seismic waves near the lunar surface or for some instrument effects.) Consequently, it appears that the lunar surface is far more stable than any place on Earth.

The scattering of seismic waves in the Moon is significant down to a depth of 25 km, but is most intense in the upper few hundred meters. This implies a lack of coherent layering in this region.

Tidal forces raise and lower the lunar surface about as much as on Earth, where body tides deflect the ground about 10-20 cm twice each day. Because the Moon is locked into a synchronous orbit, the main tidal bulge on the Moon is a permanent feature. Nevertheless, small tidal deflections stemming from librations do occur, but have much longer periods than on Earth. The tidal flexing of the lunar surface in both horizontal and vertical directions is about 2 mm along the length of a 10-km baseline (Dr. James Williams, personal communication, 1986). The precise amount of motion depends on position on the Moon. Tidal motions must be taken into account when designing, for example, long transportation systems or telescope arrays that require accurate alignment.

ATMOSPHERE

The lunar atmosphere is a collisionless gas. The total nighttime concentration is only 2×10^5 molecules/cm^3 (Hoffman et al, 1973). Its total mass only 10^4 kg, about the mass of air in a movie theater on Earth at 1 bar. This flimsy atmosphere will eliminate engineering problems associated with wind (Johnson, 1986), but might add others, such as difficulty in lubricating moving parts.

The composition of the lunar atmosphere appears in Table 2. The gases derive from the solar wind, except for ^{40}Ar, which is produced by the decay of ^{40}K inside the Moon and then diffuses out. No daytime measurements of gas concentrations were made due to instrument limitations, but Hodges (1976) calculates that gases of carbon compounds, specifically CO_2, CO, and CH_4, probably dominate. They are absent at night because they condense out of the atmosphere onto soil particles.

The tenuous lunar atmosphere can be altered significantly by large-scale operations on the Moon. Vondrak (1974) has calculated that if the density of the lunar atmosphere is increased, a point is reached where the rate at which gas is lost is slowed dramatically. This could compromise a number of scientific experiments requiring a hard vacuum, such as observations of the solar wind. Considering that each Apollo mission contributed 10^4 kg of gas (Johnson, 1971), temporarily doubling the atmosphere's nighttime mass, it would appear easy to contaminate the Moon's fragile atmosphere when regular flights to and from the lunar surface begin. The atmosphere must be monitored carefully when a lunar base is established. Studying the evolution of the Moon's atmosphere will, in fact, be an interesting research project in itself.

SURFACE TEMPERATURES

Surface temperatures change drastically from high noon to dawn on the Moon, presenting a challenge to those designing lunar structures subject to thermal expansion and contraction. At Apollo 17, for example, the temperature ranged from 384 K to 102 K during the month-long lunar day (Keihm and Langseth, 1973). Furthermore, the temperature decreases rapidly as sunset approaches, falling about 5 K/hr. These data apply to equatorial regions only. In polar regions, the predawn temperature is about 80 K (Mendell and Low, 1970). The temperature in permanently-shadowed areas at the poles could be lower. The cold nighttime temperature will permit cooling of many systems without use of cryogenics.

The temperature variation is damped out rapidly at depth in the lunar soil (Keihm and Langseth, 1973). At a depth of 30 cm the temperature is about 250 K and varies only 2 to 4 K from noon to dawn. This steady temperature might be useful for some purposes, but not as a heat sink because the lunar soil has a very sluggish thermal conductivity (see below).

MAGNETIC FIELD

No magnetic field is now being generated inside the Moon, although there was a source of magnetism several billion years ago. It is not known whether this was generated by a dynamo in a metallic core, as on Earth, or by local, transient events such as meteorite impacts. Whatever its source, the lunar magnetic field is much weaker than is Earth's (Dyal et al., 1974). On the surface, the lunar magnetic field strength ranges from 3 to 330 gamma (1 gamma = 10^{-5} oersted = 10^{-5} gauss). For comparison, Earth's field at the equator is 30,000 gamma. Also, the lunar field varies locally on the Moon. For example, at the Apollo 16 landing site, the field varied from 113±4 to 327±7 gamma.

There is also a field external to the Moon, derived from the solar wind. This ranges from 5 gamma in the free streaming solar wind to about 10 gamma in Earth's geomagnetic tail, in which the Moon resides 4 days during each lunation.

RADIATION ENVIRONMENT

Because of the Moon's small magnetic field and nearly absent atmosphere, solar and galactic nuclear particles hit its surface unimpeded. There are three sources of radiation with different energies and fluxes; see Taylor (1975) for a summary. 1) High energy (1-10 Gev/nucleon) galactic cosmic rays, with fluxes of about $1/cm^2/sec$ and penetration depths of up to a few meters. 2) Solar flare particles with energies of 1-100 Mev/nucleon, fluxes up to $100/cm^2/sec$, and penetration depths up to 1 cm. 3) Solar wind particles, which have much lower energies of about 1000 ev, tiny penetration depths (10^{-8} cm), but high fluxes (10^8/cm^2/sec). These penetration depths refer to the primary particles only. Reactions between them and lunar material cause a cascade of radiation that penetrates much deeper (Silberberg et al., 1985), up to several meters. The combination of high flux and energy make solar flare particles the most dangerous to people working on the lunar surface and to electronic devices deployed directly on the surface.

MICROMETEORITE FLUX

The lack of a significant atmosphere on the Moon allows even the tiniest particles to impact with their full cosmic velocities, ten to several tens of km/sec. This rain of minute impactors could damage some structures and instruments on the lunar surface. Almost all lunar rock samples contain numerous microcraters, commonly called zap pits, on surfaces that were exposed to space while on the lunar surface. Studies of lunar rocks (Fechtig et al., 1974) have revealed the average flux of projectiles over the past several hundred million years. However, data from the Surveyor 3 TV camera shroud returned by the Apollo 12 mission and study of Apollo windows (Cour-Palais, 1974) indicate that the present flux of particles smaller than 10^{-7}g, which are capable of making craters up to 10 microns across, is about ten times greater than that measured on lunar rocks. Study of louver material from the Solar Max satellite (Barrett et al., 1988) confirm that fluxes are greater now than the average of the past several hundred million years. Combining the fluxes of particles $< 10^{-7}$g measured on spacecraft with those $> 10^{-7}$ measured in Apollo rocks, one arrives at the flux estimate in Table 3.

REGOLITH

The lunar regolith, also called the lunar soil, is a global veneer of debris generated from underlying bedrock by meteorite impacts. It contains rock and mineral fragments and glasses formed by melting of soil, rock and minerals. It also contains highly porous particles called agglutinates, which are glass-bonded aggregates of rock and mineral fragments. Agglutinates are produced by micrometeorite impacts into the lunar regolith.

Regolith depth ranges from 2 to 30 meters, with most areas in the range 5 to 10 meters. Impacts by micrometeorites have reduced much of the regolith material to a powder. Its grain size ranges from 40 to 268 m and varies in a highly complex fashion with depth (Heiken, 1975). The chemical composition of the regolith reflects the composition of the underlying bedrock, modified by admixture of material excavated from beneath or thrown in by distant impacts.

The mechanical properties of lunar regolith samples were measured by Mitchell et al. (1972). The bulk density of the regolith is very low, 0.8 to 1.0 g/cm^3, in the upper few millimeters, but increases to 1.5 to 1.8 g/cm^3 at depths of 10 to 20 cm. Its porosity is 35 to 45% in the upper 15 cm, accounting in part for the low density. Except for the uppermost few millimeters, the lunar regolith is more cohesive, 0.1 to 1.0 kN/m^2, than most terrestrial soils and has an angle of internal friction of 30 to 50°. Agglutinates and shock-damaged rock fragments are weak and break under loads, leading to an increase in soil density (Carrier et al., 1973).

The lunar regolith is an excellent insulator. Its thermal diffusivity at depths of 30 cm is 0.7 to 1.0 x 10^{-4} cm^2/sec and its thermal conductivity is 0.9 to 1.3 x 10^{-4} W/cm K (Langseth et al., 1976). This is not surprising considering the high porosity and lack of air. At depths 30 cm, the thermal diffusivity is somewhat lower.

The lunar regolith is also an excellent electrical insulator. The dielectric properties of the regolith have been summarized by Olhoeft and Strangway (1975) and by Olhoeft (1988). For soils, conductivities are about 10^{-14} $(ohm\text{-}m)^{-1}$.

Lunar soils have dielectric constants ranging from 1.5 to 4, with a systematic variation with density: $k = (1.93 \pm 0.17)p$ where k is the dielectric constant and p is the density (g/cm^3). This relation holds for rocks as well. There is no systematic variation with soil composition (Olhoeft and Strangway, 1975), almost no dependence or temperature, and no dependence on frequency above 1 MHz. Loss tangents have also been measured on lunar materials. This quantity is strongly dependent on density and on composition, and somewhat dependent on temperature and frequency. The loss tangent is given by:

$$\log \text{loss tangent} = (0.38(\%\ TiO_2 + \%\ FeO) + 0.312p \ - 3.260)$$

where p is the density.

A small amount of lunar dust might be transported by charge differences built up by photoconductivity effects. Criswell (1972) described a bright glow photographed by Surveyor 7 and explained the phenomena as levitation of dust grains about 6 m in radius. The grains were lifted only 3 to 30 cm above the local horizon, and had a column density of 5 grains/cm^2. This does not appear to be a significant transport mechanism on the lunar surface.

UPPER FEW HUNDRED METERS

The upper few hundred meters of the Moon have been intensely fragmented by meteorite impacts. In the heavily cratered highlands and regions underlying mare basalt flows, the fragmental region extends for at least a few kilometers. Consequently, it might be difficult to find extensive areas of intact bedrock.

Active seismic experiments (Cooper et al., 1974) indicate that the velocity of compressional waves is about 100 m/sec at depths of less than 10 meters, which is in the regolith, and about 300 m/sec at depths between 10 and 300 meters. These velocities are too slow to correspond to coherent rock, implying that the upper few hundred meters of the lunar surface is rubble (Cooper et al., 1974). Rocks returned from the highlands confirm the fragmental nature of the upper lunar crust. Most are complicated mixtures of other rocks, and many are weakly consolidated. Furthermore, the rims of all craters are by their nature weakly or unconsolidated materials and, therefore, not able to withstand tensional stresses.

A few localities might have intact bedrock, however. Many mare basalt flows, for example, form visible layers in crater walls or, as at the Apollo 15 landing site, in the walls of sinuous rilles. Also, extensive sheets of impact-generated melt rocks occur on the floors of many large craters, such as Copernicus, which is 95 km in diameter.

CRATER MORPHOLOGIES

Fresh lunar craters up to 15 km in diameter have a consistent diameter/depth ratio of 5 (Pike, 1974). More specifically, craters 15 km across follow the relation $R_i = 0.196\ D_r^{1.010}$; craters 15 km follow the relation $R_i = 1.044\ D_r^{0.301}$ where R_i is the crater depth and D_r is the diameter as measured from rim crest to rim crest (Pike, 1974). Large craters are much shallower for their diameters than are smaller ones. Crater morphology changes as a crater is eroded by meteorite bombardment, during which a crater becomes wider and shallower, thereby increasing the diameter-to-depth ratio. Thus, even the smoothest areas on the lunar surface are undulating plains, so building horizontal transportations systems might require cut and fill operations. Finally, as noted above, rim materials consist of weak, unconsolidated rock. This could present problems for construction if certain facilities had to be built on crater rims.

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MICROMETEORITE FLUX

Crater diameter (m)	Craters/m²/yr
0.1	300,000
1.0	12,000
10	3,000
100	0.6
1000	0.001

PART II - PREVIOUS AND PROPOSED LOW-FREQUENCY OBSERVATORIES

THE CLARK LAKE TELESCOPE

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INTRODUCTION

Most radio astronomical observations below 100 MHz have been the result of considerable effort on the part of a relatively small number of astronomers and engineers. The size required for the instruments has precluded the construction of more than a few in the world and, until recently, technology has not allowed the design of a large decametric array which would operate over more than a limited frequency range and be steerable in two coordinates with reasonable speed. Consequently, this part of the radio spectrum has attracted very few astronomers even though much information about the physics of celestial objects may be found from the study of radiation from the cosmos at these low frequencies.

Advances in the technology of decade bandwidth antennas [Rumsey, 1966], and low cost, reliable, wideband, solid state devices have made large fully-steerable decametric systems practical. Around these developments, the design of the Clark Lake telescope evolved during the 1970's. It was operable anywhere between 10 and 125 MHz with nearly instantaneous frequency and beam positioning capability. Unfortunately, recent cuts in federal funding have forced us to discontinue operation of the system and it has now been dismantled.

OUTLINE OF THE ARRAY AND ELECTRONICS

The array was a 3.0 x 1.8 km "T" with the direction of its legs being approximately east, west, and south. The array was laid out in the plane of the Clark's dry lake which is not exactly tangential to the geoid. The south arm, which was perpendicular to the E-W arms, was laid out 18 arcsec from the plane containing the earth's axis and the center of the array. The EW arm contained 32 banks, each with 15 individual elements; the N-S arm contained 16 similar banks. The signals

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from each of the EW banks were cross-correlated with those from the N-S banks to determine the two-dimensional visibility function of the area of sky under observation. This visibility function was then Fourier transformed to produce a map of the field-of view. The beam shape of the three-armed "T" is equivalent to that obtained with a full cross but the collecting area of the fourth arm is lost. Phase tolerances between the orthogonal arms are more critical in the "T" array than in a full cross [Christiansen and Hogbom, 1969], but these phases were easily adjusted in software after calibration by the observation of strong, small angular diameter sources.

Each log spiral element [tepee, hence the nickname "TPT"] had a collecting area of about (wavelength)^2/3, and was designed to operate between 20 and 125 MHz. The low frequency limit was extended to ≈10 MHz at reduced efficiency by terminating the base of each spiral with resistors. The tepees were at 6.25 meter intervals in the E-W arm and 7.5 meter intervals in the south arm. This spacing gives rise to grating responses above 50 MHz. The response due to these grating lobes was reduced by adjusting real-time delays to make the radiation coherent only for the desired lobe. Since the elements were all fixed in the vertical direction, beam positioning was accomplished purely through adjustment of the phase gradient across each arm. The gain of the system was modulated by the response pattern of the individual elements. For good zenith distance coverage, the response pattern must be wide (≈90° in this case), and the gain of each element was correspondingly low.

As shown in Figures 1 and 2, phasing of the array was accomplished in two stages. The elements were divided into 48 banks of 15, and the signals from the 15 antennas in each bank were combined, then preamplified and sent to the central building on separate coaxial feed lines. Phasing within a bank was accomplished by electronically "rotating" each conical spiral antenna with a diode switch controlled from the central building. Phasing between the banks was adjusted in software before the map of the field of view was formed by Fourier transformation.

SINGLE ELEMENT CONSTRUCTION AND OPERATION

The basic building block of the array was the conical log spiral antenna. Ideally this antenna would consist of two conducting sheets wound on the surface of a cone but, in this case, these sheets were approximated by sets of wires. The antenna was self-conjugate with a characteristic impedance of 189Ω and it was a backward- wave antenna fed by a balanced transmission line at its apex.

The electrical and radiative properties of this antenna were as follows: The antenna radiates primarily from the region

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OF POOR QUALITY

where the circumference of the spiral is approximately equal to the wavelength, thus low frequencies are radiated from near the base of the element and high frequencies are radiated from its top. The low-frequency limit of the antenna is determined by the size of the base of the spiral (circumference ≈ wavelength) and the high frequency limit is set by the point at which the top of the spiral is truncated. This low frequency limit was extended by terminating the base of the spiral with a resistive load. Power that would ordinarily be reflected from the base of the antenna was dissipated in the load, and a constant antenna impedance was maintained to very low frequencies. At low frequencies, however, power is absorbed in the terminating resistors rather than being launched into space and the antenna efficiency decreases. Therefore the low frequency limit on the operating frequency was set by the loss of efficiency that one could tolerate rather that by impedance mismatches. Since the galactic background is intense at low frequencies, the background noise dominated receiver noise to below 10 MHz and the system could be operated down to about that frequency. The radiation pattern and the circularity of the polarization remained the same at these frequencies below the nominal cut-off of the antenna.

The radiation is unidirectional toward the apex and the polarization is in the opposite sense from the opening direction of the spiral, i.e., a right-hand (clockwise opening) spiral as viewed from the top radiates predominately a left circular wave. The far field radiation pattern is determined by the apex angle of the cone and the pitch angle of the conductors. More details may be obtained from other sources [Rumsey, 1966; Dyson, 1965; Yeh and Mei, 1967, 1968].

In actual practice the use of conducting sheets is very difficult because of cost and wind resistance for large antennas. A good approximation to a conducting sheet can be made by using three wires, one at the location of each edge of the conductor and one in the center. Thus the elements in this array used six coaxially wound spiral wires, three connected to each side of the transmission line.

Each element in the array was phased by electrically rotating it in 45° increments. Antenna rotation was a practical phasing scheme in this array because the polarization remained nearly circular in all directions observed. The conical spiral antennas have this property between the half-power points of their radiation patterns (±45°). Antenna rotation was accomplished by winding the spirals with eight instead of six wires and a diode switch was devised to select six of the wires at any given time. With the simplicity of this phasing scheme comes the disadvantage of not having continuous rotation. The phase error of any element can be as much as 22.5° due to incremental phasing. However, these phase errors caused only rather minor sidelobes [Erickson and Fisher, 1974].

Although the antenna was designed to operate only to zenith distances of 45°, it was found in practice to operate quite well to considerably larger angles. In fact, several objects were well-observed only 15° above the southern horizon.

ELEMENT GROUPING AND PHASING SCHEME

The phasing within each bank is accomplished by rotating each element, so there is no real time delay added to the signals from individual elements. All time delays are added to the 48 signal paths in the central building.

The use of simple phasing as opposed to delays in the 15-element banks limits the size of banks due to coherence loss with wide receiver bandwidths. Each bank was approximately 100 meters in length. This resulted in a coherence loss of 9% with a bandwidth of 3 MHz at a zenith distance of 45°. The loss was normally much less with smaller bandwidths and zenith distances.

TP IMPEDANCE CHARACTERISTICS

Because the conical spiral elements in this array incorporated several features which have not been tried before, it was imperative that impedance and radiation characteristics be investigated before building 720 units. For practical reasons the antenna impedance could not be measured directly. There was a length of cable, a transformer, and the phase switch between the impedance bridge and the antenna terminals. Since we were interested in the operation of the total system, the standing wave ratio (VSWR) and impedance measured through these components are perfectly valid provided the power loss in the individual components is not more than about 20%.

Significant stray reactances in the feed system arise due to the physical layout of the diode switch inside the central support pipe. These reactances were measured and compensated with small inductors incorporated in the switch. The combination of stray and lumped reactances forms a nearly symmetrical low pass filter with a cutoff frequency of about 250 MHz in series with each antenna wire. Impedance measurements were made at the base of the TP and corrected for delay feeder cable. The maximum VSWR was 1.4:1, which corresponds to a reflected power loss of 4%. Ohmic losses in the transformer and switch were less than 1 db (20%) and the loss in the feeder cable was 0.4 db (8%) at 110 MHz. The characteristic impedance of the antenna was close to 189Ω, the theoretical value for a self-conjugate antenna.

The eight wires were wound around a support system that consisted of eight parallel filament, dacron ropes. The ropes were protected by polyethylene jackets. The N-S arm of the array is shown in Figure 3.

ELECTRONICS SYSTEM

A separate receiver channel was attached to the output of each of the 48 banks. Each channel employed an up conversion from the frequency being observed to 170 MHz. This conversion places all image frequencies well above the frequency range of observation. After some amplification, the signals were converted to 10 MHz where the principal amplification occurred. Four IF bandwidths, ranging from 3 to 0.15 MHz, were selectable. A diagram of these channels is shown in Figure 4.

The 10 MHz output of each receiver channel was sampled at a frequency of 12 MHz, digitally delayed, and then cross-correlated. The correlator outputs from the 512 simultaneous interferometers were preintegrated for periods from 10 millisec to 10 sec -- periods short enough that the phase rotation for a source moving at a sidereal rate is negligible on any of the interferometer baselines--and after each preintegration period, the digital data were written on magnetic tape. Later, an off-line processor was used to remove the phase rotation and to integrate the signals for periods of up to 5 minutes. A Fourier transform then produced a map of the area of sky under observation. These maps may be averaged to effectively integrate the signals for periods of hours. A detailed description of the electronics system is to be found in von Arx, Caflisch, and Erickson (1978). The specifications of the system are summarized in Table 1.

TABLE 1

TPT SPECIFICATIONS

Parameter	Value
Frequency range	10-125MHz
Instantaneous bandwidths	0.15-3 MHz
Total collecting area	250
Resolution	
20 MHz	17'
110 MHz	3'
Steering and frequency changing time ..	1 millisec
Sky coverage	≈45° zenith distance
Sensitivity (and confusion limit)	1 Jy at all frequencies
Polarization	left circular

DATA PROCESSING

The TPT was different from most other synthesis-type radio telescopes in that all the Fourier components from a minimum spacing of about one wavelength to the maximum aperture of the system were available simultaneously. Also, the telescope operated in a frequency range where terrestrial interference is very common. It appeared to be necessary to develop procedures to reject low level interference after formation of the maps, as well as to reject obvious interference in the visibility-plane data. Therefore, rather than averaging the visibility-plane data for hours, then gridding them and transforming them, we transformed the data frequently and averaged the selected maps [Erickson, Mahoney, and Erb, 1982]. Since maps were formed frequently, this processing scheme is also more appropriate for observation of rapidly varying solar emission regions.

The length of time over which we could integrate the visibility-plane data before Fourier transformation was limited to ≈5 minutes; longer integrations would begin to smear the visibility data because of the rotation and foreshortening of the arrays caused by Earth rotation. The T-shaped antenna provides data automatically in a 32x32 grid; we do not project them onto the (u-v)-plan or regrid them before transformation. In practice, a transformation was performed and a new integration was begun whenever the phase gradients across the banks of elements were updated to follow the source under observation. Maps were generally produced at intervals of 1 second to 5 minutes. Use of longer intervals facilitates the processing speed but requires that more data be discarded when interference occurs.

OPERATION

The system operated very well. All of the design specifications listed in Table 1 were met or exceeded in practice. Only one area of problems turned out to be somewhat more troublesome than anticipated. We encountered nearly constant, low-level interference which appeared to come from a variety of sources. Much of this interference was apparently man-made radio noise reflected at glancing incidence by the ionosphere or diffracted over the mountains surrounding the telescope site. Interference from the extremely strong natural sources, Cas-A and Cyg-A, also prevented sensitive observations in their vicinity, i.e. within about 10° of their positions. Solar radio bursts could interfere with daytime observations. Strong interference was easily recognized and rejected from the raw data. However, much effort was expended in developing sophisticated algorithms to recognize and excise low-level interference that could not be found until the signals from the banks were cross-correlated and time integrated. In practice this meant that we were unable to reach the confusion limit of the system as quickly as we anticipated in its design. It should have been possible to reach the confusion limit ($\approx$1 Jy) with only a few minutes of integration; because it was necessary to excise much of the data and to average over interfering sources, about 30 minutes integration was required to reach the confusion limit.

The confusion limit depended upon the direction of observation. In simple regions near the Galactic poles we were able to reliably observe sources down to a flux density of 0.5 to 0.7 Jy, in complex regions along the Galactic plane we could only work down to about 2 Jy. In any event, the system was 10 to 50 times more sensitive than any other existing or planned telescope in this frequency range. For a variety of reasons [Erickson and Fisher, 1974], the confusion and sensitivity limits of the system were fairly independent of frequency.

The system was used as a multifrequency radioheliograph for solar studies and as a synthesis telescope for sidereal studies. As a radioheliograph it was used to determine the radio signatures of coronal mass ejection events, to show that Type III emitting electron streams propagated in dense coronal streamers, to discover meter-decameter microbursts, to determine the three dimensional structure of coronal streamers, to measure coronal electron densities on a routine basis, and to determine the spectrum and brightness distribution of the quiet sun. In the field of sidereal astronomy Clark Lake observations lead to the discovery of the first millisecond pulsar and to the discovery of the millisecond pulsar in M28, many supernova remnants were mapped, HII absorption was studied to determine the synchrotron emissivity of the Galaxy, a steep-spectrum radio lobe near the Galactic center was discovered and the Galactic plane was mapped. About one-third of the sky was surveyed with unprecedented sensitivity and resolution before we were force to discontinue operation. These survey data are now being analyzed.

FIGURE CAPTIONS

Figure 1. The layout of the array. Signals from each bank of elements were amplified and transmitted to the observatory by separate coaxial cables.

Figure 2. One bank of 15 conical spiral antennas.

Figure 3. A photograph of the N-S arm of the array.

Figure 4. Block diagram of one of the 48 channels in which signals from the 15-element banks were processed.

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Fig 1

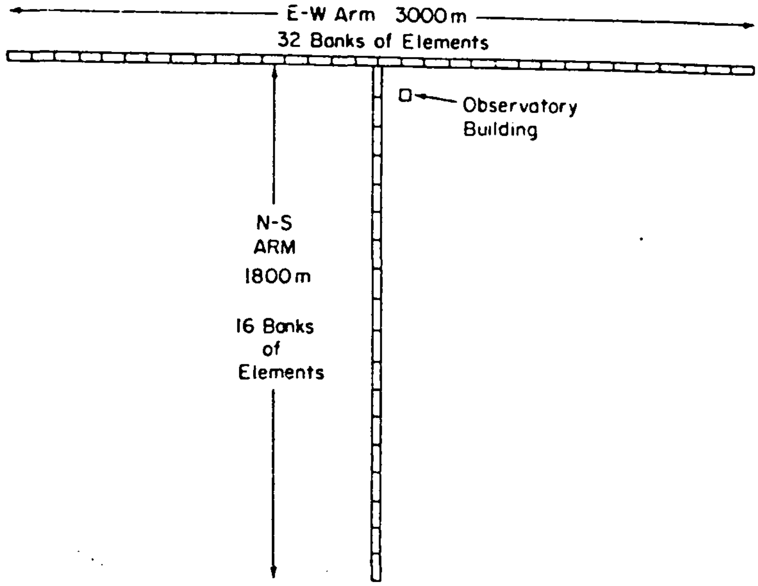


Fig 2

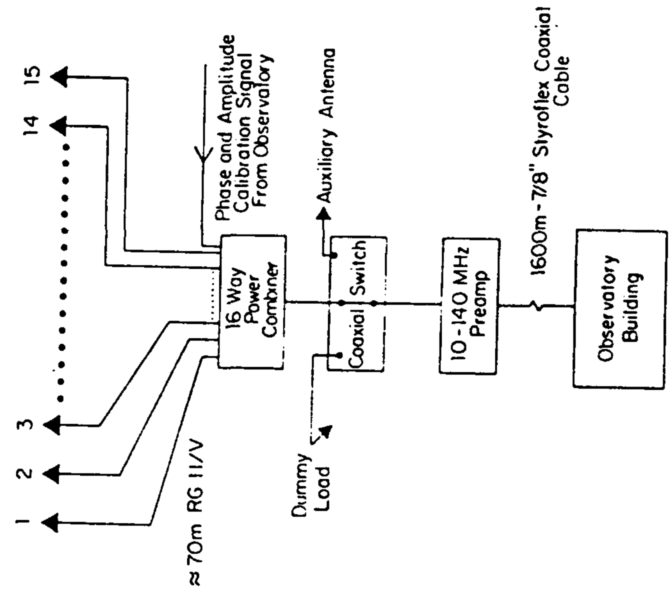
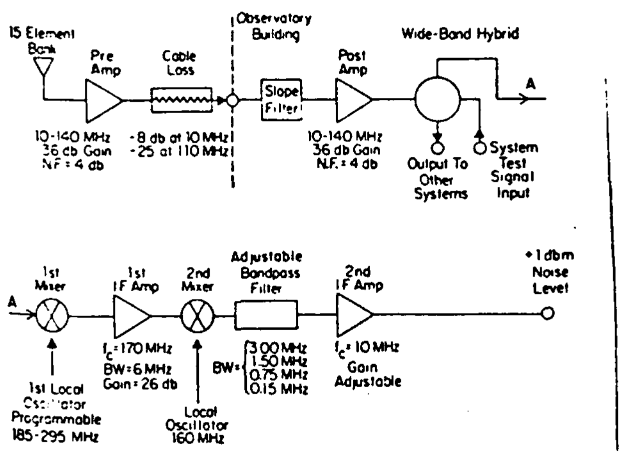


Fig 4



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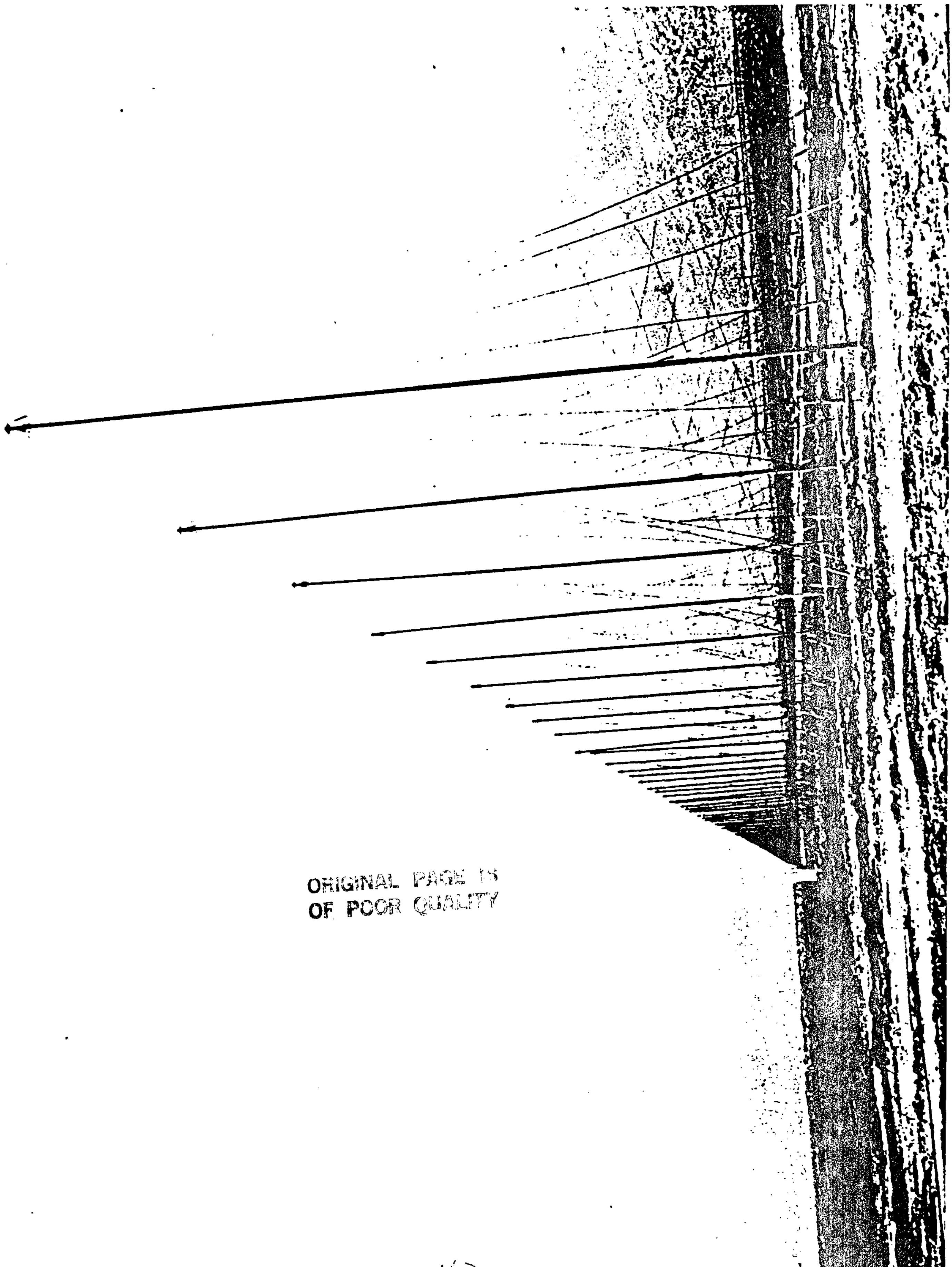

THE BOULDER-AMES DECAMETER WAVELENGTH VLBI EXPERIMENT

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ABSTRACT.

This paper reviews the VLBI experiments at 26.3 MHz conducted in the early 1970's between Boulder and Haswell, Colorado, and between Boulder and Ames, Iowa. The longest baseline was 83000 wavelengths. While it was possible to get fringes on this baseline, the reliability of the data was very poor because of the extreme amount of ionospheric-induced phase and amplitude fluctuations. Clearly, in order to obtain reliable data, very-long-baseline interferometers operating at frequencies of 26 MHz and lower must be placed above the ionosphere.

INTRODUCTION.

The objective of the VLBI experiments at a decameter wavelength was to determine source structure in order to identify radio sources of small angular extent to be used in future interplanetary scintillation experiments at low frequencies. The observing source list consisted of point-like sources, or sources with point-like components, with measured or extrapolated flux densities greater than 20 Janskys at 26 MHz. Additionally, their declinations were between -2° and 68°. Sources with right ascensions in the range 8^h to 14^h were omitted because of competing solar noise and interplanetary scintillation effects. The observing list consisted mostly of extragalactic sources in the 3C catalog plus a few supernova remnants, pulsars, the sun, Jupiter, and Saturn (Shawhan et al. 1973). Sources with similar right ascensions, but quite different declinations, were divided into two observing schedules to accomodate constraints on manually phasing the antennas. Each schedule consisted of an 18-hour program starting at about 6:00 p.m. local time and stopping at noon with the last runs on the sun. Each day 3C48 and 3C144 were observed as calibrators.

INTERFEROMETER DESCRIPTION.

Two different interferometer baselines were used in three series of experiments. In November 1970 and July 1971, telescopes in Boulder and Haswell, Colorado were used. The baseline length was 22,000λ at a position angle of 135°. The fringe width was 9 arc seconds and the 3σ detection limit was about 20 Jy. In August 1971 the Boulder and Ames, Iowa telescopes were used. This interferometer had a baseline of 83,000λ at a position angle of 76°. The fringe width was 2.5 arc seconds with about the same detection limit as the Boulder-Haswell baseline.

The antennas in all three telescopes consisted of 160 pairs of crossed full-wave dipoles in rectangular arrays. However, the Ames configuration was different than that of the other two antennas. The Ames rectangular array geometry was 10 dipole pairs east-west by 16 dipole pairs north-south. The Boulder and Haswell arrays were 20 dipole pairs east-west by 8 dipole pairs north south. The reason for the difference in geometry was that the Boulder array was phased along its local meridian while the Ames array was phased to the west. Moving the array beam off center broadened it, hence, the shorter east-west array dimension made the Ames beam more similar to the Boulder beam.

Crossed dipoles were used in the array to avoid Faraday fading problems in the ionosphere. The orthogonal antennas were connected together to be receptive to left-elliptical polarization. (The choice of LEP was arbitrary.) Most of the observations were made in a southerly direction so we oriented the dipoles in a way to keep the signal level high. The crossed dipole pairs were aligned in the northeast-southwest and southeast-northwest directions rather than north-south and east-west. This eliminated a possible low-amplitude response by observing "off the end" of a north-south dipole.

The interconnections of antenna elements in a large array is always a problem. The method used here was to connect all antennas in one east-west row to one transmission line. This was repeated for each E-W row. There were two complete and identical configurations for each polarization of the crossed dipoles so only one polarization will be explained. In the Ames array, there were 16 rows of east-west transmission lines for each dipole polarization. Ten dipoles of the same polarization were connected to each E-W transmission line. The Colorado arrays had 8 rows for each polarization with 20 dipoles per row. There was no impedance matching of each antenna to its transmission line. The input impedance of full-wavelength dipoles is relatively high compared to the transmission line. Hence, the antenna-to-line coupling is low. This reduces the amount of signal transfered to the line, which is a negative factor, but it also loads the line very little, which is a strong positive factor. We can get by with small coupling between the antenna and the transmission line at these low frequencies because the signal level is high.

The interconnection of the E-W rows was done with a branch (corporate) feed. The first step in the interconnection was to connect each two adjacent E-W rows together with an impedance-matched network. This resulted in eight connected pairs. Then, each of these two adjacent pairs were connected by an impedance-matched network. This pairs-of-pairs connecting scheme was repeated until there was just one transmission line containing the output of the entire array. The final transmission lines from each polarization were connected together with an extra quarter-wavelength section of cable to form a left-elliptically polarized signal which then went to the receiver.

During the observing runs, all antennas were operated in transit mode by manually phasing the branch feed networks in the north-south direction before each observation and then letting the source drift through the beams. The phased position of the beams was determined the the equations,

$$\sin \xi = \cos \delta \sin H$$

$$\sin \eta = \sin \phi \cos \delta \cos H - \cos \phi \sin \delta$$

where ξ is the angle from the zenith to the west, η is the angle from the zenith to the south, and ϕ, δ, and H are the latitude, declination, and hour angle, respectively.

The receivers were conventional radio-telescope type superheterodynce receivers with extra radio-frequency filtering to reduce out-of-band interference. The bandwidths were 500 kHz and noise figures were approximately 2 dB. The recording terminals used at each site were standard Mark I terminals (Moran 1976) borrowed from the National Radio Astronomy Observatory.

DATA ANALYSIS.

As much as possible, the data analysis proceeded along the standard path for VLBI analysis in that era. Additional difficulties were encountered because of the very high phase noise, and the uncertainty about the detection of fringes. The signal-to-noise ratio of the fringes was always low. The often desirable 5σ threshold criterion was reduced to 3σ. To increase the reliability of the detection, three more criteria were used. One criterion was to put +5, and then -5, microseconds delay offsets about the presumeably correct delay and let the correlation program search for fringes once more for each offset. If a detection had been originally hypothesized and the residual delay after these two additional offset searches was consistent with the origninal residual delay, the criterion was satisfied. The second criterion was to compare two or more detections from different days. The result of this test was not always positive because the ionosphere varied so much from day-to-day. However, if the day-to-day results were consistent, the test was considered to be satisfied. The third criterion was to have a residual fringe rate near zero. Since the source positions were well-known by high-frequency measurements, the fringe frequency was predictable. If the residual fringe frequency was near zero, this criterion was satisfied.

The resulting correlated fluxes for any particular source were scattered from day-to-day because of variations in the ionosphere over the two antenna locations. All multiply observed sources for all observing runs were placed on one calibration grid of normalized flux density versus time. Each flux density was normalized by the mean observed flux density for that particular source. This plot showed systematic changes in flux with time which were contributed to ionospheric variations. Correction factors were obtained from the systematic changes in the grid of sources and applied to the individual measurements. Rates of change of trends in the observed fringe amplitude due to the ionosphere ranged from zero up to 16% per day. If amplitude changes were large and abrupt, the data were likely not to be used at all because of extreme uncertainties in how to correct it (or because it may not have been real). After applying corrections to the grid of sources, the rms scatter about the normalized mean was 8%.

Flux calibration was difficult because no source initially appeared to be unequivocably unresolved. The initial procedure was to estimate the apparent observed size of the Crab Nebula pulsar using the VLBI measurement of Mutel, et al. (1974). To estimate the scattered size in the interplanetary region, we used the formula of Erickson (1964)

$$\theta_s = 0.0649\ P^{-2}\lambda^{-2} \text{ arc second}$$

where P is the closest distance the radio wave passes by the sun in astronomical units and λ is the wavelength in meters (11.4 in our case). For an average solar elongation angle of 65°, $\theta_s = 1.77$ arc sec. This was combined in quadrature with the interstellar scattering size of 1.30 arc sec. (Mutel, et al., 1974) to give an estimated apparent size of the Crab pulsar of 2.20 arc secs. Assuming a total flux density of 800 Jy for the Crab pulsar at 26.3 MHz gave a visibility of 0.064 for a Gaussian model. After adjusting the Boulder-Ames data to comply with this visibility, a check was made with 3C48. The estimated observed size of 3C48 using the Boulder-Ames data was 0.60 arc sec. (Gaussian model). With this size, the predicted Boulder-Haswell visibility was 0.98 which was in reasonable agreement with the measured visibility as calibrated by the Crab pulsar. With consideration of the data on the Crab pulsar and 3C48, plus some consistency checks with 3C43, we set the flux calibration using the apparent size of 3C48 as 0.60 arc sec. with a circular Gaussian brightness distribution. The total flux density was set equal to 37 Jy as given by Viner and Erickson (1975).

RESULTS.

As an illustration of the results, Tables I, II, and III are presented. These data should not be taken as definitive. The extreme noisiness caused by the ionosphere permeated the analysis so thoroughly that it was impossible to calculate reliable error bars. Most of the total flux densities were from Viner and Erickson (1975). The uncertainties in these fluxes range from 5% to 18%. Errors in the visibilites contain the errors in the total flux densities plus, say, 25% more error due to ionospheric variations in visibility measurements.

Table I contains sources detected on all three experimental runs between Boulder and Haswell, and Boulder and Ames. Table II contains sources detected on the Boulder-Ames baseline and one Boulder-Haswell run. Finally, Table III contains sources detected on the Boulder-Haswell baseline only. Dashes in this table indicate that these sources were not scheduled for observation. The consistency of the results is remarkably good considering the noisiness of the data. For the Boulder-Haswell 1971 results, the average rms variation of multiple observations of the same source was 3% and for BA it was 8%. A comparison of the eight sources in common on the two Boulder-Haswell runs shows deviations varying from 0% to 47% between the two epochs.

Interpretation of the data was very limited in the early 1970's. At the most we had two points in the uv plane (other than the origin) from our data. High-resolution mapping at centimetric wavelengths was only beginning so we had little knowledge of the source structure at any wavelength. We used available information from scintillations, occultations, and interferometry at all radio wavelengths. We essentially were constrained to testing the agreement between our data and the published parameters of single and double-component sources. The agreement was tested by using our uv values in models reported in the literature and then comparing the predicted visibility with our measured visibility.

DISCUSSION.

We detected 36 out of a possible 49 radio sources on one or both of the baselines. From a simple comparison of our measured visibilities with the predicted visibilities using models determined from higher-frequency data, we found that in a large majority of cases, for which comparisons could be made, our data were consistent with those models. Wilkinson, et al. (1974) had discussed the consistency of some sizes over a frequency range from 408 to 2695 or 5000 MHz. Our results suggested that this consistency extended down to 26 MHz, an overall factor of 100 or more in frequency. Within the accuracy of our consistency checks, we found no straight-spectrum source with a component that was optically thick at 26 MHz.

Our experiments showed that it is possible to find correlations in 26 MHz data collected over an 83,000λ baseline. But considering the overall effort of erecting and maintaining the telescopes, collecting the data, and calibrating and interpreting the data, very little hard information came from a large amount of work. The cause of the minimal amount of information deduced from the data was the ionospheric variation. At times, the ionosphere caused the signals to completely disappear. At all other times, the signals were severly disturbed in amplitude and phase. Another contributing factor to significant loss of data was lightning. Lightning located many miles away disrupted the correlations. During the summer season, lightning often occurs, with or without an accompanying storm. It is very clear, that to make regular quantitative high-resolution observations at frequencies of 26 MHz and lower, the antennas must be above the ionosphere.

ACKNLOWLEDGMENTS.

Other principal investigators in this project were Stanley Shawhan, Thomas Clark, and William Erickson.

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TABLE I

SOURCES DETECTED ON THREE EXPERIMENTS

		BH November 1970			BH August 1971			BA August 1971		
Source	Assumed 26 MHz Total Flux, Janskys	Vis	Solar Elongation Angle,°	No. of Obs.	Vis	Solar Elongation Angle,°	No. of Obs.	Vis	Solar Elongation Angle,°	No. of Obs.
3C48	37	0.98	152	2	0.98	107	4	0.81	112	3
3C66	149	0.30	152	2	0.16	88	2	0.12	99	2
3C123	828	0.048	160	2	0.027	64	2	0.030	78	2
3C144 (compact component)	800	0.86	149	7	U.D.	52	2	0.064	65	8
3C456	92	0.76	119	1	0.79	141	1	0.40	152	2
3C459	90	0.52	118	2	0.69	144	1	0.47	158	3

U.D. means Unrealiable Data

TABLE II

SOURCES DETECTED ON B-A AND ONE B-H EXPERIMENT

		BH				BA August 1971		
Source	Assumed 26 MHz Total Flux, Janskys	Vis	Solar Elongation Angle,°	No. of Obs.	Date	Vis	Solar Elongation Angle,°	No. of Obs.
4C13.01	28*	0.88	123	1	August 71	1.05	137	1
3C9	91	0.77	125	2	August 71	0.33	137	3
3C71	73	0.72	159	1	November 70	0.65	122	3
3C196	218	0.52	115	1	November 70	0.080	43	3
3C295	72	No Det	67	1	August 71	0.25	63	1
3C345	34	No Det	95	3	August 71	0.73	89	3
1645+17		U.D.**	106	2	August 71	54.3 Jy	95	3

* Determined by extrapolation.
** U.D. means Unrealiable Data.

TABLE III

BOULDER-HASWELL SOURCES

Source	Assumed 26 MHz Total Flux, Janskys	November 1970 Vis	November 1970 Solar Elongation Angle,°	November 1970 No. of Obs.	August 1971 Vis	August 1971 Solar Elongation Angle,°	August 1971 No. of Obs.
3C2	56	0.80	160	2		U.D.**	
3C16	65	0.26	140	1	0.19	124	1
3C23	56	0.86	144	1	0.85	117	2
3C33	222				0.19	118	3
3C43	46	0.70	153	2	0.70	107	2
3C55	88	0.26	158	1			
W3					31 Jy	81	2
3C147	39	0.64	138	2	-	-	-
3C153	42	0.63	135	2	-	-	-
3C154	119	0.28	140	2	-	-	-
3C175	132	0.20	124	1	-	-	-
3C181	57	0.42	121	1	-	-	-
3C186	54	0.70	120	3	-	-	-
3C190	64	1.22	113	2	-	-	-
3C191	46	0.97	112	1	-	-	-
3C196.1	167	0.62	105	2	-	-	-
3C208	83	0.34	101	2	-	-	-
3C336	64	-	-	-	0.42	101	1
3C380	271	-	-	-	0.076	105	1
PSR1919		-	-	-	35 Jy	133	1
3C409	381	-	-	-	0.12	138	2
3C432	59	-	-	-	0.61	147	1
3C446	55	-	-	-	1.0	141	1

A dash means no scheduled observation.
** U.D. means Unreliable Data.

THE RADIO ASTRONOMY EXPLORER PROGRAM VALUABLE LESSONS FOR FUTURE LOW FREQUENCY RADIO ASTRONOMY FROM SPACE

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The Radio Astronomy Explorer (RAE) program of the late 1960s and early 1970s represents mankinds first attempt to perform low frequency radio astronomy measurements from above the earth's ionosphere with a dedicate set of instruments. A review of the results of this program have been given recently by Kaiser [in Radio Astronomy From Space, NRAO, 1986], so only those lessons learned that have direct application to a possible lunar low frequency radio observatory will be presented here.

Figure 1 is a drawing of RAE-2, which was placed in circular orbit around the moon in 1972. RAE-1, launched into a 6000 km altitude orbit around the earth, was very similar. The spacecraft were gravity gradient stabilized and an active libration damper was used to try to remove excess yaw due to deviations from a perfectly spherical orbit. Both spacecraft were equipped with two sets of 229-meter long Vee configuration antennas oppositely oriented, one set pointing radially upward and one set pointed toward the center of gravity, namely, the earth or the moon. A third antenna system, a short dipole, bisected the two Vees.

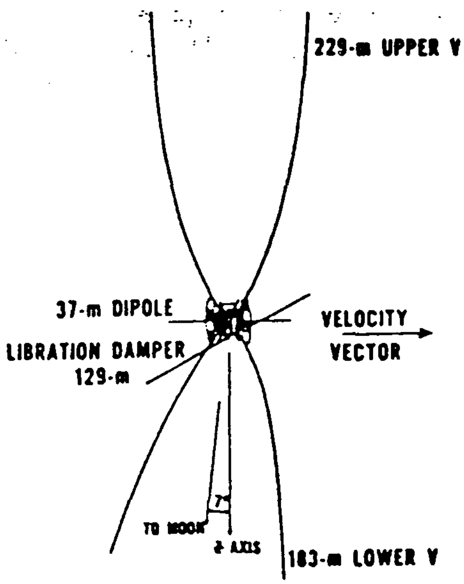


Figure 1

Connected to these antenna systems were two basic types of radio receivers, Ryle-Vonberg and total power(burst), both operating in the 25 kHz to 13 MHz range. The basic difference between these two types of receivers from a practical point of view was the way in which their preamplifier sections operated. The total power receivers were driven by a wide bandwidth preamplifier section covering essentially the entire operating frequency range, and with significant gain even outside of the nominal range. The Ryle-Vonberg receivers, on the other hand, were made up of relatively narrow band pre-amp sections, one for each operating frequency. A lesson learned very early in the life of the RAE program was that the receivers using wide bandwidth preamplifiers, although easier and cheaper to build, suffered severely from distortion due to strong signals anywhere in their passband, even at frequencies not directly sampled by the receiver. More on this will be mentioned later in connection with the terrestrial emissions observed by the RAE spacecraft. The resulting power pattern of the receiving system was not good by terrestrial standards. At frequencies of a few MHz, typical beam widths were of the order of a steradian with significant side and back lobes.

The major scientific achievements of the two RAE spacecraft fell into several categories including solar physics, planetary non-thermal emissions concentrating primarily on the earth's auroral kilometric emission (AKR), in situ plasma physics, and cosmic background mapping and spectra. In fact, the study of solar type II and III radio bursts and the study of AKR were extremely successful, accounting for more than half of all the ~60 scientific papers published from the RAE program. However, the galactic background studies were, at best, only marginally successful due to a combination of the poor angular resolution and strong inteference from AKR and other signals of terrestrial origin (e.g. thunderstorm sferics and manmade).

The effects of the terrestrial noise spectrum were alleviated for RAE-2 because of its distance (60 R_E) form the source. However, even this large attenuation was not enough as can been seen in Figure 2 where four months of data from RAE-2 in lunar orbit are shown. At the two lower channels shown, 40 kHz and 290 kHz, AKR

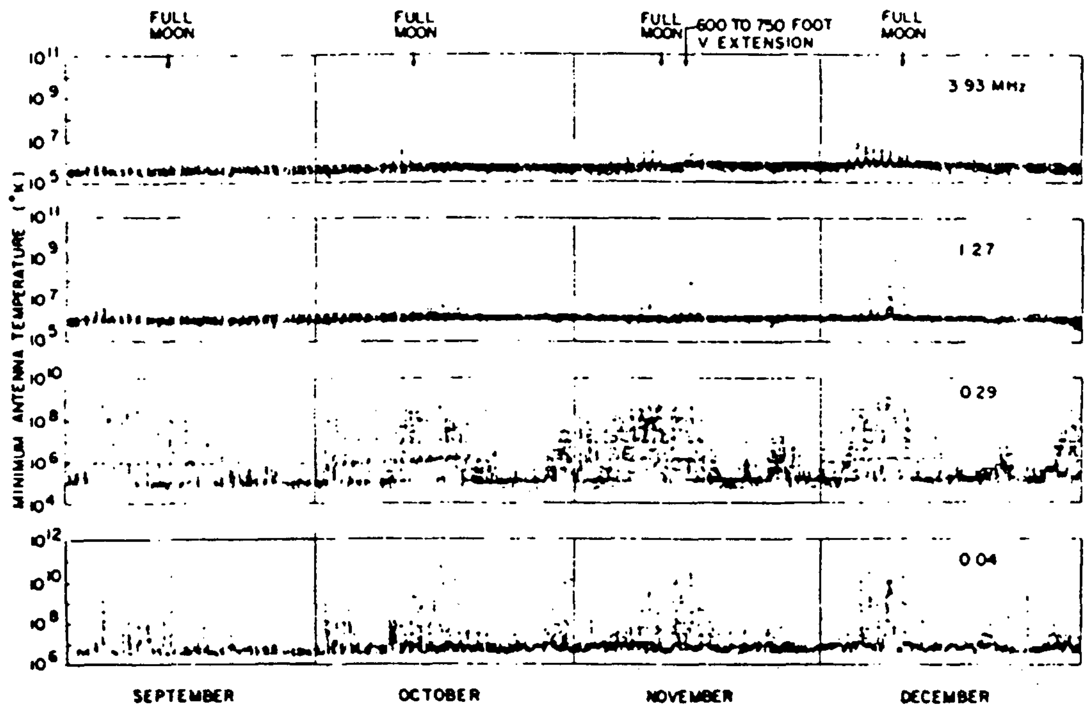


Figure 2

dominates the emission spectrum, at times reaching saturation levels and frequently causing receiver intermodulation products throughout the entire operating frequency range stemming from the wide band pre-amp mentioned above. The AKR maximizes at full moon, which corresponds to RAE being above the midnight sector of the earth, directly in the main beam of AKR. Even at frequencies of 1.27 and 3.93 MHz, well above the natural band of AKR (50 kHz to 750 kHz), terrestrial effects are important, especially over the night hemisphere. This higher frequency noise is a combination of thunderstorm sferics from the whole "visible" hemisphere, manmade broadcast stations and intermodulation from the AKR.

Perhaps an even better appreciation of the dominance of AKR can be obtained from Figure 3, again observed by RAE-2 in lunar orbit. In the top panel is a dynamic

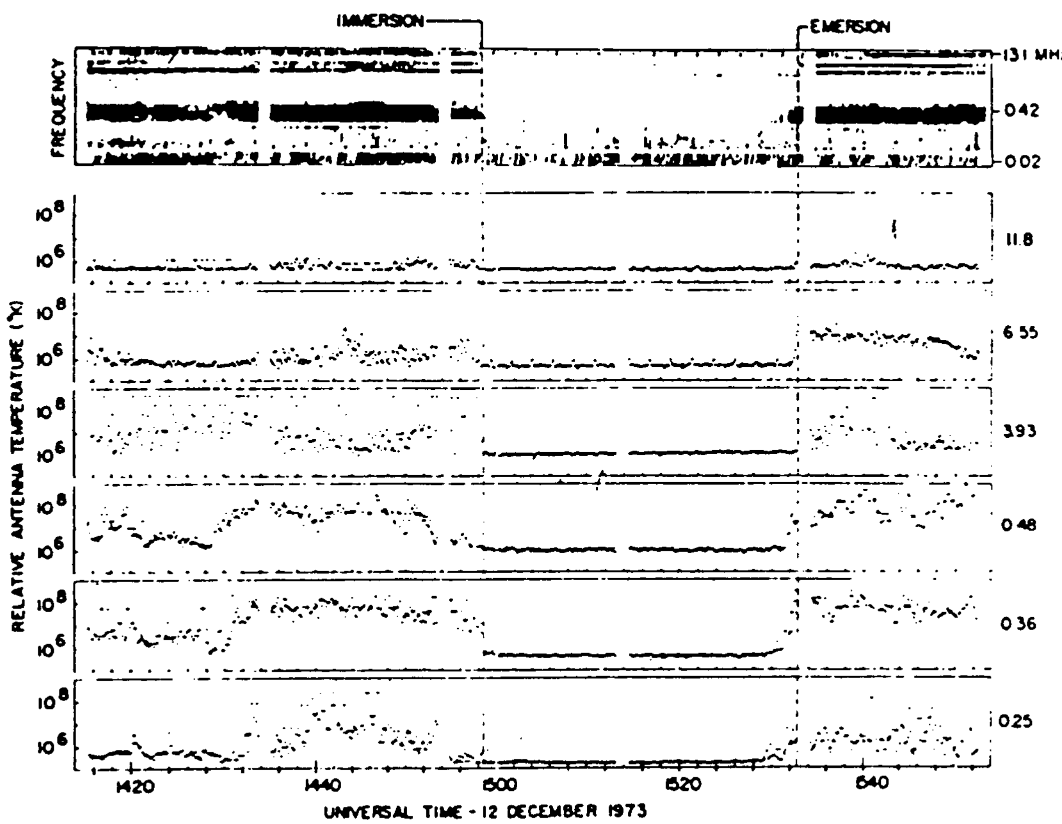
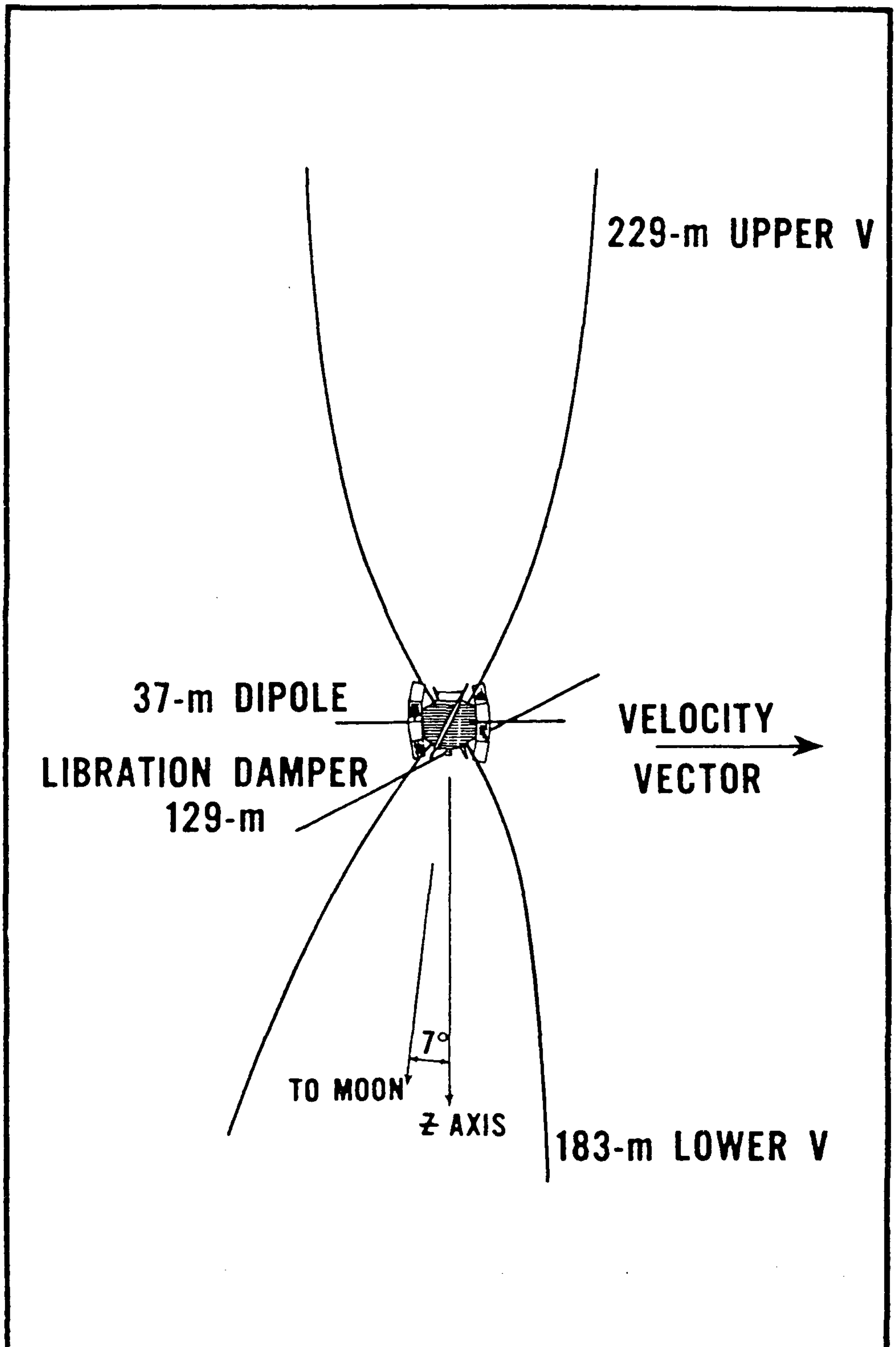


Figure 3

spectra showing received power as a function of frequency and time. The black band in the middle (near 400 kHz) is AKR and the dark strips near the top are other terrestrial signals. Just before 15:00, the AKR and terrestrial signals are abruptly cut off and do not reappear until about 15:30. This interval corresponded to the time when RAE-2 was above the back side of the moon so that the earth was occulted by the lunar disk. In the bottom pannels are individual frequency channels where one can see that this occultation effect is extremely dramatic. This single figure represents probably the stongest reason for placing a lunar low frequency observatory on the far, well away from the terminator.

In summary, the RAE program gave us two very valuable lessons for use in any future low frequency observatories. First, do not use wide bandwidth preamplifiers or receivers. Specifically, avoid the AKR frequency range. Second, avoid direct view of the earth itself, because AKR and other terrestrial noise, all of which is many orders of magnitude above cosmic background, will greatly hinder observations of intrinsically weak radio sources. side



229-m UPPER V
37-m DIPOLE
VELOCITY
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183-m LOWER V

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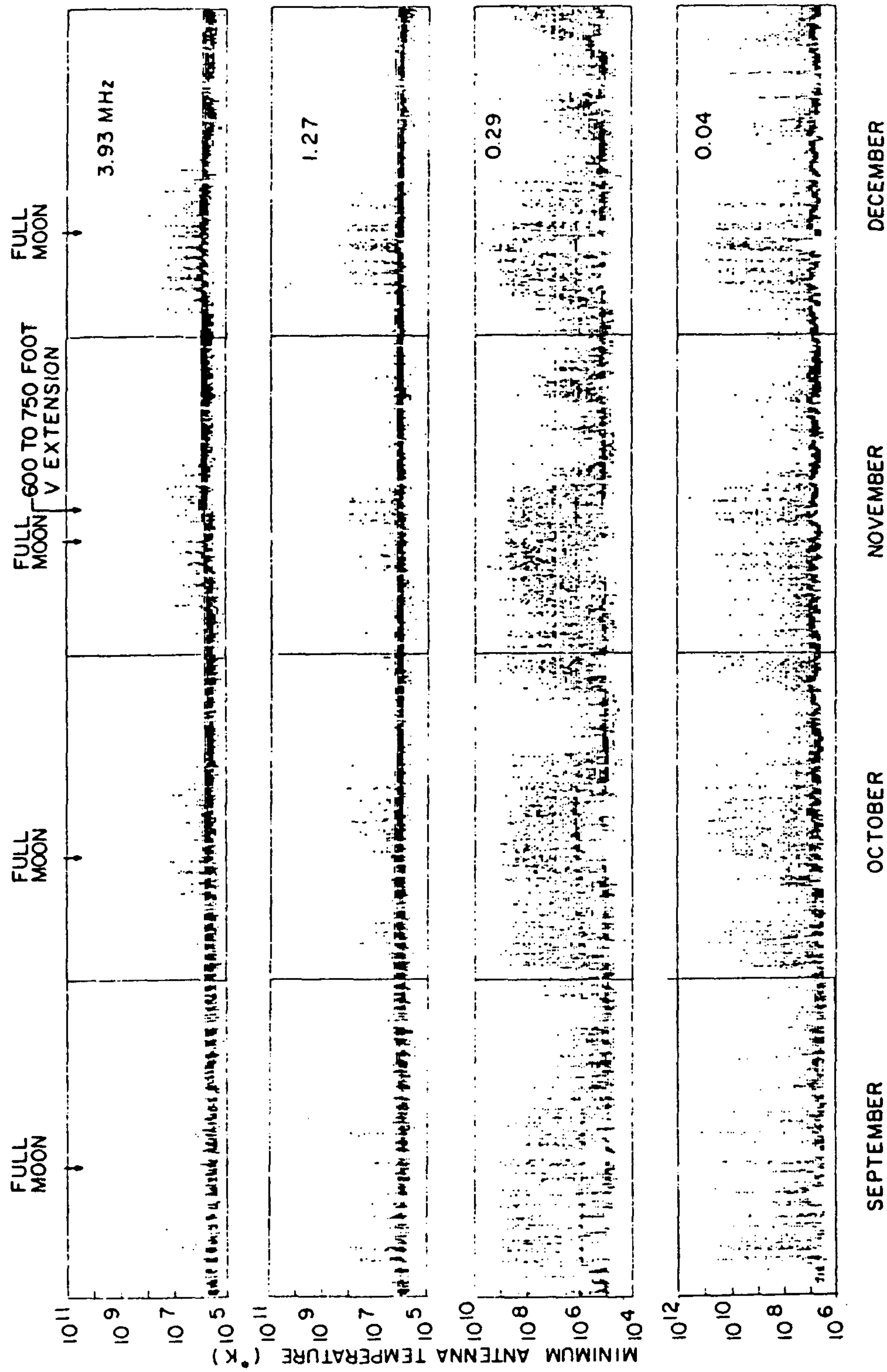

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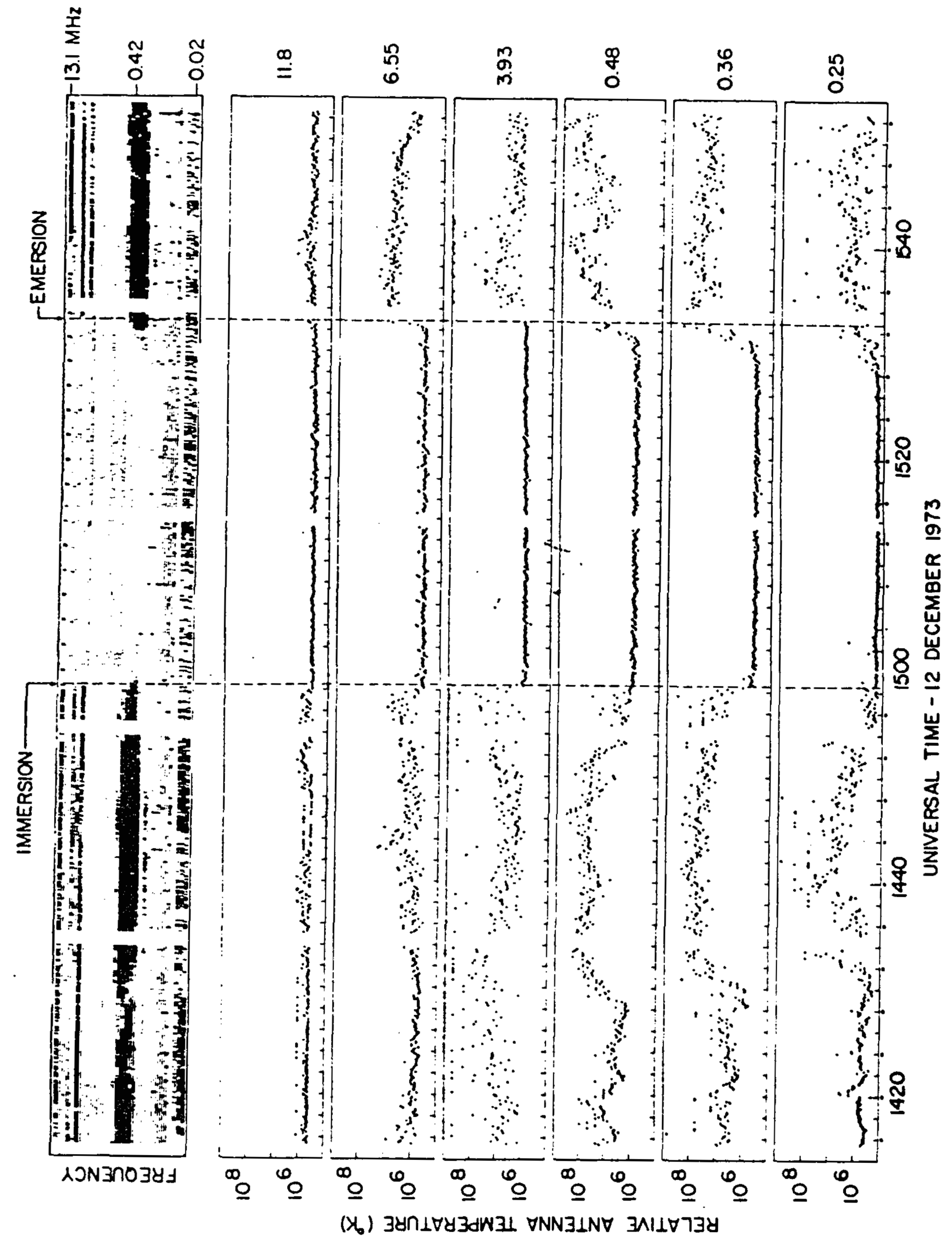

THE 75 MHZ VLA SYSTEM

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INTRODUCTION

The diffraction-limited resolution of radio astronomical instruments has encouraged the development of interferometric techniques. The technique known as 'earth-rotation synthesis' has been utilized by numerous instruments (notably the VLA), and has proved immensely successful in providing full-field mapping of celestial radio emission with resolutions orders of magnitude better than that provided by the largest single antenna.

However, the use of earth-rotation synthesis has been almost wholly limited to centimeter wavelengths. At longer wavelengths, (>>1 meter), the only instruments which utilize the technique (the University of Maryland's TPT, and Cambridge's 151 MHz array) operate over limited (5 km) baselines. There are two reasons why high resolution, low frequency interferometry has remained un-developed. First, the very long baselines required to obtain useful resolution imply a prohibitive cost in transmission of the data to a correlator. Second, and more importantly, the disruptive effects of the ionosphere make calibration of the data very difficult on baselines longer than 5 km. Recent hardware and software developments now enable serious consideration of a high resolution, low frequency instrument. The completed VLA waveguide system contains ample unused bandwidth for transportation of low-frequency astronomical signals over useful baselines, and new software techniques developed to improve the dynamic range of VLA data should enable calibration of low-frequency data taken from long baselines. These considerations lead to the development of a meter-wavelength synthesis instrument at the VLA.

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THE PROPOSED SYSTEM

In 1984 we proposed a low-frequency synthesis array that would utilize facilities already existing at the VLA site and would be operated in conjunction with the VLA [Perley and Erickson, 1984]. A useful array could be constructed for about one million dollars, and would represent an excellent investment with regard to the science that would be returned. The array design and construction present no technological problems. We believe that data calibration can be accomplished using recently developed algorithms combined with a priori information on the sources in the field of view obtained at higher frequencies.

In examining various design options, we adopted the following constraints:

(1) Operation of the low-frequency array should not displace or inconvenience the current VLA.

(2) Maximum use should be made of the currently existing hardware and software at the VLA site, without violating the first constraint.

(3) The total cost should be kept within reasonable expectations of what the NRAO RE (Research Equipment) budget can provide.

The proposed instrument would be a powerful tool for work on a broad range of astrophysical problems. The design beamwidth of 10" - 20" at 75 MHz will resolve thousands of objects whose structures have never been studied at frequencies below 100 MHz.

Since the ionosphere is a turbulent and highly refractive medium, it strongly affects the propagation of low-frequency radio waves. We have considered the expected effects on the data and have outlined a method of calibration. In addition, the problem of non-coplanar baselines was considered. These topics were discussed in some detail by Perley and Erickson [1984]. We showed that a general solution to the calibration problem should exist, based upon the fact that the system noise is entirely determined by galactic emission in the field of view and not by the receivers. We showed that for maps larger than 1° in extent, there should exist sufficient flux density from background sources to allow calibration of the data, assuming that an approximate initial model of the stronger sources is provided.

We emphasized that these conclusions were based on certain ideas concerning the typical behavior of the ionosphere. Tests are needed to confirm that the proposed method of calibration is actually practical. These tests can be made with elements of the 327 MHz system in the 'A' configuration or with the existing 25-meter VLA dishes instrumented for 75 MHz. The tests would be completed before any major expenditures are scheduled to occur. If the calibration is as simple as we anticipate (for a reasonable fraction of the total observing time), only modest computing facilities will be required. We expect that the calibration will be simple when the ionosphere is quiet, and difficult or impossible when the ionosphere is disturbed. What we have not been able to estimate accurately is what fraction of the time each of these conditions are to be expected.

The proposed design can be summarized as follows:

1. Continuum capability at or near 75 MHz. Moderately wide-band antennas are recommended for frequency flexibility, both in order to avoid terrestrial interference and to allow bandwidth synthesis. Dual polarization is strongly preferred.

2. A maximum bandwidth of 4 MHz (probably limited by interference), with two narrower bandwidths available. Attention should be paid to bandpass shaping to minimize confusion from sources outside the field-of-view.

3. An array consisting of at least 27 banks of antennas which will be permanently located near 'A' array stations. In addition, we propose to place banks with those VLBA stations within 400 km of the VLA. The proposed array will operate only when the 'A' array stations are occupied by existing 25-meter antennas. The effective collecting area of each bank should be between 100 and 200 square meters. The banks should be equatorially mounted, and fully steerable. It is highly desireable to build more than 27 banks, since this will considerably reduce the severe aliasing problems that are expected.

4. Pre-amplifiers will be located at each bank so the signals can be conducted to the nearest VLA antenna with no significant loss in signal-to-noise. Further amplification plus frequency and bandwidth selection will be done at the VLA antenna. The signals will then be injected into the existing electronics for transmission to the control building.

5. At the control building, the signals will be extracted and recorded with a VLBA (or similar) recording system. These signals will then be played back into a special, dedicated correlator. A modest, dual-channel, narrow-bandwidth correlator is proposed. Multiple fields of view can be mapped by repeated passes of the data through the correlator.

6. Calibration and mapping will be done using currently existing self-calibration algorithms.

7. Presuming 27 banks with 2 MHz bandwidth and 8 hour integration with dual polarization, the expected rms noise at 75 MHz will be about 3 mJy, two orders of magnitude lower than any other system at this frequency.

This would provide an array which would operate when the current VLA is in the 'A' configuration -- approximately 3 to 4 months per year. We estimate the cost of this system to be about one million dollars. A full-time array would cost approximately three times more.

AN ALTERNATIVE SYSTEM

After we made this proposal in 1984 it became obvious that, in spite of strong support from potential users, funding at the million dollar magnitude would not be available in the reasonably near future. We therefore considered a lower cost alternative that would take us at least part way towards our goal of the system outlined above. This alternative was to instrument the existing 25-meter VLA dishes at 75 MHz. A system was developed in which 75 MHz crossed dipoles (consisting of thin wires) are stretched between the feed support legs of the dishes. These dipoles, with the subreflector acting as a crude backplane, feed the dishes from their prime foci. With this crude feed system an aperture efficiency of about 20% and a collecting area of $\approx$100 m^2 per dish is achieved. This collecting area is marginal for calibration except when the ionosphere is quiet. However, the system should permit many of the investigations that were originally proposed. It does not satisfy constraint (1) as given above, i.e. use of this alternative 75 MHz system will require that the dishes be pointed at the object being observed and will thus strongly affect the simultaneous use of the VLA at other frequencies. Since many 75 MHz observations will be destroyed by ionospheric scintillations and will need to be repeated several times, this is not an efficient use of the VLA system.

The 75 MHz dipoles are in the shadow of the feed support legs and their presence does not affect the other VLA frequencies. Instrumentation at 75 MHz consists of only the dipoles and simple transistor preamplifiers; the preamplifier outputs are connected directly to an alternate input port on the 327 MHz modules. The principal problem with the system is caused by interference radiated by the digital equipment in the telescope. We have developed effective shielding for this interference.

Four VLA dishes are now instrumented at 75 MHz and operate satisfactorily. These have allowed us to demonstrate that strong, stable interference fringes can be obtained when the ionosphere is quiet. However, four elements do not yield enough baseline combinations for us to test mapping algorithms or to do useful science at this frequency. More dishes will be instrumented when resources permit. Once we have demonstrated that valid maps can be obtained and that useful scientific results can be produced with this alternative system, we hope to construct the full, stand-alone system that we originally proposed.

"A Proposal for a Large, Low Frequency Array Located at the VLA Site", R. A. Perley and W. C. Erickson, 1984, VLA Scientific Memorandum #146

PART III - SCIENCE WITH A LUNAR VLFA

SCIENCE AT VERY LOW RADIO FREQUENCIES

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Abstract: The broad scientific goals of a lunar based Very Low Frequency Array are presented. The frequency range of 1 - 30 MHz is defined to be the operative VLF window. The low frequency end of this window is useful for studies of the interstellar medium since the large scale distributions of thermal and relativistic gas are traceable at these frequencies. Studies of discrete objects, both galactic and extragalactic, are possible at the higher frequency end of the window. The VLF window is ideally suited for studies of phenomena not manifested in any other spectral band. These include studies of low energy cosmic ray particles, thermal environments of discrete radio sources and coherent radiation arising from collective plasma processes.

Introduction: Very low frequency radio astronomy is bounded by two major constraints. The first is the Earth's ionosphere. It has a characteristic and variable plasma frequency of $\approx$ 10 MHz. Combined with radio interference, both man-made and geomagnetic, routine observations are limited to > 30 MHz. Reliable, straightforward observations at lower frequencies can only be made outside the ionosphere and a handful of satellite-borne antennas have been used to do this (Kaiser, this workshop). A more fundamental limit to VLF observations is set by the interstellar medium (ISM). Although its plasma frequency is relatively low ($\approx$ 30 kHz), the ISM absorbs and suppresses radio emission, in a number of ways, at frequencies substantially greater than the plasma frequency. The Razin-Tsytovich effect, free-free absorption and synchrotron self-absorption are examples of mechanisms that inhibit radio emission near, and sometimes above 1

MHz. These mechanisms limit observations of discrete galactic and extragalactic objects but afford a better opportunity to probe the properties of the ISM. In fact, the VLF window defined by the 1 - 30 MHz range, spans a wide enough frequency range to allow both studies of the ISM and studies of discrete objects, relatively immune from foreground plasma effects.

Density fluctuations in the interstellar plasma lead to another effect, namely interstellar scintillation. Refraction by the inhomogeneities distorts the incoming wavefront, thereby blurring the observed image. This interstellar "seeing" sets a fundamental limit to the achievable resolution in direct VLF observations. At 1 MHz, this limit is ≈ 0.5 degrees, whereas at 30 MHz it falls to $\approx 2''$. As discussed by Dennison (this workshop) some resolution can be recovered, under certain conditions.

The inhomogeneous nature of the ISM all but eliminates the possibility of polarization studies at these frequencies. Within the effective beam of the VLF array, the ISM presents many such inhomogeneities with differing densities and magnetic field strengths. This leads to differential Faraday rotation which acts to depolarize the observed source. Polarization measurements will therefore only be possible for the closest sources. This does not present a great setback because polarization measurements can be obtained at higher frequencies with Earth-based telescopes.

The presence of a strong nonthermal background, which has a mean flux density of $\approx$ 1 Jy/(arcminute)^2 at 1 MHz (Cane, 1979), also limits VLF observations. The observed noise level depends on the system temperature

$$\Delta T_{rms} \propto T_{sys} = T_A + T_R,$$

where T_A and T_R are the antenna and receiver temperatures respectively. For normal cm-wave observations $T_A \ll T_R$ and the system temperature is dominated by the receiver noise. However, for strong sources at low frequencies the condition $T_A \gg T_R$ can be met. In those cases the signal to noise ratio becomes independent of the receiver noise and therefore independent of integration time since,

$$\Delta T_{rms} \propto T_A \approx 30S(Jy)/\Omega(Sr)$$

at 1 MHz. The measured flux density is S and the angular size of the beam is Ω. In the case of the galactic background $T_A \approx 2 \times 10^7 K$, much greater than the receiver noise temperature (Erickson, 1988; Douglas, this workshop). Furthermore, local enhancements in the background augment this effect. The extra sources of noise will limit the dynamic range of the observations and therefore limit the possible science.

Finally, the ISM is a source of coherent plasma processes which are manifested at very low frequencies. Such processes have been observed in solar radio bursts and in the magnetospheres of Jupiter, Saturn, Uranus and the Earth. Coherent effects are expected to be important near 1 MHz. The reduction and analysis of such observations is not straightforward as described later in the paper.

VLF radio astronomy is , today, in an analogous position to where X-ray astronomy was in the late 1960's. The Sun and a few strong galactic sources dominate the known sources in the VLF sky, as observed from the Earth. Unlike X-ray astronomy though, balloon-borne VLF astronomy is not possible since it is necessary to climb above the ionosphere. Only direct satellite observations can be made and these have been few with rudimentary antennas. An array could be constructed in space but this would not obviate the problems associated with the emissions from the Earth's magnetosphere (Kaiser, this chapter). VLF astronomy is therefore in the untenable position of being bounded by the presence of the Earth itself. It is these disadvantages which have slowed the development of VLF radio astronomy relative to X-ray and even γ-ray astronomy. An important lesson to be learned from the X-ray precursor missions is that not all the science can be predicted and quantified. The precursor missions themselves, provided much of the guidance in defining the scientific goals of more advanced missions such as the Einstein X-ray telescope. It is therefore desirable that the VLF lunar base mission be preceded by precursor projects aimed at mapping out the sky with an IRAS-style all sky survey. The scientific goals currently envisioned are summarized below, keeping in mind that better focus and even some redirection will be dictated by the proposed precursor missions.

Source Spectra. The synchrotron spectrum of a radio source is determined by the shape of the energy spectrum of the radiating electrons. The slope and the energy cut-offs of the energy spectrum have a one to one correspondence with the radio spectrum. The emissivity of the source depends on the strength of the ambient magnetic field and the density of relativistic electrons so that:

$$S_\nu \propto NB^{(1-\gamma)/2}\nu^{-(\gamma-1)/2} \tag{1}$$

between the 2 energy cutoffs, where S_ν is the synchrotron emissivity (incoherent radiation) at the observing frequency ν and γ is the index of the power law energy spectrum ($N(E) \propto E^{-\gamma}$).

Observations of synchrotron sources reveal surprisingly similar energy spectra having implied slopes close to the universal value of 2.5. High frequency cut-offs can generally be observed (although they don't always fall into the radio window). The low energy cut-offs, on the other hand, have rarely been observed because they fall below 30 MHz. The same holds true for direct detections of Earth-bound galactic cosmic rays for which the high energy end has been studied through air-shower experiments but the low energy end (below $\approx$ 0.1 - 1 GeV) is inaccessible because of solar modulation. Interestingly enough, the particle detector experiments and radio observations, both fail as probes of energetic particles in the same energy range. It is true to say that almost nothing is known about the spectra of relativistic particles at energies corresponding to the VLF window.

The calculation of radio source luminosities is directly dependent on the spectral cutoffs according to:

$$L \propto \nu_{max}^{1-\alpha} - \nu_{min}^{1-\alpha} \tag{2}$$

where L is the luminosity and the frequency cutoffs are labeled as max and min. The lower cutoff is important for steep spectrum source having α close to or greater than 1. This is of direct consequence to equipartition calculations of the energetics of radio sources since these are dependent on the source luminosities. The knowledge of the lower frequency cutoffs will improve calculations of the magnetic fields strengths and the energy contents of relativistic particles in steep spectrum sources such as pulsars and the lobes of radio galaxies. Better equipartition calculations, combined with observations of absorption effects and polarization studies at higher frequencies will shed more light on the energetics and environments of radio sources.

A relativistic electron radiates at a characteristic frequency given by

$$\nu_{syn} = 10^{18} B_{\perp} E^2 \, Hz. \tag{3}$$

An electron with an initial energy E_0, will lose half its energy in

$$t_{\frac{1}{2}} = 8 \times 10^9 \left(\frac{B}{\mu G}\right)^{-2} \left(\frac{E}{GeV}\right)^{-1} \quad yrs. \tag{4}$$

Energy losses of relativistic electrons quickly steepen an injected spectrum at the high frequency end. The longer the source ages without further particle injection, the lower the frequency, ν_k, at which the spectrum begins to steepen. Combining equations 3 and 4, it is possible to relate the turnover frequency ν_k to t, the age of the source. This frequency is given by:

$$\nu_k = 3.4 \times 10^8 B^{-3} t^{-2} \quad Hz \tag{5}$$

where t is in years. For $\nu_k = 1$ MHz and $B = 3 \times 10^{-6} G$, $t = 3 \times 10^{10}$ years. Thus, once accelerated, the lower energy electrons remain for a long time and may represent the original injection spectrum. VLF radio observations can therefore trace low energy electrons and, in a sense, probe the fossil records of radio sources.

The shape of the low frequency end of the radio spectra (free of absorption effects) will also provide a useful constraint on theories of particle acceleration which are relevant to the more general problem of the origin of cosmic rays. Perhaps the most widely proposed acceleration mechanism is the diffusive shock acceleration process which is a first-order Fermi process by which particles are accelerated across shock fronts. Direct observational evidence for such a mechanism has come from *in situ* observations in the interplanetary medium, with particles up to a few MeV in range, observed. In astrophysical settings, such as SNRs, radio observations indicate the presence of particles having energies of a few GeV to hundreds of GeV. The overlap between the lower energy particles produced in the Solar System and the higher energy particles at astrophysical sites has not been observed because of the dearth of observations at the appropriate energies. Such an overlap is needed in order to test the applicability of the observationally verified acceleration processes in the solar system to the more extreme astrophysical environments. Most theories of particle acceleration predict approximate energy cutoffs for the energized particles (e.g. Volk, 1988; Blandford, 1988). Observations of the low energy end of the particle distribution function will strongly constrain these theories.

VLF Radio Spectra.

The spectra of radio sources below 30 MHz, have never been reliably measured with Earth-based observations. Furthermore, poor angular resolution has prevented detailed mapping of sources even at frequencies as high as 300 MHz. The mechanisms that modify the spectra at low frequencies through extrinsic (environmental) and intrinsic effects are described below.

The Razin-Tsytovich Effect

According to Tsytovich (1951) and Eidman (1958) and Razin (1960), relativistic electrons embedded in a thermal plasma have their synchrotron radiation suppressed below the critical frequency given by:

$$\nu_R \approx 20 \frac{N_e}{B_\perp} Hz. \tag{6}$$

The spectrum steepens in shape from

$$S_\nu \propto \nu^{-(\gamma-1)/2} \qquad \nu > \nu_R$$

to (Lang, 1980)

$$S_\nu \propto \nu^{\frac{3}{2}-\gamma} e^{-3.7\nu_R/\nu} \qquad \nu < \nu_R.$$

For typical interstellar gas densities of 0.1 cm $^{-3}$ and $B \approx 10^{-6}\,G$, $\nu_R \approx 10^5$ Hz, well below 1 MHz. However, if the environments in which the particles are radiating have locally higher electron densities and/or lower B fields this effect could easily manifest itself in the VLF window defined above. The effect could be searched for in galactic objects such as SNRs and pulsars and in extragalactic objects such as active galactic nuclei (AGNs) and lobes of radio galaxies. Recognition of the unique spectral shape predicted by the Razin-Tsytovich effect is the primary means by which this mechanisms would be identified in radio sources. The environment of such a source can be probed since the ν_R depends only on N_e and $B_\perp$. An independent means of calculating one of the 2 parameters would immediately yield the second. For example, equipartition calculations of $B_\perp$ and the observed value of ν_R yield the thermal gas density according to equation 6. Comparison of such calculations with Faraday rotation measurements made at higher frequencies would provide a consistency check and a direct test of the validity of equipartition calculations. Such a test would be of profound significance in understanding the physics of the interaction between magnetic fields and relativistic particles in radio sources.

Free-free (Thermal) Absorption

Plasma between the observer and the source can absorb radiation through the free-free transitions of ions. The optical depth of such a plasma to radio frequency radiation is given by

$$\tau_\nu \propto T^{-1.35} \nu^{-2.1} \int N_e^2 dl \qquad (7)$$

which is unity at a frequency given by:

$$\nu_T \approx 0.3\, T_e^{0.68}\, N_e\, l^{1/2} \quad GHz \tag{8}$$

where l is the path length in parsecs. Taking $T_e = 10^4 K$ and $N_e = 0.1\ cm^{-3}$ the turnover frequency is a function only of l. Looking through the galactic plane, the turnover frequency is $\approx$ 1 MHz. Perpendicular to the plane, $\nu_T \approx 0.2$ MHz. Extragalactic sources and distant galactic sources will therefore be strongly attenuated looking through the galactic plane and marginally attenuated otherwise. A statistical study of extragalactic sources could therefore be used to map out the distribution of thermal, ionized gas in the galaxy. Such a study can be combined with statistical studies of the Faraday rotation (e.g. Simard-Normandin, Kronberg and Button, 1980) to determine length averaged B fields through the galaxy. This would impact significantly on our very limited knowledge of the structure and strength of the global interstellar magnetic fields. Similar studies of discrete sources can be used to probe their local environments. A good example is the thermal environment thought to exist around extragalactic jets and lobes. If the thermal gas is mixed in with the radiating particles, the spectrum assumes a flatter shape given by:

$$S_\nu \propto \nu^{2.1-(\gamma-1)/2} \quad \nu < \nu_T.$$

If the absorbing gas is between the source and the observer then:

$$S_\nu \propto e^{(\nu/\nu_T)^{-2.1}}$$

In either case, the absorbed spectrum has a recognizable shape.

Synchrotron Self Absorption

A source is opaque to its own synchrotron radiation when:

$$\nu_s \approx 34\left(\frac{S_{\nu_s}}{\theta^2}\right) B^{1/5} \quad MHz \tag{9}$$

where B is in Gauss and θ is the angular size of the source in seconds of arc and S is in Janskys. Normally a radio source is self-absorbed under the most extreme conditions such as the cores of radio galaxies. However, for very low frequency these conditions can be less extreme. Consequently, the frequency turnovers of many more sources will be uncovered. A self absorbed spectrum has a theoretical shape, below ν_s given by ($S_\nu \propto \nu^{2.5}$). In practice, the observed slope does not reach such high values because different parts of the source reach self-absorption at different frequencies. This dilution is minimized at very low frequencies because the absorption can occur over greater angular scales. Nevertheless, this important lesson from high frequency radio observations needs to be considered when interpreting the observations. The magnetic field of a self absorbed source can be derived very accurately because of its strong dependence on ν_s. In order to disentangle synchrotron self-absorption from other absorption effects, these searches should best be carried out at the higher frequency end of the VLF window ($\approx$ 10 MHz). Once the self absorbed sources are identified, their angular sizes can be determined form GHz observations, leaving the magnetic field as the only remaining variable. The determination of B for a large sample of radio sources is important in understanding their internal energetics and the lifetimes and replenishment of relativistic particles.

Coherent Plasma Processes: When the particles that make up a plasma are separated by scale lengths comparable to the wavelength of the EM radiation, collective effects can and do become important. If a population inversion of emitting particles exists ($\frac{\partial N}{\partial E} > 0$) this leads to stimulated emission which is a coherent process. This presents both an opportunity and a problem. The opportunity lies in the possibility of studying a new and interesting phenomenon as has been done in the Solar system with solar radio bursts, and the magnetospheres of Jupiter, Saturn, Uranus and the Earth (Kaiser; Desch, this chapter). The parameters under which this process operates suggests that it may be observable in other stellar systems and in many radio sources where modestly high gas densities ($10^3 - 10^5\, cm^{-3}$) and B fields ($0.01 - 1G$) exist (Stone and Erickson, 1976). The problem, lies not in the science but in the manner in which the data is analysed.

Most interferometric techniques assume that source radiation is spatially uncorrelated. Coherent radiation processes may, however, correlate over large spatial scales and the assumption is therefore invalid.

The VLF array is sensitive to:

$$V_\nu(\mathbf{r_1},\mathbf{r_2}) = \langle E_\nu(\mathbf{r_1}) E^*_\nu(\mathbf{r_2})\rangle$$

where V_ν is the measured correlation and E is the electric field at the 2 source locations, $\mathbf{r_1},\mathbf{r_2}$. The normal simplifying assumption for incoherent radiation is that, at the source positions $(\mathbf{R_1},\mathbf{R_2})$, $\langle E_\nu(\mathbf{R_1}) E^*_\nu(\mathbf{R_2})\rangle = 0$ for $\mathbf{R_1} \neq \mathbf{R_2}$. In the case of coherent radiation that assumption cannot be made. Spatially correlated radio emission can be mapped in the usual way but the interference pattern is a function of not only the relative antenna positions but also the *absolute* antenna positions. The data reduction techniques are therefore not as straightforward (Anantharamaiah, 1988).

In the case of interstellar scintillation the emission is coherent over short time scales (fractions of seconds). Extremely short *snapshots* can be used to record the instantaneous coherence pattern which can then be analysed using techniques analogous to speckle imaging (Dennison, this workshop).

Solar and planetary observations will make up a major fraction of the research effort of the array, particularly if frequencies below 1 MHz are used. Absorption effects are not important at the relatively short distances involved and observations in the kHz bands are possible. Such observations will allow better studies of the coherent processes that take place in the solar system. With the right combination of frequencies and angular resolution, it may be possible to spatially resolve the regions of coherent emission in the magnetospheres of the giant planets. The data reduction, in such cases, has the same complications discussed above.

Interstellar Scintillation. An EM wave travelling through an inhomogeneous plasma will have its phase altered in response to changes in the refraction index along its path. This leads to a bending of the propagating ray in a manner analogous to a random walk. The average angular deviation from a straight line-of-sight path determines the apparent angular size of the source. As discussed by Dennison (this chapter) this leads to an effective lower limit on the angular size of any radio source, which at 1 MHz is ≈ 0.5 degrees. Since the effect is $\propto \nu^{-2}$ the limit approaches $1''$ at 30 MHz. Although this limits the resolution of the VLF array it also acts as a well defined constraint on its design. Under some conditions extra resolution may be achievable through deconvolution techniques and this may need to be taken into account when designing the array.

By inverting the problem, interstellar scintillation can be used to probe the nature of the scatterers in the ISM. Comparing VLF and higher frequency measurements of the angular sizes of a large number of extragalactic sources, statistical studies can be made to map out scintillation effects as a function of galactic longitude and latitude to gain insight into the global distribution of the scatterers in the galaxy. The same technique can be used to study the properties of the IPM.

Summary. The major research effort of the lunar based VLF array can be divided into 4 broad categories:

(a) Solar, planetary studies (≤ 1 MHz)

The low frequency end of the VLF window is ideally suited for solar and planetary work because of its sensitivity to coherent effects. Absorption effects are avoided since the sources are relatively nearby. The bulk of this research will be aimed at monitoring transient phenomena and resolution is therefore not an important issue. Nevertheless, the design of the array will allow for good directionality to avoid sidelobe interference, particularly from the sun and the Earth's magnetotail.

(b) Studies of the ISM (absorption effects, scintillation; ≈ 1 MHz)

Observations at frequencies near 1 MHz will be used to probe the ISM and the environments of radio sources. The Razin-Tsytovich effect, combined with high frequency Faraday rotation measurements can be used to estimate N_e and B in the local environments of radio sources. Thermal absorption of extragalactic sources can be used to measure the distribution of HII in the galaxy. These measurements can then be combined with pulsar dispersion measurements, and observations of foreground Faraday rotation to more accurately determine the column densities of HII, as a function of both latitude and longitude. Measurements of radio disks caused by scintillation can be used to probe the second moment of the density distribution ($< \Delta N_e^2 >$).

(c) Discrete radio sources (galactic and extragalactic sources; 10 - 30 MHz)

Observations of synchrotron self-absorption allows the determination of intrinsic source magnetic fields. Consequently, energetics of radio sources can be better understood and the validity of equipartition assumptions can be directly tested.

The medium to high frequency portions of the VLF window provide observations of synchrotron spectra relatively free of absorption effects. Locating intrinsic low energy cutoffs will allow more accurate determinations of radio source luminosities which in turn improve calculations of source energetics. Studies of the long lived electrons that radiate at these frequencies will provide useful insights into source histories. The shape of the spectrum at low frequencies, and the locations of the low energy cutoffs are important constraints of particle acceleration theories since these 2 parameters are model dependent.

(d) High resolution, all-sky survey (1 - 30 MHz)

History is the best judge of the importance of all-sky surveys. The precursor mission will provide initial low resolution maps of the sky. The lunar base array will produce high resolution, sensitive maps of the sky and generate a data base for detailed statistical studies and follow-up studies of individual sources. Most importantly, the survey will set the foundation for long term planning and optimal use of the array.

The VLF window is not just another portion of the radio spectrum. It is an unexplored spectral region which offers insights into phenomena impossible to study from the Earth or even in Earth orbit. It promises insights into the ISM and gaseous environments of radio sources. A window will literally be opened onto the uncharted fields of plasma effects and coherent processes. Serendipity, is a factor whenever a new field is opened to exploration and we expect discoveries to shape much of the future research the VLF array will generate.

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Voyager Radio Astronomy and
The Low Frequency Near-Earth Radio Environment

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Abstract. I review the major features of the radio astronomy experiment onboard the two Voyager spacecraft. In addition, the major sources of noise in the near-earth environment are identified. They are: (a) terrestrial atmospherics, (b) terrestrial kilometric radiation, and (c) solar Type III bursts. Interference from all but the last is eliminated by observing from the lunar far-side. Based on experience with numerous classes of spaced-based radio astronomy experiments, the use of relatively simple antennas with a sophisticated observing scheme seems not only expedient but most likely to operate effectively on the moon. Finally, a science rationale for lunar-based planetary observations is presented.

1. Planetary Radio Astronomy on Voyager

The Planetary Radio Astronomy (PRA) receiver shares space on Voyager with 9 other instruments (Figure 1). The 198-channel swept-frequency receiver is driven by two 10-meter orthogonal monopoles. The input is alternately switched through a quarter-wave hybrid to sense right-hand and left-hand circular polarization. The PRA receiver covers the frequency range from 1 kHz to 40 MHz in 198 steps. Figure 2 shows how this band covers the well-known earth-based transmitter frequencies and the natural planetary radio sources now recognized in the solar system.

Both Voyagers were launched in 1977. As illustrated in Figure 3, encounters with Jupiter occurred in 1979, with Saturn in 1980 and 1981, and with Uranus (Voyager 2) in 1986. Voyager 2 will encounter Neptune in August 1989. Thus far, every encounter has resulted in the discovery of new radio phenomena.

Figure 4 shows a 24-hour radio spectrogram. The frequency band ranges from 1 kHz at the top of each panel to 1320 kHz at the bottom of each panel in 70 equal steps. Both solar type III bursts and Saturn kilometric radiation (SKR) are clearly visible. Some jovian emission (HOM) is visible near 1 MHz, especially from about 7 to 8 hours in panel 2. At the time this spectrogram was made, Voyager 1 was about 0.7 AU from Saturn and about 4 AU from Jupiter.

The spectrogram in Figure 5 illustrates a type of emission, namely SED, never before seen by Voyager. The SED, or Saturn Electrostatic Discharges, are visible as the short vertical streaks, extending from 40 MHz to 100 kHz.

C-17

Because of the way our receiver samples, we know that the SED are actually very broadband (probably at least many hundreds of MHz) and of short duration (50 - 100 msec). They have been shown to originate in Saturn's atmosphere, near its equatorial region, and are very likely due to lightning-like emissions. By observing the low-frequency cutoff of these atmospheric emissions, we were able to generate a global model of Saturn's equatorial ionosphere density.

2. RF Environment Near 1 AU

Observations from Voyager and from other spacecraft carrying radio instruments have helped us assess the radio noise environment in the vicinity of 1 AU. Figure 6 shows a radio spectrogram of Voyager data when it was near earth, shortly after launch. The spectrogram is dominated by the earth's natural auroral radio noise (AKR). AKR is observed in the frequency band from about 50 kHz to 750 kHz. Also seen in this spectrogram is a type of noise generated at the antennas, due to coupling with the antennas to the solar wind. It is always observed when the solar wind plasma density is unusually large, say, greater than a few tens of particles per cubic centimeter. In the spectrogram, this antenna coupling noise (or thermal noise) occurs over the band from a few kHz to about 200 kHz. A solar type III burst can be seen in the first few hours of the spectrum. The top panel in this spectrogram illustrates the polarization signature of these emissions. Black represents LH circular polarization. The AKR appears LH polarized, while the Type III burst and the antenna coupling noise are unpolarized because they do not show a consistent polarization signature.

Figure 6 does not give a good indication of the potential level of noise from solar Type III bursts. Figure 7, however, shows how the Type III bursts can completely dominate the noise background at times. The top panel cartoon is useful in helping to identify the Type III bursts and the AKR.

At somewhat higher frequencies, typically above 1 MHz, man-made noise and lightning rf contribute significantly to the general level of noise observable at 1 AU. Figure 8, from RAE (Radio Astronomy Explorer) satellite observations near 9 MHz, shows antenna temperature contour levels in dB above 300 ^{o}K. The observations were made from an altitude of about 6000 km. Given the proper ionospheric conditions, almost any location on earth can produce easily detectable noise in the terrestrial vicinity.

Figure 9 compares quantitatively some of the noise sources discussed above. The observations were made from the ISEE spacecraft, from out near the earth-sun libration point, several hundred R_E from earth. Lacking is the contribution due to man-made noise, which would dominate at frequencies above 1000 kHz. Man-made noise would certainly be comparable to the IP storm level above 1 MHz, however. Two spectra are shown for the AKR; the lower one (dashed) illustrates the intensity level of AKR when observed over the dayside of earth, the higher one (solid) shows the level observed over the nightside. IP storm is the radio noise due to a succession of Type III solar bursts observed during an interplanetary (IP) "storm".

Table 1 provides a summary of the recognized low-frequency noise sources near 1 AU. They are tabulated in approximate order of importance. Items with an "X" in column 1 would be detectable on the lunar far side. Others should be undetectable due either to the shielding provided by the moon, or because, as with Uranus, the emission is too weak.

Table 1

LOW-FREQUENCY NOISE SOURCES at 1AU

	Source	Frequency	Pol	Level	Spacecraft
	Terrestrial Atmospherics	1 MHz - Day	yes	v. strong	
x	Type III Bursts	20 kHz - Day	no	strong	
	AKR	20 - 750 kHz	yes	strong	
x	AKR'	20 - 100 kHz	?	moderate	ISEE
x	Antenna Coupling	dc - 200 kHz	no	strong	
x	Jupiter	20 kHz - 40 MHz	yes	moderate	V1, V2, ISEE, RAE
x	Saturn	20 kHz - 1 MHz	yes	moderate	ISEE
	Uranus	20 kHz - 850 kHz	yes	v. weak	IMP-6 (?)
	Neptune	?			

Notes to Table 1

- Note that two of the major sources, namely, terrestrial atmospherics and AKR do not present a noise problem on the far side.
- Type III bursts would be detectable over approximately 1/2 of each lunation.
- AKR' (AKR prime) is a recently discovered low-frequency component of the AKR that is of moderate intensity, but very infrequently observed. It may originate on field lines far down the earth's tail and would therefore be observable over half a lunation. This particular component may represent an important science objective of the array.
- Antenna coupling noise would be detectable during times when the array is immersed in the solar wind (approximately half a lunation) provided the solar wind density is extremely high (a few hours each month).
- Finally, the planetary sources Jupiter and Saturn should be relatively easy to detect and would represent important science objectives of the array.

Table 2 provides a rough comparison of spacecraft antenna systems with which the author has had personal experience. A scorecard is included to evaluate the relative successes of each. It is clear, of course, that for planetary observations, 'going there' is the best bet. Hence Voyager, with only a 10-meter dipole, has detected the most planets. If confined to the near-earth vicinity, then residing in an rf quite neighborhood is the next-best thing. Hence ISEE, fixed at the earth-sun libration point on the earth's dayside has done extremely well. If ISEE had to make observations above 9 MHz (the dayside critical frequency), however, it would not do very well owing to likely interference from terrestrial atmospherics. RAE, surprisingly, has not done that well considering the size of its antenna system. This is particularly true below 1 MHz. I believe this fact underscores the importance of the observing mode in helping to properly identify the myriad of signals incident on space-based receiver systems. ISEE, with a small antenna system, maintains a very good identification record due to the spinning platform from which the observations were made. Harmonic phase fits to the intensity modulations from various sources helped considerably in identifying signals.

Table 2

Spacecraft Antenna Systems: Comparison

Spacecraft	Antenna			Scorecard
	physical length	electrical type	observation mode	
RAE	460 m	travelling wave V	transit	J (not easily)
IMP-6	90 m	electrically short	spinning	J (easily), U(?)
ISEE	90 m	electrically short	spinning	J, S (both easily)
Voyager	10 m	electrically short	encounter	J, S, U

J = Jupiter
S = Saturn
U = Uranus

Summarizing, in low frequency radio astronomy a quiet environment and a clever observing scheme are far more important than massive antenna structures. This general 'philosophy of low frequency radio astronomy' argues strongly for individual array elements that are very simple in design, such as simple short dipoles, rather than individual elements that are complex, such as log spirals or large Vee structures. This simplicity is more than made up for by the use of aperture synthesis, which is in a sense the modern-day equivalent of the spinning platform used on ISEE to identify the direction of arrival of incoming signals.

3. Science Rationale from a Planetary Perspective

As is apparent from Figure 10, most of the interesting work that can be done in planetary radio astronomy is below 1 MHz. The spectral peaks of the emissions discovered by the PRA experiment onboard Voyager are all in the neighborhood of 100 - 1000 kHz. These are radio components that cannot be observed from the ground, and can only be detected from space in quiet rf environments.

Planet by planet, the emission components of immediate interest and whose study would most likely have far-reaching implications for solar-system research are as follows:

earth Although the primary component of the terrestrial emission, AKR, is invisible from the lunar far side, of potentially great interest is the recently discovered (what I have called here) AKR' emission. It occurs only infrequently, but may be very important in mapping the plasma dynamics of the earth's magnetic tail regions during disturbed, auroral-related, conditions. Very little is known about this emission at present, but if its importance to tail dynamics holds up, then a lunar monitoring platform would be ideal for its study.

Jupiter The kilometer-wavelength components of Jupiter's emission (labelled KOM in Figure 10) are important in understanding the dynamics of the Io plasma torus. Presently, and for the foreseeable future, Io and its related plasma torus, play an extremely important role in understanding the physics of the jovian system. Study of the KOM provides one of the few ways, along with spectroscopy, in which the Io system can be monitored remotely.

Saturn The Saturn kilometric radiation (SKR) is fairly well understood, although further monitoring would almost surely turn up additional surprises. An important advantage in monitoring the SKR, however, is the fact that the SKR is strongly solar wind controlled. Therefore, a record of the intensity level of the SKR is also a highly reliable record of solar wind conditions, in particular the solar wind density, at a distance of 10 AU. This record may be of importance in deconvolving the low-frequency synthesis maps from the effects of interplanetary scintillations.

In addition to the SKR, the Saturn Electrostatic Discharges, or SED, have provided extremely important information on Saturn's ionosphere, besides being an interesting phenomenon to study in their own right. Observations of SED near 5 MHz yield information on variations in the ionospheric density; there is no other remote sensing technique capable of doing this.

Figure Captions

Figure 1 Line drawing of the Voyager spacecraft showing its complement of instruments.

Figure 2 Frequency coverage of the radio astronomy instrument on Voyager. The radio emissions from Saturn, Uranus and the low frequency portion of the Jovian emission were unknown before Voyager.

Figure 3 Trajectory of the two Voyagers. Voyager 2 encounters Neptune in August, 1989.

Figure 4 Radio spectrogram showing 24 hours of data from Voyager 1. In each frame, 1 kHz is at the top and 1320 kHz is at the bottom. Several solar type III bursts and several episodes of Saturn kilometric radiation (SKR) are visible.

Figure 5 (top) Radio spectrogram taken near Saturn showing 20 minutes of activity during an intense period of SED (Saturn electrostatic discharge) activity. (bottom) Periodic nature of the SED 'storms' over a 7-day span from which it was deduced that the SED were coming from an equatorial storm system in the atmosphere of the planet.

Figure 6 Typical radio spectrogram of radio frequency 'noise' environment near 1 AU.

Figure 7 Radio spectrogram from ISEE-1 showing the intensity of solar type III bursts [from Farrell and Gurnett].

Figure 8 Isocontours of antenna temperature from RAE-1 observations at an altitude of 6000 km. Noise is due to the combination of lightning activity and man-made transmissions at 9 MHz.

Figure 9 Flux spectrum showing typical noise levels of the most important noise sources at 1 AU. Included are AKR (lower limit), solar type III emission (IP), and antenna thermal noise due to coupling with the solar wind.

Figure 10 Median flux densities of the known planetary sources normalized to a standard distance of 1 AU. The dashed portion of the Jupiter curve is what was known before Voyager.

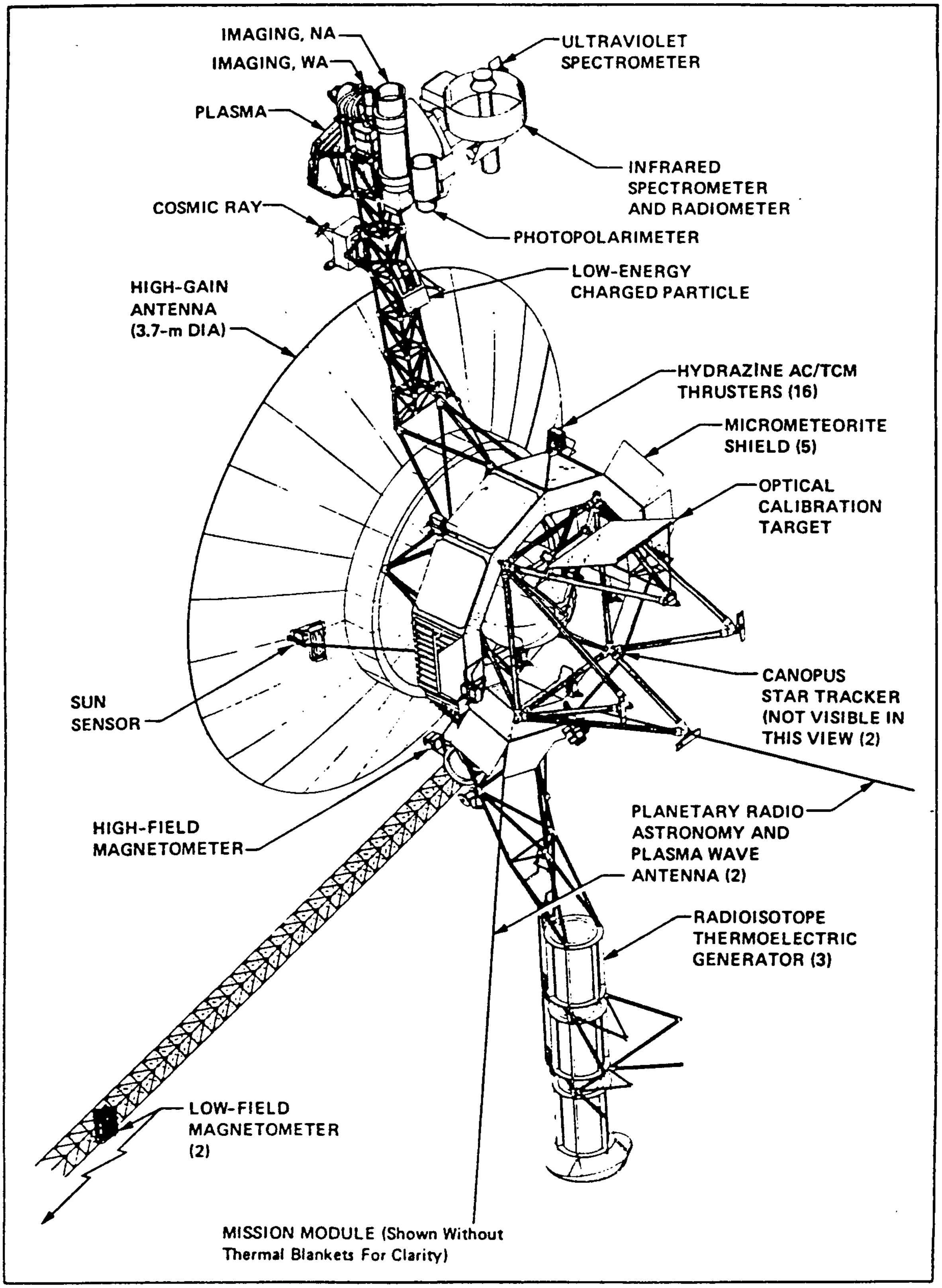

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Fig 1

VOYAGER COVERAGE OF PLANETARY RADIO EMISSIONS

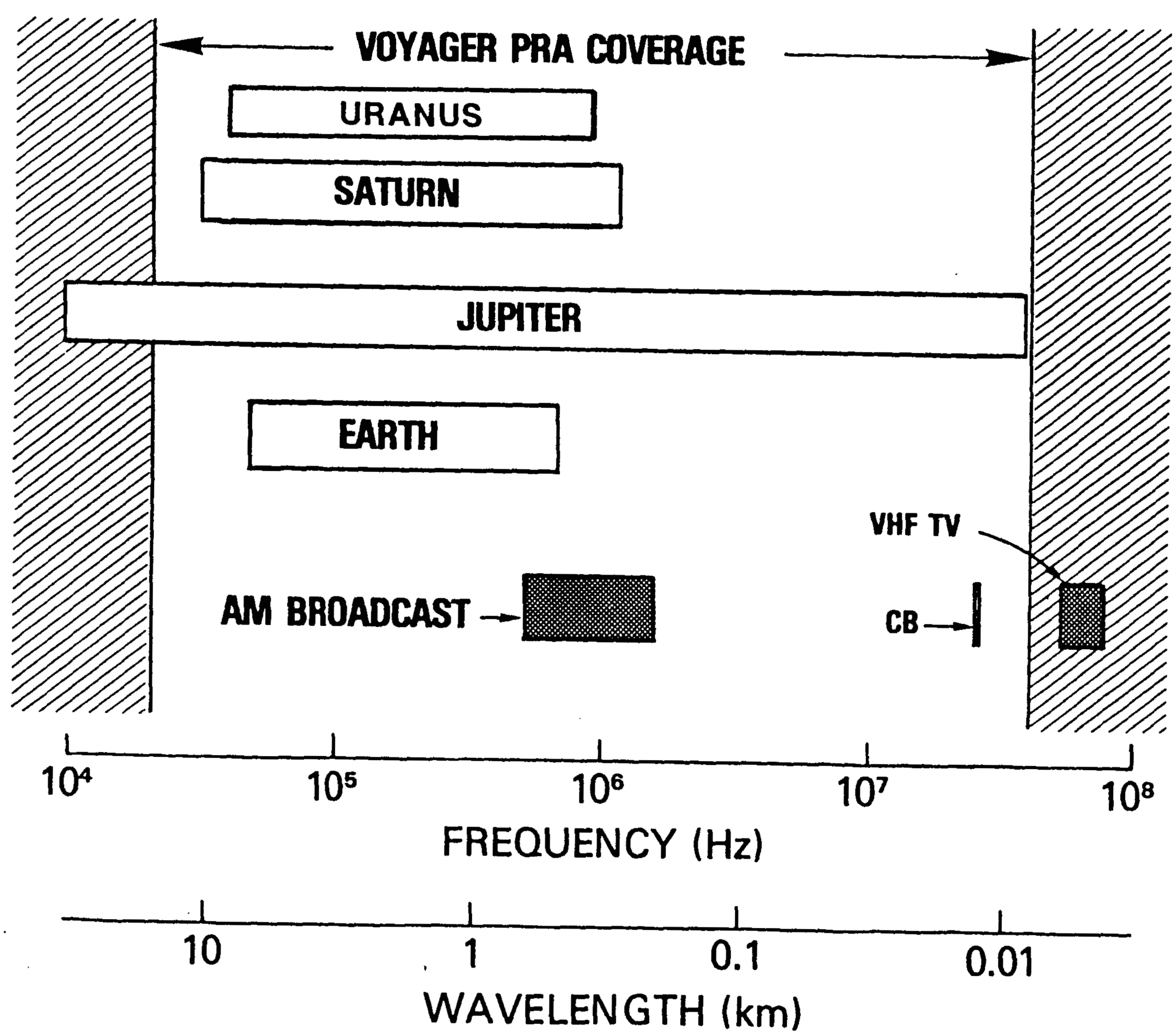


Fig. 2

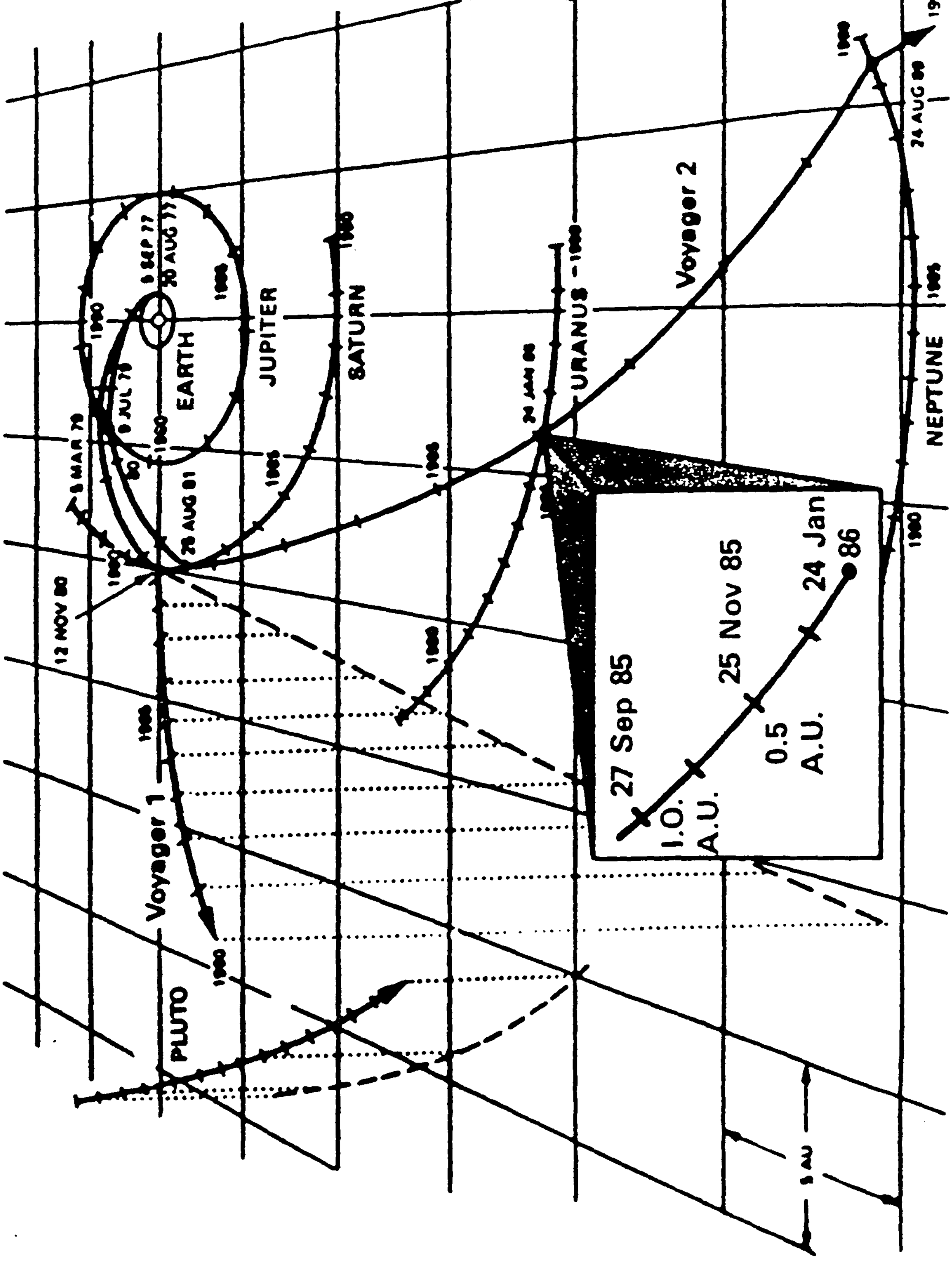
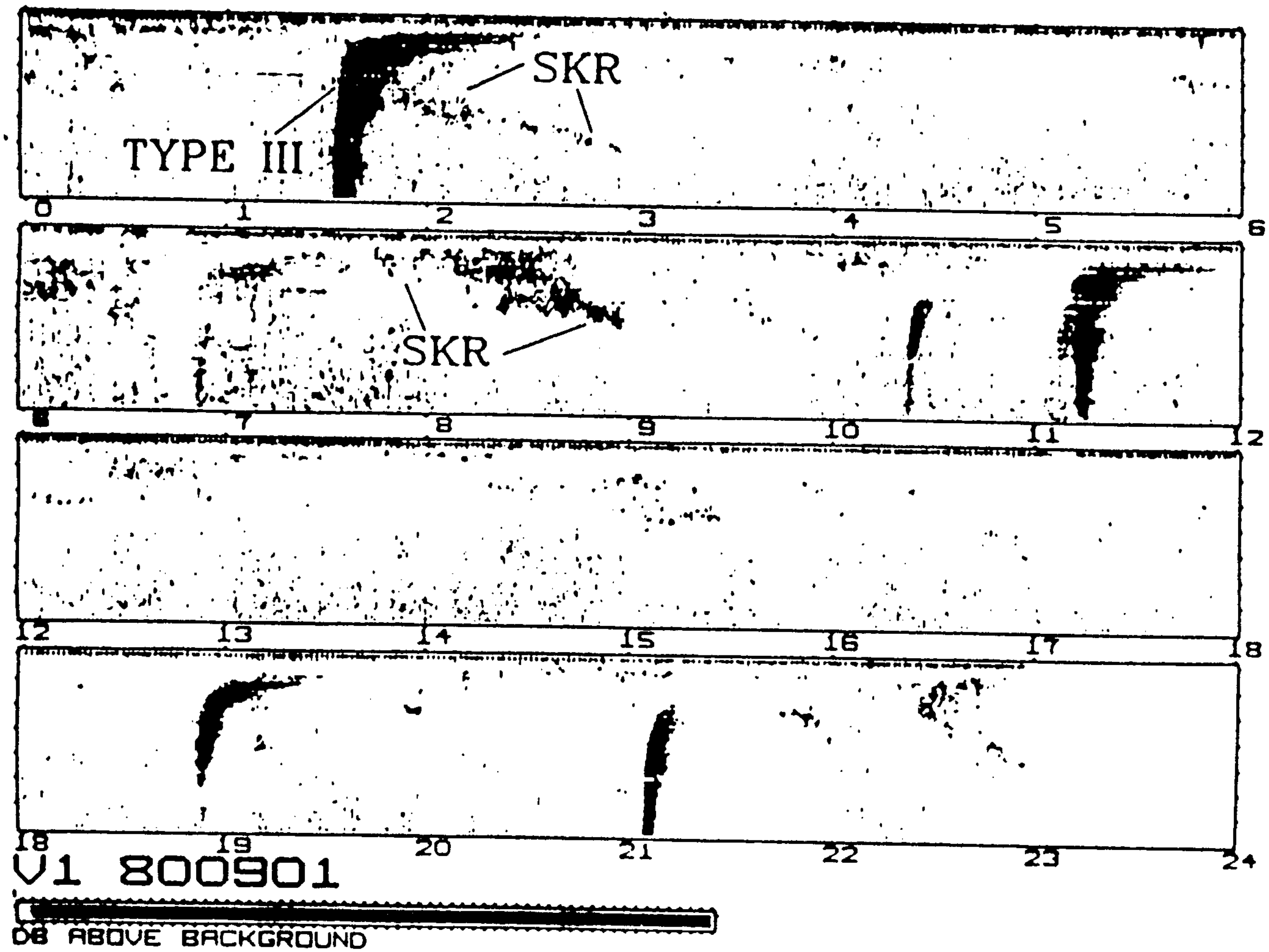

Voyager 1
Voyager 2
EARTH
JUPITER
SATURN
URANUS
NEPTUNE
PLUTO
12 NOV 80
24 JAN 86
24 AUG 89
1990
27 Sep 85
25 Nov 85
24 Jan 86
0.5 A.U.
1.0 A.U.

Fig. 3



TYPE III
SKR
SKR
0
1
2
3
4
5
6
7
8
9
10
11
12
13
14
15
16
17
18
19
20
21
22
23
24
V1 800901
DB ABOVE BACKGROUND

Fig. 4

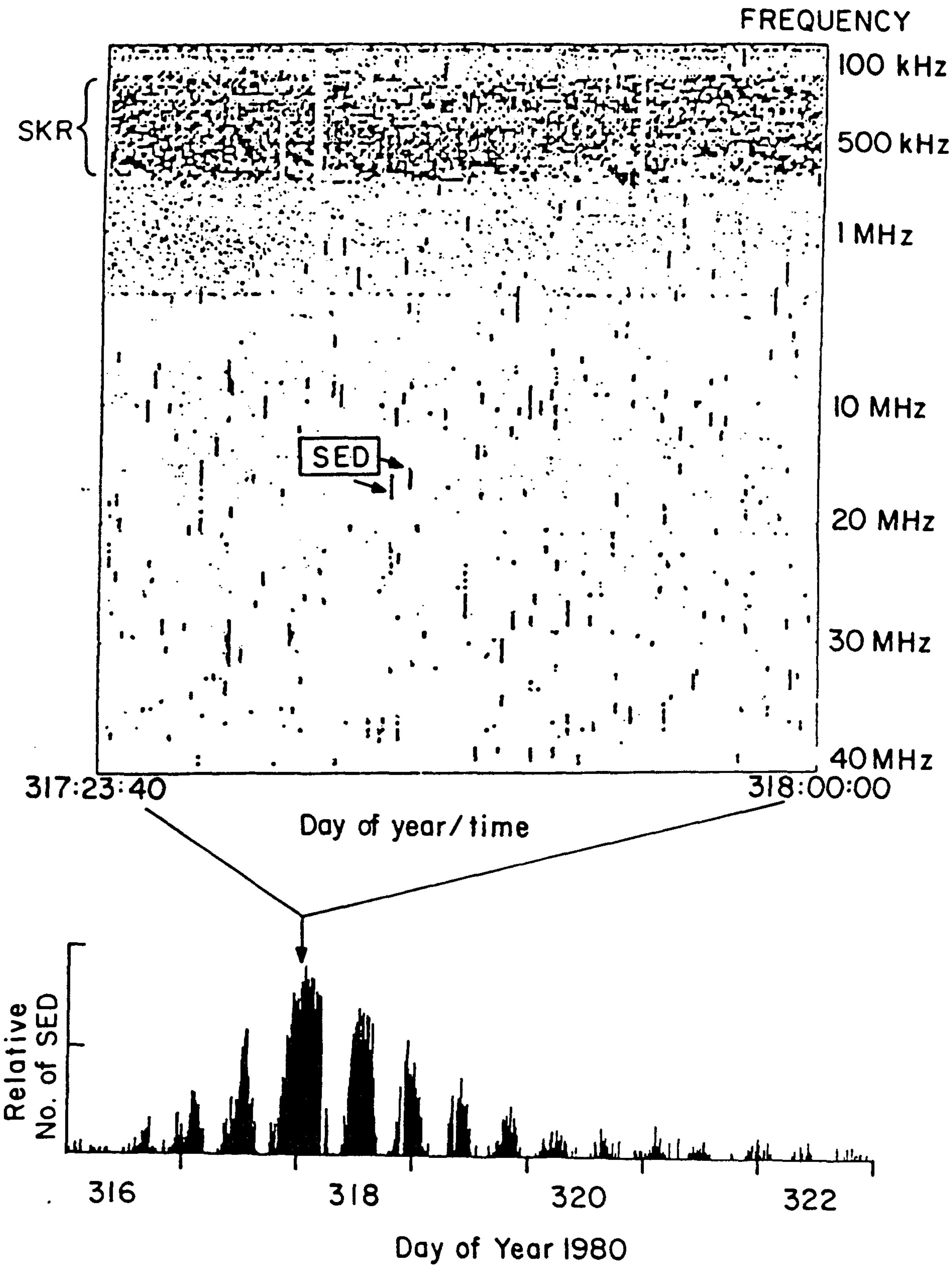

FREQUENCY
100 kHz
500 kHz
1 MHz
10 MHz
20 MHz
30 MHz
40 MHz
SKR
SED
317:23:40
318:00:00
Day of year/time
Relative No. of SED
316
318
320
322
Day of Year 1980

Fig. 5

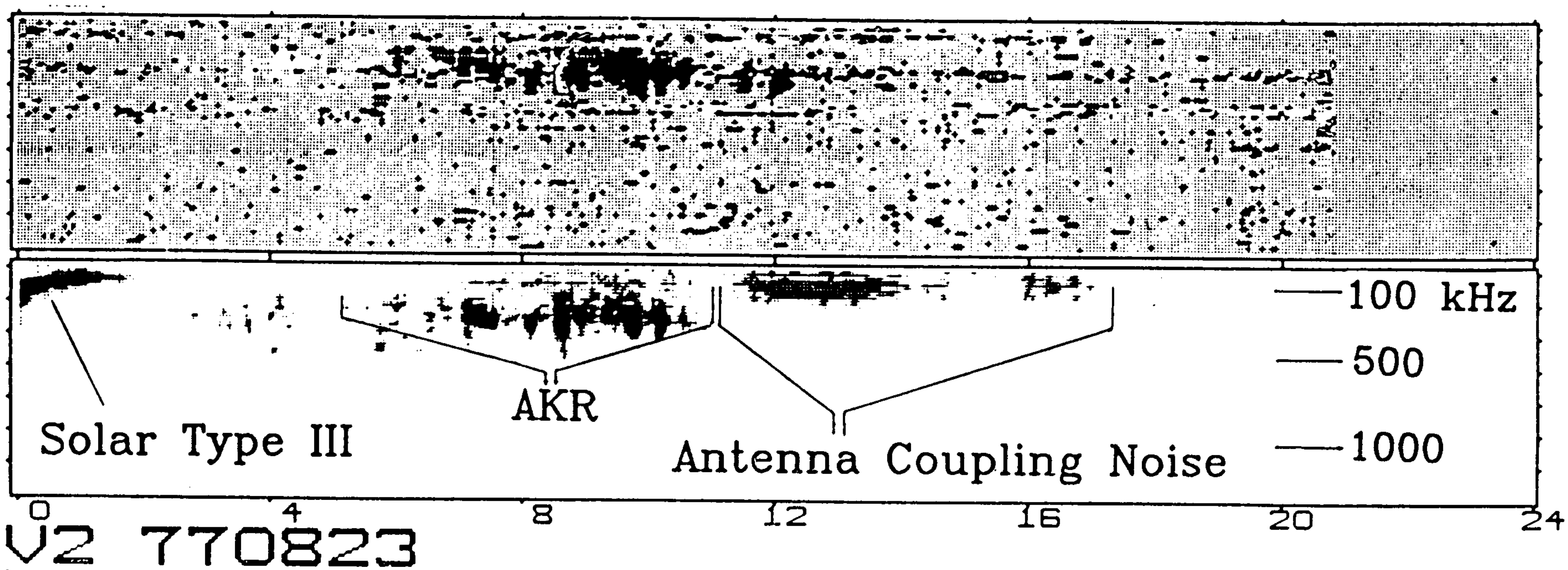

100 kHz
500
1000
Solar Type III
AKR
Antenna Coupling Noise
0
4
8
12
16
20
24
V2 770823

Fig. 6

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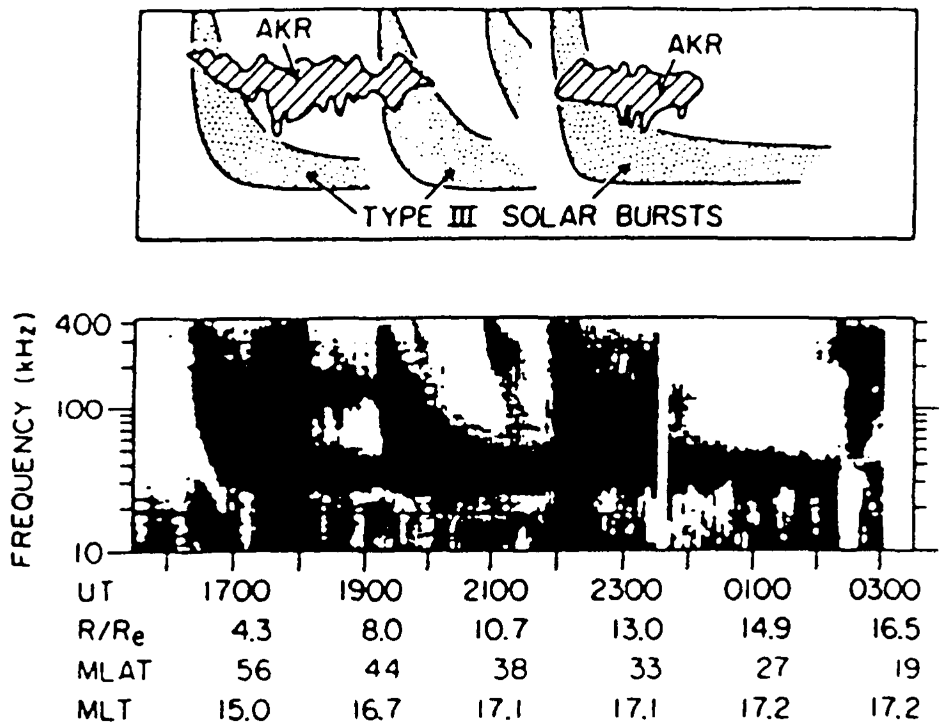


ISEE-1 3-4 JULY 1979

FIG 7

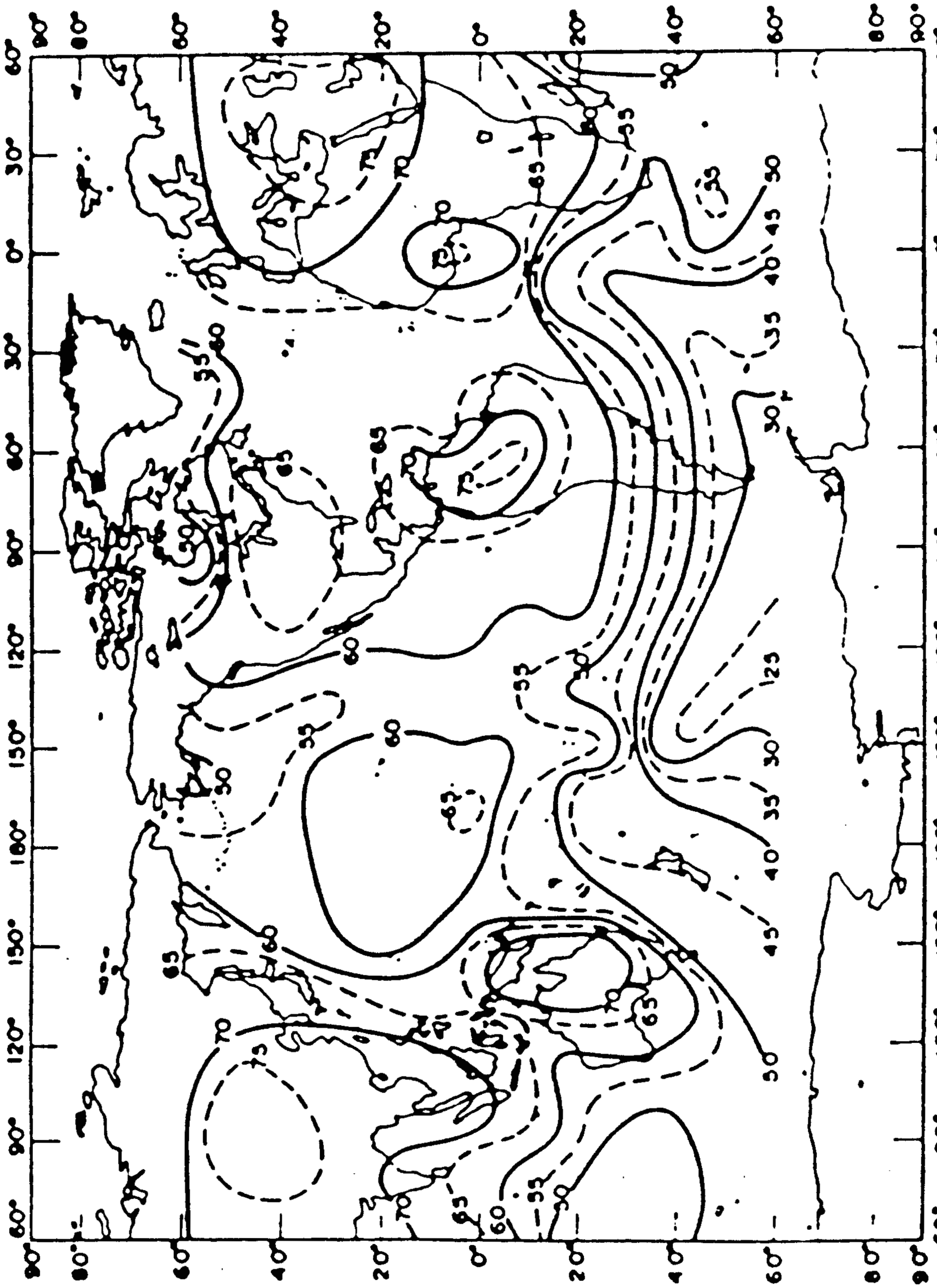


Fig. 8

SOLAR STORM
SEPT, 1983

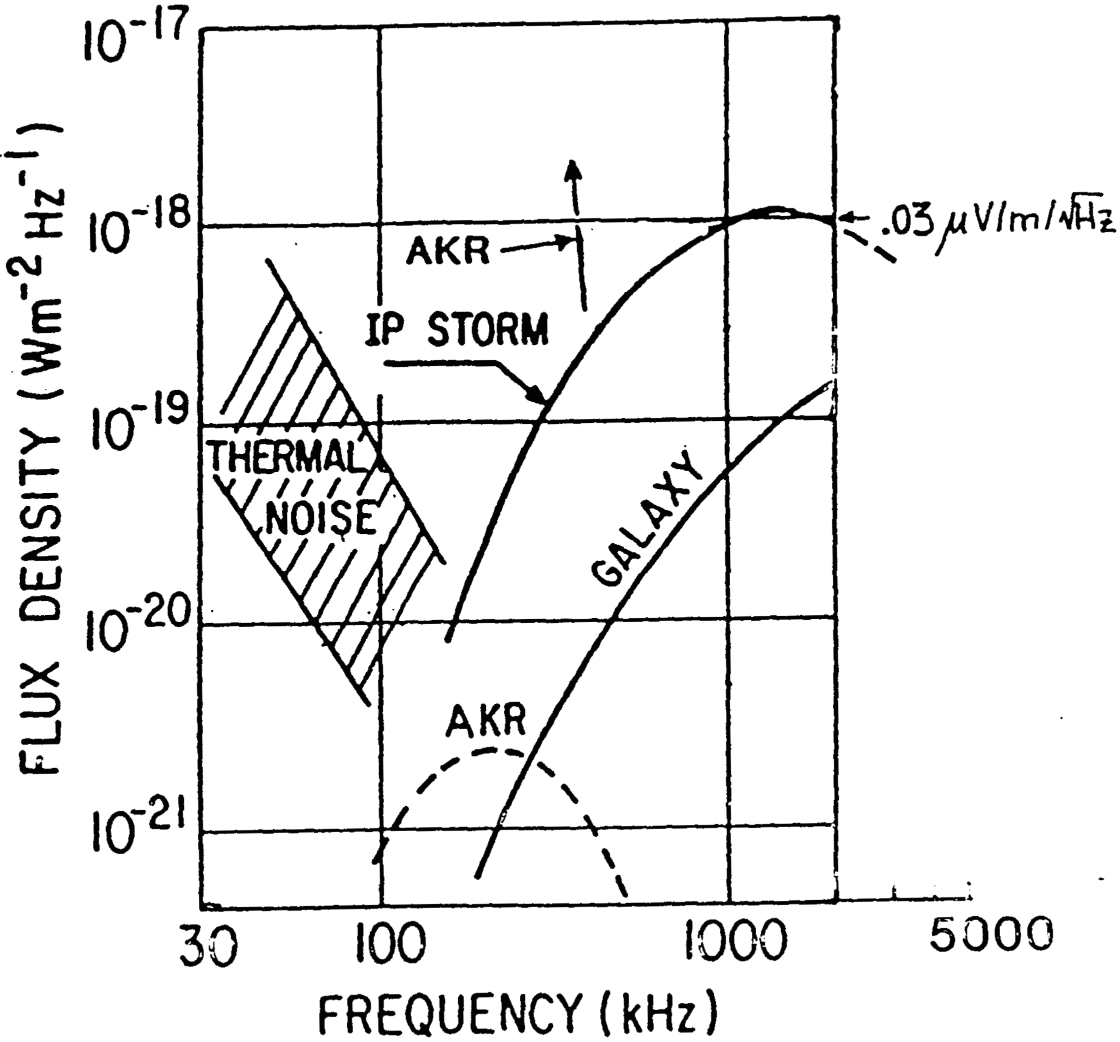
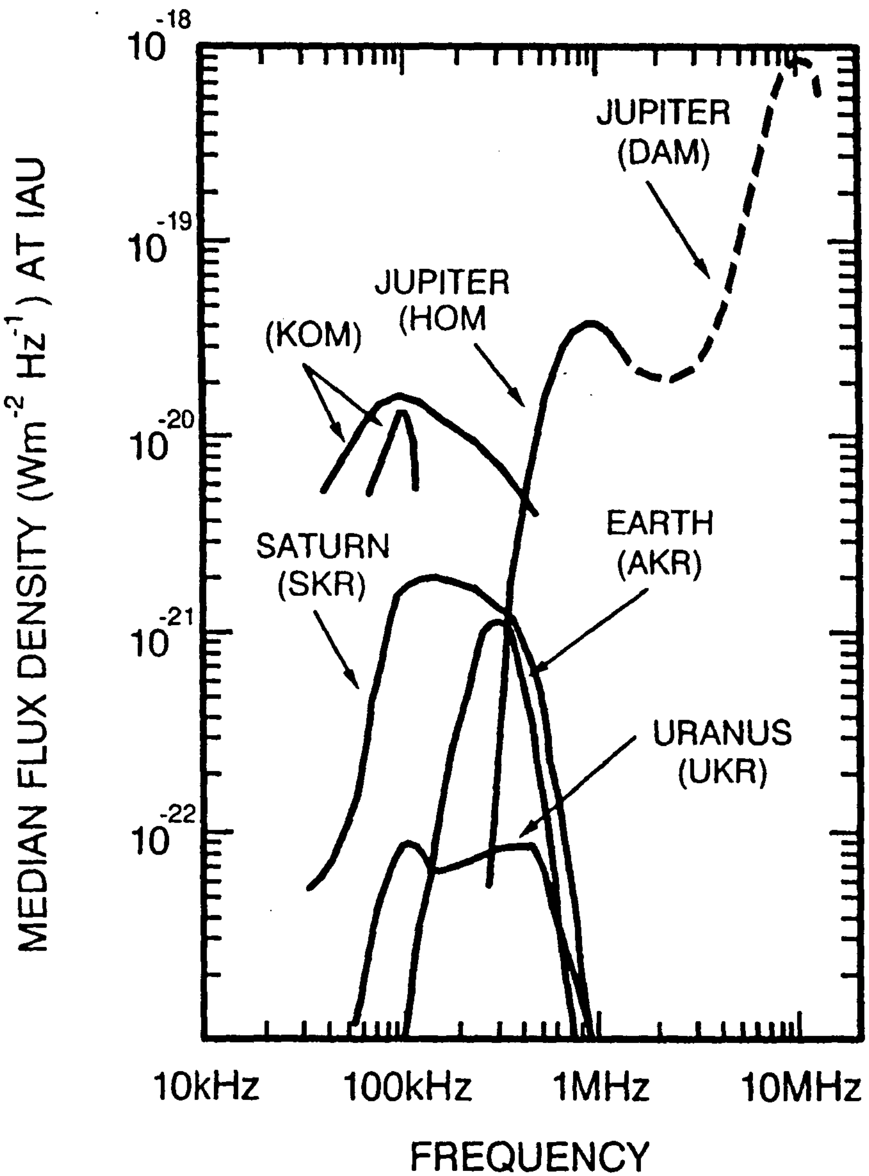


Fig. 9



JUPITER
(DAM)
JUPITER
(HOM
(KOM)
SATURN
(SKR)
EARTH
(AKR)
URANUS
(UKR)
10^{-18}
10^{-19}
10^{-20}
10^{-21}
10^{-22}
MEDIAN FLUX DENSITY (Wm^{-2} Hz^{-1}) AT IAU
10kHz
100kHz
1MHz
10MHz
FREQUENCY

Fig. 10

THE EFFECTS OF SCATTERING AND SCINTILLATION IN THE INTERSTELLAR AND INTERPLANETARY MEDIUMS

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ABSTRACT

Radio waves are scattered in propagating through the interstellar and interplanetary mediums. This effect sets limits to the effective resolution that can be achieved with a low frequency array. Here, we review the "standard model" for scattering, and we use it to estimate the limitations relevant to a lunar low frequency array. At a frequency of 1 MHz, the maximum useful baseline is several tens of kilometers. At higher frequencies the scattering is smaller, and therefore longer baselines can be used to achieve greater resolution. Within these limitations a wide range of forefront scientific investigations would be possible. Regarding propagation phenomena, a lunar array has the distinct advantage of being outside the Earth's magnetopoause in which significant refraction and birefringence occur at low frequencies.

INTRODUCTION

Density irregularities in the interstellar medium (ISM) and the interplanetary medium (IPM) are known to scatter radio waves; and in some cases cause fluctuations (scintillation) in the received amplitude, phase, and intensity. Because these effects occur in an ionized gas, they are stronger at lower frequencies. Not suprisingly, scattering phenomena result in fundamental limitations relevant to any low frequency array.

In what follows, a simplified, heuristic presentation of the "standard model" for interstellar and interplanetary scattering is discussed, drawn mainly from Cohen and Cronyn (1974), and Dennison (1987, 1988). The reader should consult Rickett (1977) and Cordes et al. (1985) for greater detail. Using this model and extrapolating existing data, the limitations relevant to a lunar low frequency array (LLFA) are then determined.

THE "STANDARD MODEL" FOR INTERSTELLAR AND INTERPLANETARY SCATTERING

It is well known that inhomogenities on a range of spatial scales contribute to scattering. The power spectrum of electron density fluctuations can be adequately represented by

$$P_n = C_n^2 \, q^{-\alpha} ,$$

where q is the spatial wavenumber, and the coefficent, C_n^2, quantifies the "strength of turbulence" at any particular location within the medium. The power law index, α, is in most cases thought to be in the range $3.5 < \alpha < 4.5$, with the Kolmogoroff value, i.e. $\alpha = 11/3$ frequently invoked. The power law dependence is generally valid over a range of spatial wavenumbers, given by $q_1 < q < q_2$, where q_1 and q_2 are referred to as the outer and inner scales, respectively.

A common situation is depicted in Figure 1. Because of the density irregularities (and the consequent irregularities in refractive index), a wavefront emergent from the medium is corrugated, whereas it was previously plane, or more generally spherical. The phase fluctuation in the emergent wave decorrelates on transverse scale, L_ϕ. For $\alpha = 11/3$,

$$L_\phi \propto \nu^{6/5} \left[\int C_n^2(z)\,dz \right]^{-0.6}$$

where the integral is taken along a ray path through the medium. A direct consequence of the corrugated structure of the emergent wavefront is that ray trajectories are no longer parallel (for an initially plane wave). Hence, the rays are scattered through various angles. The width of this distribution is characterized by the scattering angle, θ_s, which is given approximately by

$$\theta_s \approx \lambda / L_\phi \quad ,$$

where λ is the wavelength of the radiation.

Weak Scattering

In Figure 1, the observer is located sufficiently close to the scattering screen that the ray displacements in the observer's plane due to scattering are much smaller than the phase fluctuation scale, L_ϕ. That is

$$z\theta_s \ll L_\phi \quad ,$$

where z is the distance to the scattering screen. (Many of the features of an extended medium may be approximately understood using this picture by setting z equal to the distance to the

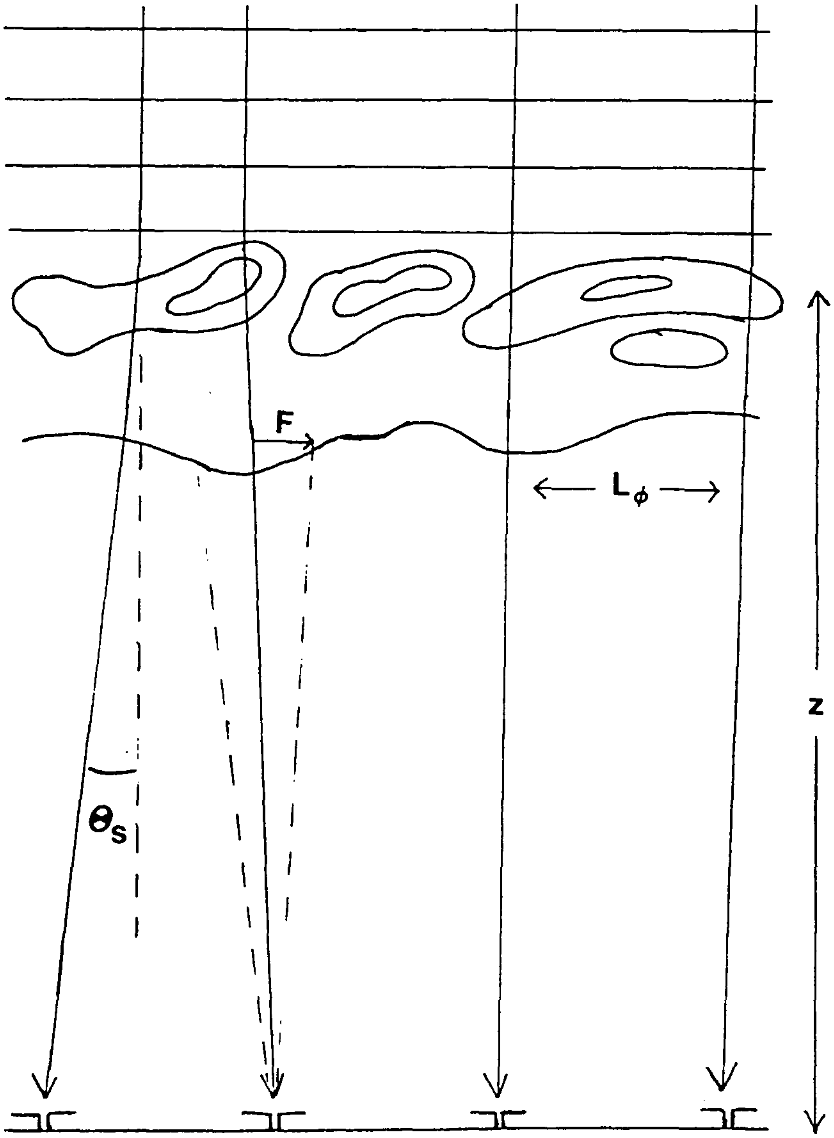

Figure 1 - Weak scattering. Initially plane waves emerge from the screen corrugated, having phase coherence length L_ϕ. The rms deflection is θ_s. The dominant contribution to the diffraction integral for the complex voltage at an array element comes from a region of scale size $F = \sqrt{\lambda z}$, the Fresnel scale. For weak scattering $z\theta_s << F << L_\phi$.

midpoint in the medium.) This condition is known as weak scattering. According to Fermat's principle, the complex voltage at each array element in the observer's plane is obtained by integrating the contributions from all parts of the wavefront, each of which is to be regarded as a radiation source. The dominant contribution comes from a region of scale size $\sqrt{\lambda z}$ at the screen, the Fresnel scale. Since this scale is just the geometric mean of $z\theta_s$ and L_s, we have

$$z\theta_s \ll \sqrt{\lambda z} \ll L_s$$

for weak scattering.

Since the wavefront phase does not fluctuate significantly over the Fresnel scale, the phase at an array element is well determined from the optical path length of the ray reaching that element. Each array element then suffers some "propagation phase." The resultant shift in fringe phase on some baseline is just the difference in propagation phases for the two elements. Baselines much shorter than L_s, would have negligible corruption in the fringe phase ($\delta\phi_{rms} \ll 1$ radian). On baselines much longer than L_s, the fringe phase would be essentially random, fluctuating on timescale $\tau \approx L_s/v$, where v is the transverse velocity of the medium, relative to the line of sight. Note that no degradation of the instantaneous visibility amplitudes occurs. The amplitudes do, however, fluctuate slightly due to diffraction; for the limit discussed here, the corresponding intensity scintillation index is much less than 1.

These effects can also be readily understood as apparent image wander over angular scales $\approx \theta_s$, occuring on timescale, τ. Clearly, the fringe amplitude on baselines $> L_s$ is severely diminished if integration times $> \tau$ are used.

With a sufficiently short integration time ($\ll \tau$), the propagation-induced distortions of the fringe phase can be removed, if three or more array elements are used. This is possible because $(N^2 - N)/2$ independent fringe phase measurements are made per integration period. (N is the number of array elements.) Various well-known analysis techniques such as self-calibration and global fringe fitting have been developed to handle this problem. Therefore, recovery of much, if not most, of the source structure information is possible.

Scattering in the interplanetary medium is weak at frequencies $\gtrsim$ 300 MHz, at solar elongations $\gtrsim$ 15 degrees; and in the interstellar medium at frequencies $\gtrsim$ 7 GHz, at galactic latitudes above about 20 degrees. An LLFA, however, can be expected to operate in domains in which these conditions do not obtain. Hence, it is necessary to consider strong scattering as well.

Strong Scattering

Figure 2 depicts a situation in which the observer is sufficiently far from the screen that

$$z\theta_s \gg \sqrt{\lambda z} \gg L_\phi.$$

This is the condition for strong scattering. Note that since $L_\phi \propto \nu^{1.2}$, and $\theta_s \propto \nu^{-2.2}$, this condition occurs in any case at a sufficiently low frequency. An observer receives an angular spectrum of rays of approximate width, θ_s. If the instrumental bandwidth is sufficiently small ($\lesssim 2c/(z\theta_s^2)$), then interference among the received rays results in significant intensity fluctuations (strong scintillation, with index ≈ 1). Also, the phase in the observer's plane fluctuates randomly, decorrelating on spatial scales $\approx L_\phi$. Therefore, the fringe phase on baselines longer than L_ϕ is random, and it fluctuates on timescales $\approx L_\phi/v$.

For a <u>point source</u> of radiation, the fringes could, in principal at least, be recovered using techniques similar to those discussed in the context of weak scattering. Because both amplitude and phase scintillation are present, each element is corrupted by a complex factor, which varies on timescale, τ. With four or more elements, this corruption can be removed using phase and amplitude closure. The result, however, would be uninteresting unit visibility fringes! To be interesting, a source should have structure on angular scales resolvable by the array. If B_{MAX} is the longest baseline in the array, then to be at least partly resolved, the source should have structure on scales $> \theta \approx \lambda/B_{MAX}$. It is important to compare this angular scale to the critical angle, θ_c, approximately defined as the angle subtended by the phase coherence length at the screen, i.e. $\theta_c \approx L_\phi/z$. If $\theta > \theta_c$, then the source can no longer be considered a point source, different parts of the source suffer uncorrelated phase corruptions, and the true source structure cannot be reconstructed. Conditions in the ISM result in a very small value for θ_c, much smaller than most sources of interest. For ultracompact sources, such as pulsars, the baselines required to resolve angular scales comparable to θ_c, or smaller, are $\gtrsim z\theta_s$. In most situations, this implies exceedingly long baselines, of astronomical unit dimensions at GHz frequencies, given the constraints set by the ISM.

We find, therefore, that the structures examined with an LLFA would be considerably larger than θ_c. In this case, intrinsic structure is irretrievalbly smoothed by an angular distribution of width, θ_s, if the scattering is strong. This sets a firm limit to the achievable resolution. Baselines longer than $\lambda/\theta_s \approx L_\phi$, would show diminished visibility due to scattering. Hence, the longest useful baselines would be several times L_ϕ.

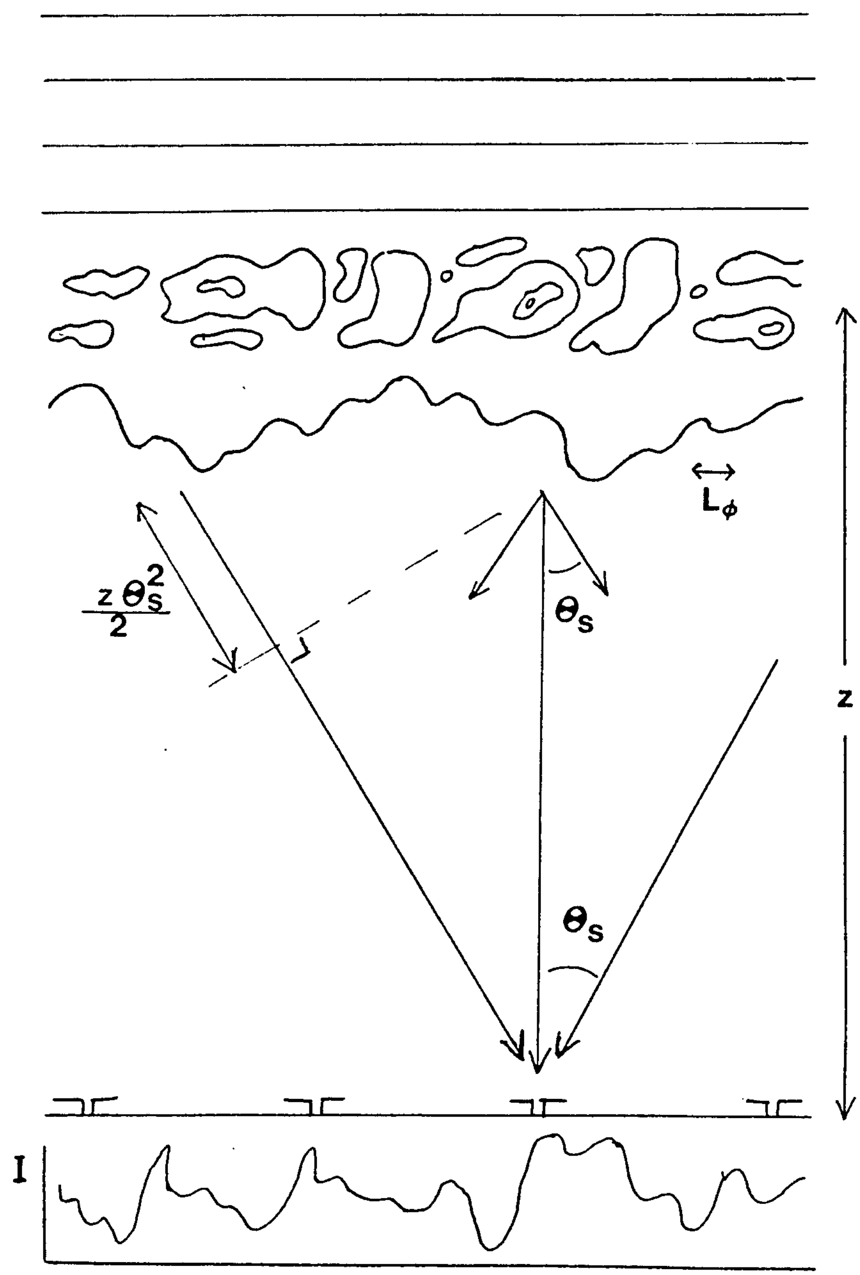

Figure 2 - Strong scattering. In this case the ray displacements exceed the phase coherence scale, and an observer receives an angular distribution of rays of width, θ_s. Interference among the received rays results in intensity scintillations if a point source is observed with short time resolution, and if the bandwidth is sufficiently narrow that approximate coherence is maintained over the extra path length $(z\theta_s^2)/2$. The fluctuating intensity is shown schematically. In virtually all cases of interest, the scattering angle is a firm lower limit to the achievable angular resolution.

CONSTRAINTS ON THE LLFA

The Interstellar Medium

Interstellar scattering is typically strong for $\nu < 7$ GHz, along lines of sight to extragalactic sources at moderate to high latitudes (> 20 degrees) (Cordes et al. 1984). At lower galactic latitudes, the transition frequency (separating strong and weak scattering) is even higher. Clearly, we are well within the limit of strong scattering as far as the LLFA is concerned. Extrapolating from measurements made at GHz frequencies (Cordes et al. 1984), we have for the critical and scattering angles

$$\theta_c \approx [10^{-10} \to 10^{-9} \text{ arcsec}]\ \nu_M^{1.2}$$

$$\theta_s \approx [20 \to 120 \text{ arcmin}]\ \nu_M^{-2.2}$$

for $|b| > 20$ degrees. ν_M is the frequency in Mhz. The extrapolation is somewhat uncertain as the most appropriate value of the power law index is unknown. Also, there is considerable variation in the scattering magnitude from one line of sight to the next. Therefore, these magnitudes must be regarded as typical. Since the effective resolution cannot usefully be much better than θ_s, we find that we are limited to baselines,

$$B \lesssim 50 \text{ km } \nu_M^{1.2}.$$

It is also noted in passing that below about 1 MHz, the galaxy is at least partially opaque due to free-free absorption. Hence, this frequency is probably close to a lower limit for extragalactic, and many galactic, observations.

The Interplanetary Medium

Recently, Dennison et al. (1988) reported interferometric measurements of interplanetary phase scintillations at 327 MHz. Extrapolating their results to lower frequencies yields $L_s \approx 5$ to 15 km at 1 MHz and a solar elongation of 75 degrees. The corresponding scattering angles are 2 to 3.5 degrees. Since under these conditions the scattering is strong, these scattering angles represent resolution limits. Of course, the scattering angles will be somewhat smaller at larger elongation angles. At 1 MHz, then, the maximum useful baselines will be limited to several tens of kilometers. At somewhat higher frequencies, the limitations will much less severe, since $\theta_s \propto \nu^{-2.2}$. These results are approximately consistent with Erickson's (1964) formula for θ_s, which predicts $\theta_s \approx 1.25$ degrees at an elongation angle of 75 degrees and $\nu = 1$ MHz. The minor difference between the two predictions may be attributable to the phases in the solar cycle when the respective data sets were taken.

In general, the limit set by interplanetary scattering is

$$B \lesssim 30 \text{ km } \nu_{M}^{1.2} \qquad \text{(for strong scattering).}$$

As indicated above, interplanetary scattering is strong in virtually all directions at 1 MHz. The transition to weak scattering occurs around 40 MHz at an elongation of 75 degrees, and varies with elongation angle. At the higher frequencies, at which the scattering is weak, the above limitation need not apply, provided sufficiently short integration times are used.

DISCUSSION AND CONCLUSIONS

The resolution limits set by interstellar and interplanetary scattering are comparable at very low frequencies. The maximum useful baseline is of order tens of kilometers at 1 MHz, and increases in rough proportion to frequency. Within these resolution limits, a wide range of forefront scientific investigations will be possible, including mapping of the galactic nonthermal emission, galactic thermal absorption studies, spectral observations of pulsars, supernova remnants, and extragalactic sources, and detailed studies of planetary emissions (Dennison et al. 1986). Indeed, an array of 10 km dimensions may well be quite appropriate for an initial deployment. Later stages could conceivably involve longer baselines which could be used profitably for detailed mapping of individual sources at frequencies ≈ 10 MHz.

Although scattering imposes limitations upon an LLFA, it is also an important object of investigation (Dennison et al. 1986). In this regard, as in other cases, an LLFA would be a unique instrument. By having baselines several times the phase coherence length, the apparent scattering disks of individual sources could be accurately mapped, and properties such as diameter and eccentricity determined. Interplanetary and interstellar scattering could be separated using the dependence of the former on solar elongation, as well as the fact that interplanetary scattering becomes weak at frequencies of tens of MHz. Since a scattering size measurement would effectively be made for every source observed, it would be possible to map the galactic distribution of interstellar scattering. This has not been possible, since at higher frequencies intrinsic source sizes dominate over scattering. The combination of scattering size measurements and low frequency absorption measurements would facilitate a determination of the fractional modulation of the ionized gas density. Eccentricity in the scattering disks would indicate the presence of an anisotropy in the turbulence. By combining scattering measurement at very low frequencies with those at higher frequencies, it will be possible to accurately determine the power law index, and to search for the effects of an inner scale in the wavenumber spectrum. Finally, refractive scattering, if present, would produce noticable distortions such as image doubling. It is widely suspected that refractive scattering occurs in the ISM, in addition to the diffractive scattering discussed above (Rickett et al. 1984; Simonetti et al. 1985).

Another medium which should be mentioned briefly is the Earth's magnetosphere. Close to the Earth significant refraction and birefrince are known to occur at low frequencies. In addition, intense auroral kilometric radiation (AKR) is generated at high magnetic latitudes. The lunar farside has the significant advantage of being well beyond the Earth's magnetopause, and therefore outside the region in which significant refraction and birefringence occur, even at 1 MHz. This site, of course, has the tremendous advantage of being shielded from the interfering effects of the AKR (as well as man-made terrestrial interference).

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PART IV - DESIGN CONSIDERATIONS FOR A LUNAR VLFA

ENGINEERING

for

A Lunar Far-Side Very Low Frequency Array

Stewart W. Johnson

The BDM Corporation

June 1988

PRELIMINARY CONSIDERATIONS

The design drivers for a lunar far-side very low frequency array (VLFA) for astronomical observations from the lunar surface are listed in Table 1. The VLFA must be designed to gather data at frequencies of interest. It must achieve the desired sensitivity to small changes in brightness and also capability to respond to wide ranges of brightness. It must achieve spatial and temporal resolution to meet the needs of the community. A reasonable field of view of an appropriate part of the sky must be achieved. The operational mode must be selected whether it be scanning or aperture synthesis.

Options discussed at the workshop included having a large area sparsely populated with dipoles, a small area densely populated, a small area densely populated with outriggers, and finally a large area densely populated with dipoles.

The individual units of the array were discussed with the outcome being a conclusion that the individual units should be kept simple. Two options were dipoles and three-axis dipoles. Cross-links between the individual units and groupings of units are required and can take several forms including wires between units, radio links, fiber optic links, and some hybrid mix of linkages.

A computer capability is required for the VLFA. The questions discussed related the computer being in situ at the VLFA on the lunar far-side or in lunar orbit or elsewhere. Other questions discussed related to the time-phasing of the establishment of the VLFA and the prospects for enhancing its

Table 1

VLFA DESIGN DRIVERS AND ISSUES

Drivers for Design	Design Results	
• Frequency	LARGE AREA	
• Bandwidth on Frequency	sparse	
• Numbers of Frequencies	SMALL AREA	Computational Algorithm
• Range of Brightness	dense	
very dim to very bright	DENSE	Computer Capability
• Brightness Sensitivity	with outriggers	• in situ
distinguish narrow ranges	LARGE AREA	• orbit
of brightness	densely populated	• other locations
• Polarization		
• Spatial Resolution	Individual Units	Time to Build
(angular)	monopoles	
• Temporal Resolution	dipoles	
(time)	three axis dipoles	Enhancing Capability
• Field of View	Long vs. Short	Over Time
• Operational Mode		
	Crosslinks	
	wires, fiber optics,	
	radio, mix	

capability over time (Table 2). A phased development is desirable commencing with precursor missions to the Moon and evolving to an initial capability which can subsequently be enhanced.

Table 2

PHASED DEPLOYMENT

- Precursor
  - Rudimentary Capability
  - What Frequencies and Mode of Operation...and Why?
  - Area/Volume/Mass/Power/Data
  - Data Processing
  - What do we need to learn early?
- Initial
  - Areal Extent, Numbers and Locations and Sizing of Dipoles
  - Frequencies and Modes of Operations
  - Features for Growth
  - Constraints
- Initial plus Enhancements
  - Fill in and extend area?
  - More frequencies of interest and why?
  - Mass and volume of components
  - Constraints

Initial Design

During workshop discussion it was decided that a strawman preliminary layout would involve 300 dipoles sited in a 17 km circle on the lunar far-side. The suggested location is in the crater Tsiolkovsky which has some relatively flat mare-like surfaces and is at reasonable latitude for viewing and longitude to escape Earth-originating interference. The operating frequencies range from 1 to 30 MHz and the mode of operation is as an interferometer rather than a phased array. The resolution achievable at 1 MHz is to be 1°. A power law distribution of dipoles within the 17 km diameter circle is desirable. The properties of the lunar array suggested are as in Basart and Burns Table II in this volume with incremental dipoles each one meter long, observing frequencies of 1, 3, 10 and 30 MHz; bandwidths such as 20 kHz at 1 MHz; and beamwidths ranging from 1° at 1 MHz to 2' at 30 MHz. The engineering challenge is to design and deploy this array at the preferred site on the far-side of the Moon.

Engineering for the VLFA

The need for the VLFA is to engineer the installation with the technologies that make it possible for the VLFA to perform well for long periods of time with minimal intervention by humans or robots. Contamination and interference can best be limited by reducing the need for nearby operations which produce rocket plumes, space suit effluents, other gases, particulates, ground shock, and extraneous radio frequency signals. Technology needs for the VLFA are listed in Table 3 and the engineering challenges are enumerated in Table 4.

Table 3

Technology Needs for the VLFA

- Deployment Capability (Vehicle with Robot)
  - Delivery to Site
  - Surface Transport
  - Variable Terrain Accommodation
  - Positioning capability
- Dipoles
  - Solar cell
  - Battery
  - Radiation-hardened chip

    receiver

    transmitter
- Communication links between dipoles and central station
  - Broad bandwidth with high radio band (GHz)
  - Receiver Chips with Very-Large-Scale Integrated Circuitry (VLSIC)
- Central Station
  - Computer
  - Shielding (from cosmic rays, UV, infrared, meteoroids, etc.)
  - Power
  - Transmitter/Receiver
  - Thermal Control
  - Solar Flare Shelter for Human Visitors
  - Vehicular servicing/control station
- Computer Algorithms
- Avoidance of Radio Interference
  - Internal versus other VLFA components
  - External-operations not associated with VLFA

Table 4

ENGINEERING CHALLENGES FOR THE ARRAY

- Deployment in a Dusty Environment on Cratered, Unprepared surface; crater depth to diameter ratio 1 to 5; many blocks/boulders even on mare surfaces.
- Performance in Vacuum/Vacuum Outgassing
- Maintaining Calibration in Thermal Environment
  - temperature range 384 K to 102 K.
  - temperature change 5 K degrees per hour at sunset
- Designing Parts/Components for Temperature Extremes and Thermal Gradients

  Note: Number of cycles about one per month compared to LEO with about 480 per month.
- Radiation Hardening/Radiation Shielding for Cosmic and Solar Radiation Particularly for Electronics and Software
- Environmental Degradation (e.g., from Ultraviolet Radiation Induced Degradation of Thermal Control Coatings)
- Micrometeoroid Impact/Damage and Debris

A transportation system is needed to deliver the observatory components (the 300 dipoles and the central station) to the site in the crater Tsiolkovsky on the far-side of the moon. Surface transportation is needed to deploy the 300 dipoles in the desired power law distribution within a 17 kilometer diameter circle. Navigation options for the vehicle are noted in Table 5. Construction capability is needed to set up and shield the central station with its computer, transmitter/receiver, power supply, batteries, and thermal control capability.

Table 5

DEPLOYMENT VEHICLE

<u>Options</u>:

- Inertial Navigation Units
- External Beacon

  Satellite
- Dead Reckoning

  steering angle and wheel rotation and slippage estimator
- Human operator (direct or telepresence)
- Hybrid-combination of capability

The surface mobility units first used for deployment should subsequently be useful in maintenance and repair of the array.

VLFA COMPONENTS AND DESIGN

Table 6 presents some design considerations for the components of the VLFA. Dipoles offer opportunities for innovative design. Each of 300 dipoles involves wires one meter in length, solar cells and batteries for power, and a receiver and transmitter requiring radiation-hardened chips, and thermal control which may be either passive or active to assure that high and low temperature bounds are not violated.

Table 6

Design Considerations for VLFA Components

- Electronics Packages -- Sizes, Power, Reliability, Redundancy, Repair, Aspects of Radiation Hardened Electronics--How Hard? Shielding required vs. hardness aspects; temperature range for operation vs. thermal control; heterodyne receiver?
- Recording of Data - how much, what type
- Data Processing - computational capability on the Moon
- Data Transmission

Frequencies	Uplinks	separation
Bandwidths	Downlinks	

Compatibility with Science

Compatibility with Satellites for Communication

- Power sources

solar	RTG (Radioisotope Thermal Generator)
batteries	DIPS (Dynamic Isotope Power System)

- Life-time of system and how to plan for life (degradation, upgrade capability, etc.)

The communications links between dipoles and the central station offer opportunities for technological innovation and creative engineering. Each of the 300 or more dipoles must have its own radio frequency probably in the GHz range. Very-large-scale integrated circuitry is rapidly advancing (according to Basart) and may lead to a compact light-weight solution to the communications links problem.

Table 7

Options for VLFA Observatory Operation

(The Human Interface)

- Never visited by humans--completely automated (completely remote operation)
- Human presence on site at time of need
  - Initial setup
  - Maintenance and Troubleshooting
  - Upgrading of capabilities
- Human presence via telepresence and telerobotics

The design of the VLFA must involve a human interface aspect that is presented in Table 7. If the site is to never be visited by humans, there is called for a high level of sophistication in automation and robotics which may not be attained in a reasonable time frame. The options for involving humans in VLFA on-site development and upgrades need further trade-off studies before decisions are made as to which option or combinations of options are feasible. Human presence via telepresence and robotics is hampered by the long delay times if the communications are to a far-side site from a near-side lunar base or a Earth-based monitor. Human presence at the VLFA, even periodically, will require furnishing adequate shelter to protect people in the event of a solar flare. From two meters up to 3-1/2 meters of compacted lunar soil may be required according to Silberberg and coworkers (1985).

INFRASTRUCTURE FOR VLFA

A transportation system, communications, data processing/data reduction/interpretation, and Earth-based VLFA science center are essential. The maintenance/resupply network and the upgrading planning/implementation activity are also essential if the long-term mission of the far-side VLFA is to be completed over a ten-year lifetime. Life cycle costing of the VLFA cannot avoid any of these five major categories of support listed in Table 8.

Table 8

Infrastructure for Far-Side VLFA

- Transportation System
  - VLFA Delivery Capability (to far-side)
  - Surface Transportation within an On-site 17 km diameter circle
  - Access from Lunar Orbiting Station - Periodic
- Communications System
  - Data Relay Satellite in Lunar Orbit
  - Data Relay Satellite in Earth Orbit
  - Earth Orbiting Space Station Monitor
- Maintenance/Resupply Network
  - Robotics
  - Telepresence
  - Human EVA Intervention Capability
- Upgrading Planning/Implementation Activity
  - Relates to Earth-based VLFA Science Center
  - Monitor Health/Status of System
  - Programs/Executes Modifications and Upgrades

Table 8 highlights the support the VLFA on the far-side of the Moon will probably require at various locations to function successfully.

Table 9

Mass To Be Delivered to Far-Side Site*

*(Rough estimates)

(Offered to Stimulate Discussion)

- Central Station (could be reduced)

  Note: the ALSEP mass was about 136 kilograms including power/communications, and science packages ~~(could be reduced)~~ according to the ALSEP Review by Bendix Corporation of August 1972 (NASA CR 128597). — 800 kilograms

- Dipoles

  One dipole with solar power, battery, receiver, transmitter, thermal control, shielding -- 2 to 5 kilograms (could be much less) For 300 dipoles — 600 to 1500 kilograms

- Vehicle with Associated Robotics, Sensors, Power, Guidance, Communications, Construction Options — 2000 kilograms

  Note: (fully loaded with two astronauts, suits, and supplies, the Apollo LRV was 708 kilograms)

- Backup Equipment/Redundancy Kit

  (Optional — 500 kilograms

3900 to 4800 kilograms

MASS TO BE DELIVERED TO VLFA SITE

Table 9 presents rough estimates of the mass of the VLFA to be delivered to the far-side site. The VLFA may require more than 4000 to 5000 kilograms delivered to the far-side. Shielding of the central station will be with compacted lunar regolith to a depth of two meters to protect the computer and software from upsets caused by cosmic rays and solar flare radiation. The compacted lunar regolith is a poor conductor so waste heat rejection from the central station may require use of heat pipes. Heat pipe technology is available for such applications.

POWER NEEDS

The most significant uncertainties in masses associated with the dipoles and central station are in the power supplies, batteries and thermal control systems. The day-night cycle on the Moon requires long life batteries which are probably unavailable at this time if solar power is to be used. Batteries are likely to drive up the masses associated with power and thermal control systems. Fortunately, the power needs (Table 10) of the system are relatively modest and battery development is being pursued for other applications.

Table 10

POWER NEEDS FOR VLFA*

*(Rough Estimates for Discussion Purposes)

Central Station	500 to 1000 watts (upper bounds)
Each Dipole	1 watt* (~~higher~~ upper bound)

~~Basart — milliwatts~~

*Basart suggests power needs at each dipole ~~will~~ may be in the milliwatt range

The central station is a candidate for an RTG and DIPS power system. The Apollo Lunar Surface Experimental Package or ALSEP was powered with a SNAP27 radioisotope thermoelectric generator (RTG) which furnished about 70 watts. The Dynamic Isotope Power System (DIPS) requires future development but is being considered for some systems.

THE DEPLOYMENT VEHICLE

Energy requirements for the vehicle to deploy the dipoles will probably be greater than noted for the Apollo LRV. On Apollo 17 LRV energy consumption (Carrier, 1988) was 1440 watt-hours for a distance traversed of nearly 36 kilometers in three EVAs. Vehicle energy consumption in deploying the dipoles will be a function of vehicle parameters and the length of the traverses, the amount of soil compaction as the wheels interact with the soil, the surface roughness, and the elevation change. Soil compaction is a factor here because more compaction leads to greater rolling resistance which leads to greater energy consumption. The cratered, unprepared surface and the low lunar gravity restricted Apollo LRV maximum cruise speed to 6 to 7 km/hr. This LRV could not climb slopes greater than 19°-23°.

The nature of the site determines the vehicle energy consumption. Is the floor of the crater Tsiolkovsky comparable in trafficability to the sites encountered by Apollo 15, 16, and 17 LRV? It may be. That hypothesis of comparable trafficability should be verified before a vehicle and its power supply are decided upon to serve the VLFA. Carrier (1988) points out that a wheeled vehicle will perform satisfactorily if ground contact pressure is no greater than 7 to 10 kPa. We lack experience with more massive vehicles on the Moon. The fact that the Apollo LRV was successful cannot lead to the conclusion that a vehicle twice as massive would respond satisfactorily in the same mission. Ground contact pressure and other factors must be taken into account. It is possible to become stuck with wheels spinning on the lunar surface as did happen with the Apollo LRV. The resolution of this problem was for the astronaut operators to lift the vehicle and move it to better ground. Such an option will not exist for an unmanned or much more massive vehicle.

ENVIRONMENTAL EFFECTS ON VLFA COMPONENTS

Surveyor III components including a television camera, a soil mechanics scoop, and sections of tubing, some polished and some painted, were returned to Earth by Apollo 12 astronauts after 31 months on the Moon. Tests after recovery verified the integrity of most parts even after the extended exposure to the lunar environment. There were some failures which related to thermal cycling (e.g., a tantalum capacitor and some glass envelopes).

Thermal control coatings were noted to have degraded because of exposure to the environment on the Moon. Inorganic coatings originally white became tan in appearance because of solar radiation, adhering lunar dust, and effects of outgassing from spacecraft parts. As the appearance of the coatings was altered, the solar absorptance, which was originally 0.2, more than doubled. Lunar soil particles adhered to all surfaces and were noted to significantly alter the properties of thermal coatings. A small amount of adhering lunar soil could increase absorbed solar thermal energy by a factor of 2 or 3. The Lunar Module (LM) engine was a significant source of dust found on Surveyor III components. Apparently the descent engine accelerated dust to velocities in excess of 100 meters per second so that the effect was to literally sandblast Surveyor III's painted surfaces, even though the LM landing was 155 meters from Surveyor III. Landings near a VLFA will have to be planned to avoid comparable sandblasting and dust contamination by accelerated dust particles. Keep-out distances required may be as great as 300 to 400 meters. Counts of hypervelocity impact pits on

Surveyor III parts place bounds on this meteoroid threat to VLFA parts. Solar wind sputtering was noted to have had little effect on the returned tubing after 31 months exposure.

Several points which apply to the VLFA are apparent from a review of Surveyor III and other data:

- Protection of thermal control surfaces is essential (Degradation with time must be a design factor)
- Shielding of sensitive components from micrometeoroid impact may be necessary
- Laboratory investigations of degradation of operational integrity are necessary during development.

Components of the VLFA should be designed to survive in the lunar environment and then be subjected to extensive tests to assure that degradation will not be excessive over the lifetime of the system. For example, thermal-vacuum tests will be essential in development and preflight preparations of components. These tests are necessary to show that components can operate under cold and hot conditions and survive large thermal gradients. Connections involving ~~of~~ dissimilar metals which result in thermal stress should be of particular concern in the design and testing phases of VLFA development.

PRECURSOR MISSIONS

Enhanced understanding of degradation processes and their rates is a reasonable goal for precursor missions to the Moon. We have a limited knowledge of lunar surface degradation of proposed VLFA components. Our knowledge can be enhanced by revisiting selected Apollo sites and recovering components for study. Also, an effort is needed to quantify the amount of disturbance and dust contamination occurring when an EVA astronaut does maintenance on the VLFA Central Station. Table 11 lists questions to be addressed on early missions to the Moon.

Table 11

ENGINEERING QUESTIONS FOR PRECURSOR MISSIONS

- Position Locations on Far-side Need Improvement (~10 km now in some areas) and better knowledge will help deployment operations.
- Sites-details of topography needed for deployment robot and/or final site selection. Contour maps needed.
- Verify trafficability parameters for specific competing sites for vehicle power/energy needs. Reduce ranges of uncertainty.
- Obtain information for terrestrial engineering tests of deployment vehicle system and risk analyses/trade studies of alternative deployment systems
- Instrumental Long-Term Degradation on Moon (needed to quantify design margins and reduce risk).
- Lunar Ionosphere Uncertainties
    - Diurnal Variation?
- Simplified VLFA with Dipoles Deployed by Inexpensive Means on Moon
    - Impactors
- Terrestrial Precursor of Lunar VLFA
    - dipole arrangement
    - computational algorithms
    - answer what-if questions

THE INITIAL ENGINEERING EFFORT

The VLFA that is finally deployed on the lunar surface should be the result of a phased development that involves careful design and test of each component and includes not only hardware but also software. A terrestrial prototype should be built and tested. It could be operated at somewhat higher frequencies to show the viability of the system and the readiness of components. The terrestrial prototype and associated hardware and software could be applied to the task of reaching a near optimal layout and helping improve algorithms and data interpretation capabilities. The initial terrestrial prototype could be a very simplified version of the proposed lunar VLFA.

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LUNAR FAR-SIDE VERY LOW FREQUENCY ARRAY

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ABSTRACT

We suggest an initial general design for a low-frequency array for the lunar surface. Deployment would occur over several phases. In the first phase, incremental dipoles would be placed in a semiregular pattern over a region 17 km in diameter. This would provide a resolution of 1° at the lowest operating frequency of 1 MHz. Operating frequencies for the array would range from 1 to 30 MHz. The array would operate in an interferometric mode rather than as a phased array. During subsequent deployment phases, additional antennas would extend the array to a larger size.

INTRODUCTION

It is appropriate to initiate the design of a low-frequency radio telescope array at this time, even though a possible launch date for such an instrument is many years away, because the large expense of deployment demands that many stages of planning and re-thinking of the entire design be executed before any earth based hardware construction is initiated. In this paper we discuss the principal attributes of a lunar array, and suggest a base plan from which future plans can evolve.

The justification for installing a radio telescope on the moon is to avoid the effects of the earth's environment. It follows that the telescope would be placed on the lunar far side in order to shield the telescope from radio radiation from the earth. The spectral region for which it is impossible to do radio astronomy from the earth is the low-frequency region below the plasma frequency of the earth's ionosphere. In this region the ionosphere totally blocks all celestial signals. In the range from the plasma frequency to several times the plasma frequency, the ionosphere corrupts the signals so badly that it is nearly impossible to collect reliable high-resolution data. To maximize the amount of information collected versus the cost of implementation, it is appropriate to consider collecting data over as much of the low-frequency range as possible. A suitable frequency range for initiating discussion, is from 1 to 30 MHz.

ARRAY SIZE

With the frequency range roughly specified as above, the next consideration is the physical size of the array. This depends upon the frequency of operation and the resolution desired. In the very low frequency part of the spectrum, the resolution limit will be determined by wave scattering in the interstellar medium. A suitable formula for calculating this is

$$\theta_s = 1.1^\circ \, \nu_{MHz}^{-11/5} \, (\sin|b|)^{-3/5}$$

where ν_{MHz} is the observing frequency in MHz and b is the galactic latitude. If we choose a latitude of 90°, which is the best case, the scattered size of a point source is 1°. While a telescope of 1° resolution seems rather crude, it would be vastly superior to anything available at 1 MHz at this time. The length of an array operating at 1 MHz and 1° resolution is 17 km. This length could be assumed as a lower limit on the array diameter.

The upper limit on array size is determined by the desired resolution at the upper end of the frequency band which produces a smaller scattered size. At 30 MHz the scattering size is roughly 2 arc seconds. The array size for this resolution is approximately 1000 km. This is an appreciable fraction of the lunar circumference may be totally inappropriate to consider. In practice, the upper limit would be set by cost and deployment problems. We suggest a multi-stage deployment program with 17 km being the diameter of the antenna array at the first stage. Subsequent stages would gradually increase the array size, considering both resolution and beam shape, until the maximum resolution at 30 MHz would be obtained.

Array expansion can be divided into two broad categories. The first method is to simply expand the array in all dimensions with the same mathematical law for the antenna element separation as the initial array. The second method of expansion is to place the new elements at a significant distance beyond the initial array. This would increase the resolution more quickly. The trade-off is that large sidelobes are created.

SCANNING VS. APERTURE SYNTHESIS

The two principal modes of operation for an array of antennas are scanning a beam in real time using a phased array, and recording data and later synthesizing a beam via signal processing. The trade-offs between these two procedures are outlined in Table 1. One of the most serious problems for a lunar array is the transportation of a lot of mass to the moon. This problem essentially eliminates the use of copper wire to interconnect the elements. The table shows an alternate method of relaying signals by radio. We will discuss the trade-offs in the suggestions made in the table.

<u>Antenna Element.</u> In a harsh environoment, such as the moon, we do not want any array element that requires mechanical movement. Beam movement must be done electronically. The simplest type of antenna is a dipole whose length could be either an appreciable fraction of a wavelength, such as $\lambda/2$, or it could be much less than a wavelength. In the latter case, the antenna is called an incremental dipole.

The advantage of a long dipole is that it has a directivity that can partially screen out unwanted signals coming in off the side of the main beam. A dipole's pattern goes to zero off the end of a dipole. Patterns for nine dipoles whose overall lengths range from 1/2 to 2.7λ are shown in Fig. 1. In our application, we can think of the physical size of the antenna being fixed, and the pattern varying as the operating frequency varies. A disadvantage of the pattern is that we would have to place a second set of dipoles orthogonal to the first in order to have complete sky coverage. A second disadvantage is that the pattern changes with wavelength. If the antenna were $\lambda/2$ long at 1 MHz, the same antenna would be 15 wavelengths long at 30 MHz. As can be seen in Fig. 1, longer dipoles have complicated patterns. The beams split into multiple lobes pointing in unwanted directions.

Table 1. Scanning Array vs. Synthesis Array

Attribute	Scanning Array	Synthesis Array
Antenna element	Incremental dipole	Incremental dipole
Antenna config.	Roughly n x m uniform array	Nonuniform spacing
Relative number of elements	Most	Least
Element connection	2 x 2 elements in group	2 x 2 elements in group
Group connection	Rows with possible amplitude taper Connection by wire or optical fiber	None
Phase shifters	Within rows and groups	Within groups
Large-scale connection	Radio relay at each row	Radio relay at each group
Phasing control path	Transmit to rows, run wires to groups	Transmit to groups
Sidelobe level	Highest	Lowest
Observing time	Relatively short, depends on integration time	Few earth days
Transmission-line mass	Most	Least
Computer controlled?	Yes	Yes
Backend processing	Simple	Complex

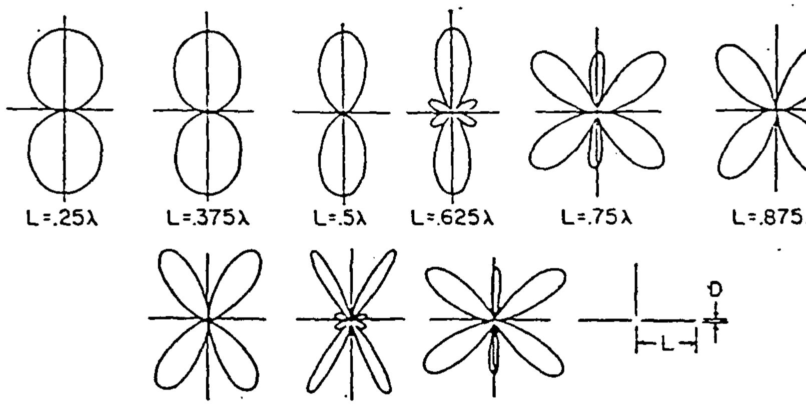

Fig. 1. Patterns for dipoles of varying lengths in terms of wavelength. L is one-half the dipole length. The dipole axis is the horizontal line through the center of the pattern. (After Jasik, 1961.)

The messy pattern situation can be eliminated by using a dipole whose length stays well below a wavelength at all operating frequencies. At the upper operating frequency of 30 MHz, the dipole length could be, say, one tenth of a wavelength. A $\lambda/10$ dipole has a three dimensional pattern like a very fat doughnut with a tiny hole in the center. As the frequency is lowered, the dipole pattern grows in fatness somewhat. Generally, the shape of the pattern would not change very much throughout the entire operating range of the array. While the incremental dipole would allow us to easily track a source without physical movement, it could not reject an unwanted signal coming in from some other direction. Source discrimination is obtained when the dipole elements are used either in a phased array or in a correlator (interferometer) array.

Antenna Configuration. Aperture synthesis imposes the least constraints in the placement of array elements. In an extreme case, elements can be placed randomly. The only requirement is a phase-stable communication link between each element and the control area. On the other hand, to create a real-time beam and scan it with minimal complications requires a dense uniformly-spaced array of antennas. Dense arrays have elements spaced on the order of one wavelength apart. A high density is required to keep sidelobe-levels low. But with a synthetic aperture array, sidelobes are reduced by tracking a source over a wide range in the uv plane (spatial frequency domain) and combining the data later. Consequently, fewer elements are required in synthesis and the spacing is noncritical. Unfortunately, real-time use of a synthesis array is impossible unless the celestial signal source is so strong that sidelobe effects can be ignored.

One way to reduce the number of communication paths between the elements and the control area could be to group the elements in some fashion. One of the simplest groups is a mini-array of two elements by two elements. The elements within a group could be phased to produce a small amount of beam shaping. Phasing information would be sent to the group center where the phasing occurs. For synthesis, no more interconnecting is necessary. The group output would be transmitted via radio, or possible infrared, to the control area.

For a phased array, much more interconnection is needed beyond the group level. One possibility is to connect the groups in rows and then interconnect the rows in a column arrangement. If appropriate, tapering could be applied at the group level or the row level to further shape the beam. Signal strength would be lowered, but the loss generally would not be significant since celestial sources in the very low frequency region are strong. The interconnection between groups could be made with optical fibers to keep the transportable mass low.

One strong disadvantage of a phased array in our application, is that the pattern of the entire array changes with wavelength. Fig. 2 shows a set of antenna patterns for 16 uniformly spaced antenna elements placed in a single line (linear array). The spacing between elements is fixed physically, but changes electrically as the wavelength changes. Horizontally, across the top, the numbers give the spacing between elements in terms of wavelengths. Hence as we change the operating frequency of the array, we are changing columns. Vertically, along the left side, are numbers giving the phase relationship between the elements. T represents the period of the observing frequency. Notice the large variation in shapes of the antenna pattern. We could design an array so as to minimize the pattern variation from one frequency to another, but the remaining variation would still cause difficulties in operation and data analysis.

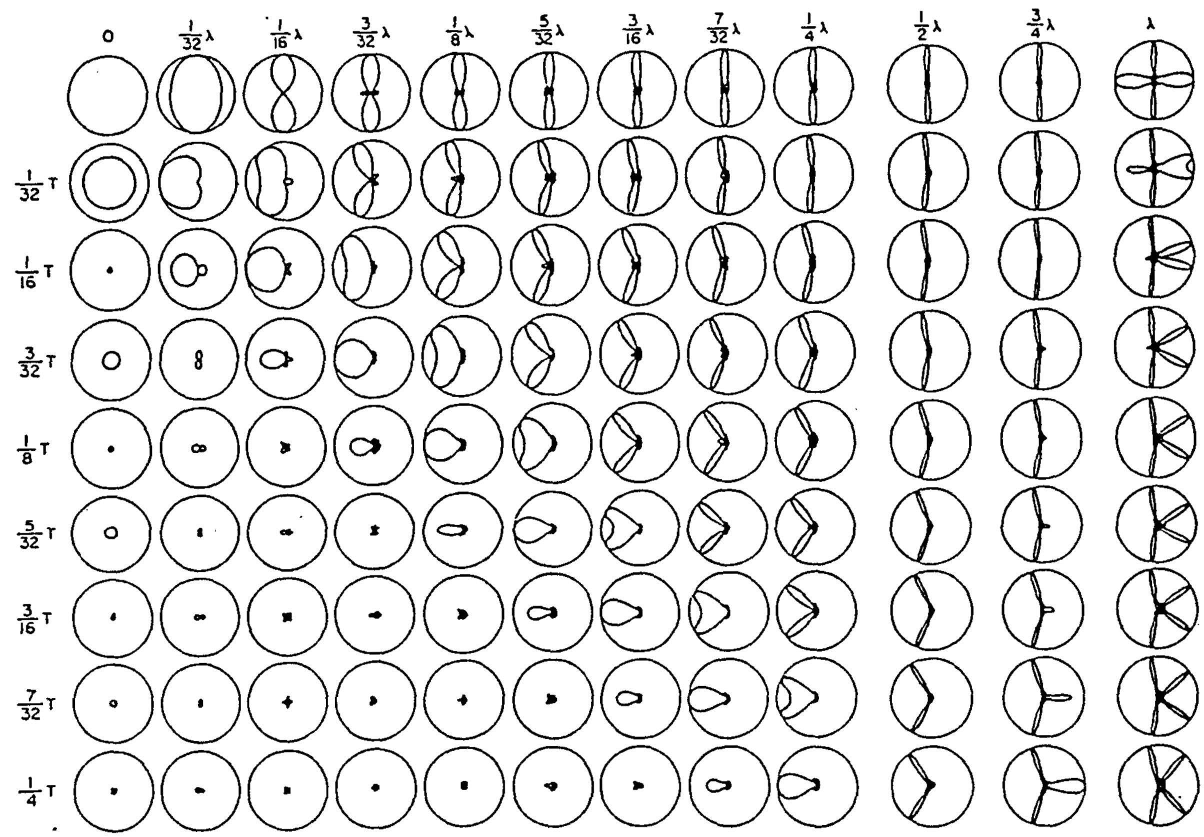

Fig. 2. Element patterns for a linear array of 16 elements. The individual elements have no directivity. The patterns show variations caused by changing frequency, and by changing the phasing between the elements. (After Jasik 1961.)

In both types of array configurations, a problem can arise if there is no beam shaping before the signals are sent from an array element, or elements, to the electronic equipment. Hardware always has a limited linear dynamic range. A strong unwanted signal could drive the electronics beyond the linear dynamic range. The small desired signal would either be lost completely or badly distorted. It may be necessary to shape the beam somewhat before sending the signals to the electronics.

Sidelobe level and observing time. A phased array would have a sidelobe level determined at the time of design. Element location in wavelengths and element type completely specify the sidelobes. This is also true for a synthesis array, but since element spacing is equivalently determined by the tracking time, we can lower the sidelobes by longer tracking. The longest tracking time would be for about 1/2 of a lunar day which would be about 14 earth days. Thus, for low sidelobes, the trade-off is the complexity of a phased array vs. long tracking time for a synthesis array.

Other considerations. Regardless of array type, there would be a computer in a control area. For a phased array, the computer would be busy sending signals to the elements throughout the observation, but there is only one output signal from the array. It would easily be put into an image format by forming a raster. For a synthesis array, any element phasing would be minimal. However, the imaging is complex. The number of antenna elements chosen may have to be limited by the availability of computer power.

ARRAY EXAMPLE

To provide a focus for the characteristics of a very low frequency lunar array, we discuss a specific array element, the incremental dipole, and a square array of nonuniformly spaced array elements.

Properties of an incremental dipole are shown in Table 2 for four frequencies within the operating range of the lunar array. Attributes of the dipole and their units are listed in the left column. The physical length of the dipole is one meter in all cases. This was chosen so as to make the dipole's electrical length (length in wavelengths) 0.1λ at the shortest wavelength of 10 meters. Over this wide frequency range from 30 to 1 MHz, the directivity and the beam solid angle are essentially constant. This constancy is the principal reason for choosing the incremental dipole. No corrections to the mapping operation as a function of frequency will be necessary due to the dipole pattern. A directivity of 1.5 means that the dipole collects very little power more than an isotropic antenna which by definition has a directivity of one.

The fifth row lists the effective aperture area of the dipole. This effective area does not relate to any physical area as in the case of a paraboloidal antenna. The effective aperture area for a lossless antenna is defined as 4π times the wavelength squared divided by the directivity. As a wavelength increases with the directivity remaining constant, we get huge numbers for the effective aperture area. While the effective area of an aperture antenna, such as a dish, is smaller than the physical area, the effective aperture area of a wire antenna is much larger than the projected area of the wire be cause the physical wire intercepts a very small amount of power. Electrically, the wire looks much bigger than its physical size.

The last two rows of the chart illustrate the wide variation in impedance of the dipole over the frequency band. The radiation resistance is a measure of how much power a transmitting dipole radiates into space. The power radiated by the dipole is dissipated by this fictitious resistor. As resistance decreases, the radiated power decreases. The reciprocity theorem of antennas assures us that this property applies to receiving dipoles. Thus, as the radiation resistance decreases, the amount of received power appearing at the antenna terminals decreases. This characteristic is acceptable for the lunar array be cause the celestial signals are strong. The relevant property is resolution, not the total amount of power appearing at the antenna terminals.

Table 2. Characteristics of a One-Meter Dipole

	Frequency, MHz			
Attribute	30	10	3	1
Wavelength, meters	10	30	100	300
Electrical length, λ	0.1	0.03	0.01	0.003
Directivity	1.5	1.5	1.5	1.5
Beam Solid Angle, Ster.	8.4	8.4	8.4	8.4
Effective aperture area, m^2	8×10^2	7×10^3	8×10^4	7×10^5
Radiation Resistance, ohms	2×10^{-1}	2×10^{-3}	2×10^{-5}	2×10^{-7}
Input resistance, ohms	2	2×10^{-1}	2×10^{-2}	2×10^{-3}

The last item in Table 2 is the input resistance. Input reactance, which is not shown, be comes increasingly negative (capacitive reactance) as frequency decreases. The input resistance and reactance determine the dipole's impedance characteristics as seen by a transmission line connected to the center terminals of the dipole. This impedance varies considerably over the frequency band causing a variable power loss due to an impedance mismatch between the dipole and the transmission line. Again, the loss would be sustained rather than minimized by installing impedance matching devices since the absolute power level is not of primary importance. The total amount of power received when observing would be calibrated against a standard celestial reference source.

Fig. 3 shows an example of a dipole with the receiver box attached which contains the receiver and solar power supply. The antenna rising vertically from the box receives and transmits information to the central control area.

Properties of a synthesis array are shown in Table 3. The table has two sections. The left half is for Phase One, initial deployment, in which the array size is confined to 17 km. In Phase Two, the array would be considerably expanded to increase the resolution at the high frequencies. The large dimension of 1000 km may be unrealistically large, but it sets an important goal of nearly one arcsecond of resolution at 30 MHz.

For convenience of this initial design study, array properties were calculated for a square array configuration with elements spaced at locations 2^n wavelengths at 30 MHz in both rows and columns. The smallest spacing is one wavelength (n=0). This configuration would give uv plane coverage similar to that of a uniformly spaced array after several days of tracking a source. This drops the synthesized sidelobes to a maximum of 13 dB below the main beam without any deconvolution procedure.

The number of elements in the two cases, 169 and 361, is minimal considering the large sized of the two arrays. However, the number of correlations between all possible pairs of elements is substantial. The number of correlators shown is for two correlations between each pair. In synthesis we must correlate both real and imaginary parts of the signal.

Table 3. Synthesis Array Characteristics

Attribute	Frequency, MHz							
	Phase One				Phase Two			
	1	3	10	30	1	3	10	30
Length & Width, km	17	17	17	17	1000	1000	1000	1000
Physical Area, km^2	289	289	289	289	10^6	10^6	10^6	10^6
Number of Elements	169	169	169	169	361	361	361	361
Number of Correlators	<——— 28392 ———>				<——— 129960 ———>			
Directivity	8×10^4	7×10^5	8×10^6	7×10^7	3×10^8	2×10^9	3×10^{10}	2×10^{11}
Effective Area, km^2	289	289	289	289	10^6	10^6	10^6	10^6
Resolution	60'	20'	6'	2'	44"	15"	4"	1.5"
Sidelobe Level, dB	-13	-13	-13	-13	-13	-13	-13	-13

The directivity entries in the table were obtained from the theory of uniformly spaced arrays. Hence, they apply to the synthesized beam after several days of tracking, and not to an "instantaneous" beam obtained with a "snapshot" of data. Because of the similarity to a filled uniformly spaced array, the effective aperture area of the synthesized array is essentially the same as the physical area.

The resolutions listed represent the narrowest fringewidths at the various frequencies. Sources not passing over the zenith of the telescope would have lower resolutions decreasing as cosine of the smallest zenith angle during the observation run. Actual synthesized beam widths would be comparable to the figures shown.

The synthesized sidelobe level is stated as 1 dB below the main beam for all cases. It does not decrease for the larger array since the sidelobe level for a uniformly spaced array is always 13 dB down regardless of the array length. One significant effect has been neglected in the array calculations. This is mutual impedance between the elements. At the lowest frequency, many array elements will be less than a wavelength apart. Accounting for the mutual impedance between these elements will alter the array characteristics. Nevertheless, the numbers tabulated will be similar to those obtained from more detailed calculations.

An example of a circular array in the Crater Tsiolkovsky is shown in Fig. 4. The element spacing is dense near the center and decreases radially outward. This is an indication of how the Phase I deployment would appear.

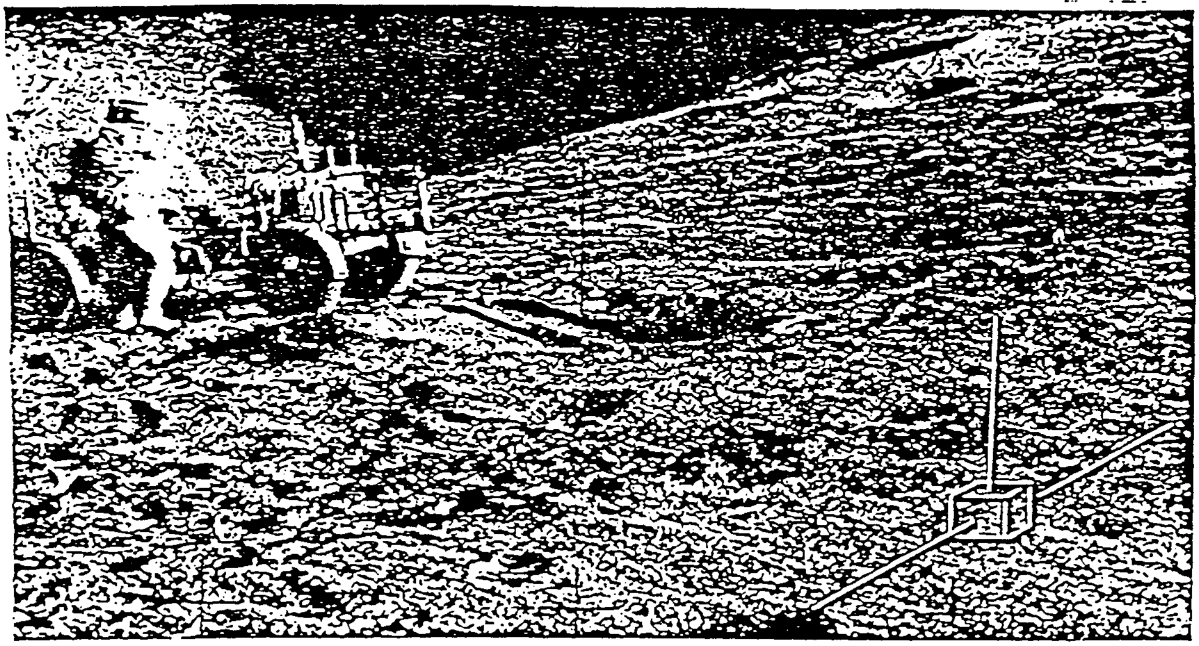

Fig. 3. Schematic illustration of a lunar low-frequency radio telescope illustrating a simple 1-meter dipole antenna laying on the lunar surface, an antenna to relay data back to a central processor, and a box to house the electronics. For scale, the dipole antenna is shown superposed on an Apollo 15 photograph.

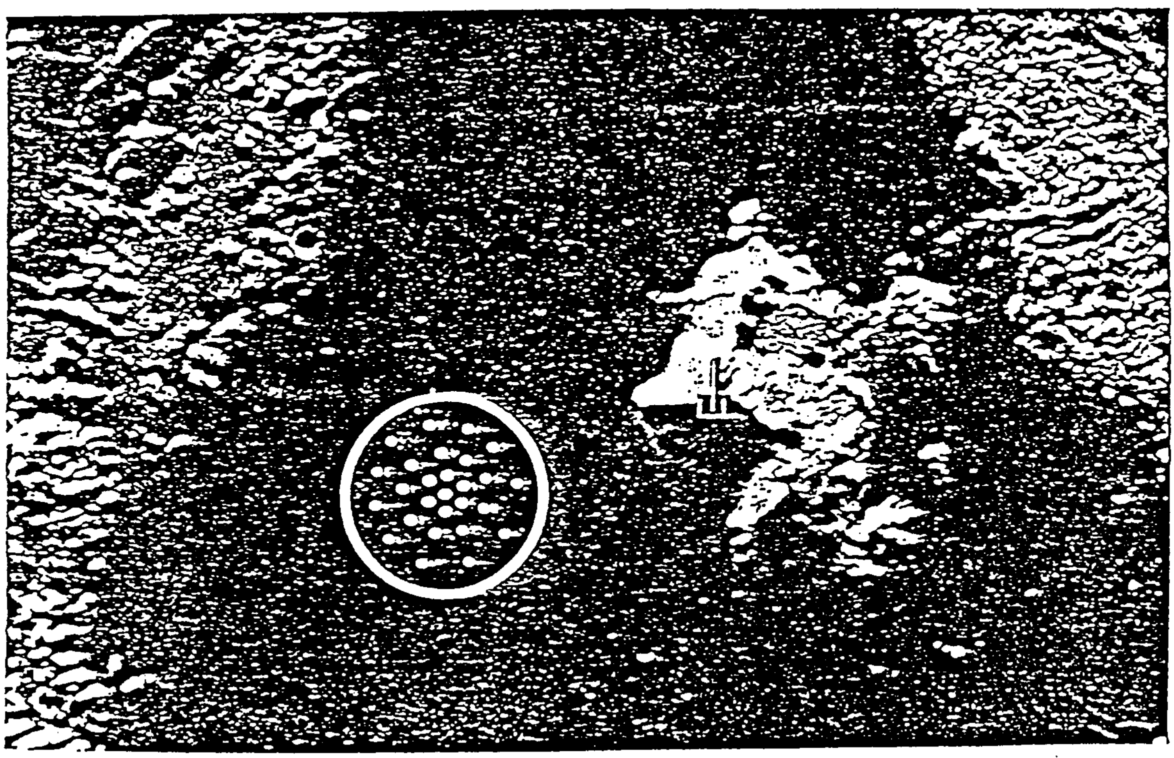

Fig. 4. One possible location for array on lunar far-side within the crater Tsiolkovsky. The circle is about 20 km diameter. Dots represent a subset of 300 dipoles that will be deployed in a circular, power-law pattern.

COMMUNICATION LINK

With either a phased array or a synthesis array, considerable attention must be given to the communication link. For a phased array, many control signals with phasing information must be sent from the central control area to the elements. For a synthesis array, many signals from the antenna elements must be sent to the control area. The simplest communication method which avoids the transportation of mass to the moon is to use either infrared or radio links. Transmitters and receivers for line-of-sight paths would be small electronic chips. One advantage of a radio link, rather than IR, is that the chip at each antenna element could be phase-stable a radio relay. The celestial signal would be upconverted and transmitted on to the control area. The local oscillator at the array element would be kept phase stable by a reference signal sent to it from the control area. A small box containing a solar cell, a battery, and a radiation hardened chip could easily be deployed by each dipole at the same time the dipoles are deployed.

The large number of communication links between the elements and the control area would require a significant amount of bandwidth in the electromagnetic spectrum. To obtain this bandwidth, the transmitted signals would need to be at a high radio band in the GHz portion of the spectrum. However, with the advancing technology in very-large-scale integrated circuits, receiver chips may be available in the appropriate frequency range by the time the lunar radio telescope project is financed.

An example of a receiver-relay is shown in Fig. 5. It receives signals at the four principal frequencies, one at a time. Control signals must be sent from the central control area to each receiver-relay to change observing frequencies. This is done at the frequency at which the receiver is currently operating. The digitally encoded signals are trapped by the decoder and sent to a controller which selects the appropriate bandpass filter, the receiver gain, and the down converter local-oscillator frequency. Upon the completion of this operation, the incoming radio waves from the celestial radio source, containing no digitally coded control signals, are passed through the encoder to the upconverter and then transmitted to the control center.

In addition to frequency switching information, the receiver-relay must receive a master local oscillator signal from the central control area for synchronizing its own local oscillators. This information is sent by radio to each receiver-relay. With a closed-loop path between the central control area and the antenna element, samples from the local oscillators can be sent to the master oscillator for synchronization purposes. The near vacuum of the lunar atmosphere will keep phase fluctuations due to propagation at a minimum.

COMPUTER CONTROL AND PROCESSING

The real-time computer systems' functions are to control the entire array operation, correlate all the signals received from the receiver-relays, and store the output. Several computers can be used for the various functions. One computer controls the transmission and reception of signals to the receiver-relays. A second computer controls the correlator system which itself is a special purpose computer. The data storage computer serves as a link between the on-line system and the off-line system.

The amount of operations to be performed per second depends upon the sampling frequency of the relayed signals. Assume for illustrative purposes that the bandwidth of each receiver is a maximum of 5 MHz. This must be sampled at 10 million samples per second. For 361 receiver-relays, we acquire roughly four billion samples per second. These signals are fed to the 130,000 correlators. The fringe rate is very slow on the moon so a considerable amount of signal averaging can be performed on the correlator outputs,

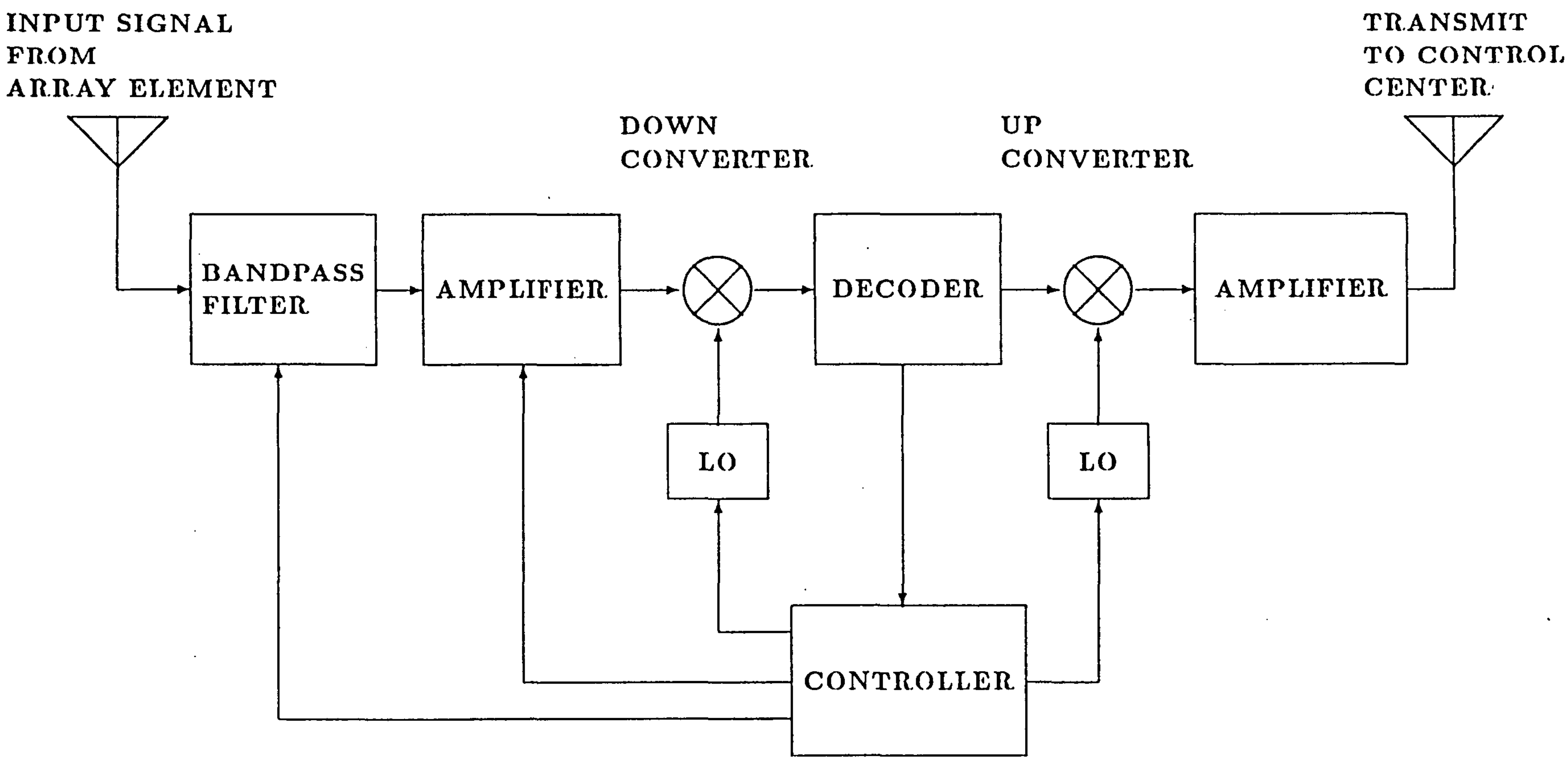

Fig. 5. Simplified block diagram of a receiver-relay. The device receives commands from the central control area through its receiving antenna in addition to the celestial signals. The celestial signal is upconverted and transmitted to the central control area.

perhaps for minutes. Assuming one-minute averages, the system would output about 2000 numbers per second for a single polarization.

At the present rate of performance increase in computer systems, several future minicomputers will be able to handle the computer requirements for the very low frequency array.

PROPOSED ARRAY

Table 4 contains a brief summary of our suggestions for the general properties of a lunar array. We have chosen synthesis over a phased array because of the complications of

Table 4. Lunar Radio Telescope Properties

Mode of operation:	Synthesis
Antenna elements:	Incremental dipoles about one meter long
Characteristic size:	17 km for Phase I 1000 km for Phase II
Impedance matching:	Ignore impedance matching. Large signal levels allow a loss
Observing frequencies:	1, 3, 10, 30 MHz
Bandwidth:	5 MHz maximum with several smaller bandwidths
Beamwidth:	1° @ 1 MHz, 20' @ 2 MHz 6' @ 10 MHz, 2' @ 30 MHz for Phase I 1" minimum @ 30 MHz for Phase II
Dynamic range of electronics	At least 1000
Polarization:	Linear

the phasing in a phased array, and because of the flexibility offered by synthesis. The detailed suggestions are offered as a starting point for further evolution of a design. This array has a minimum resolution of 1° at the lowest observing frequency of 1 MHz and a maximum of 1" resolution at 30 MHz for the Phase II array. One frequency would be observed at a time, but future evolution of the instrument could include simultaneous observations at four frequencies since the incremental dipoles have frequency independent patterns. Also, a second polarization could be added. The technological advancements in electronic chips will keep the hardware at each element to a physically small size. We conclude that a lunar synthesis array that operates at frequencies and spatial resolutions significant for the advancement of astronomical research is technically feasible.

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Fig 1

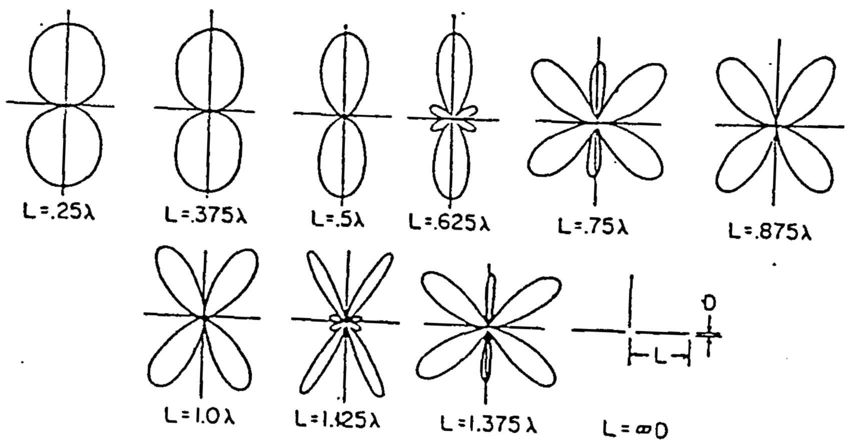

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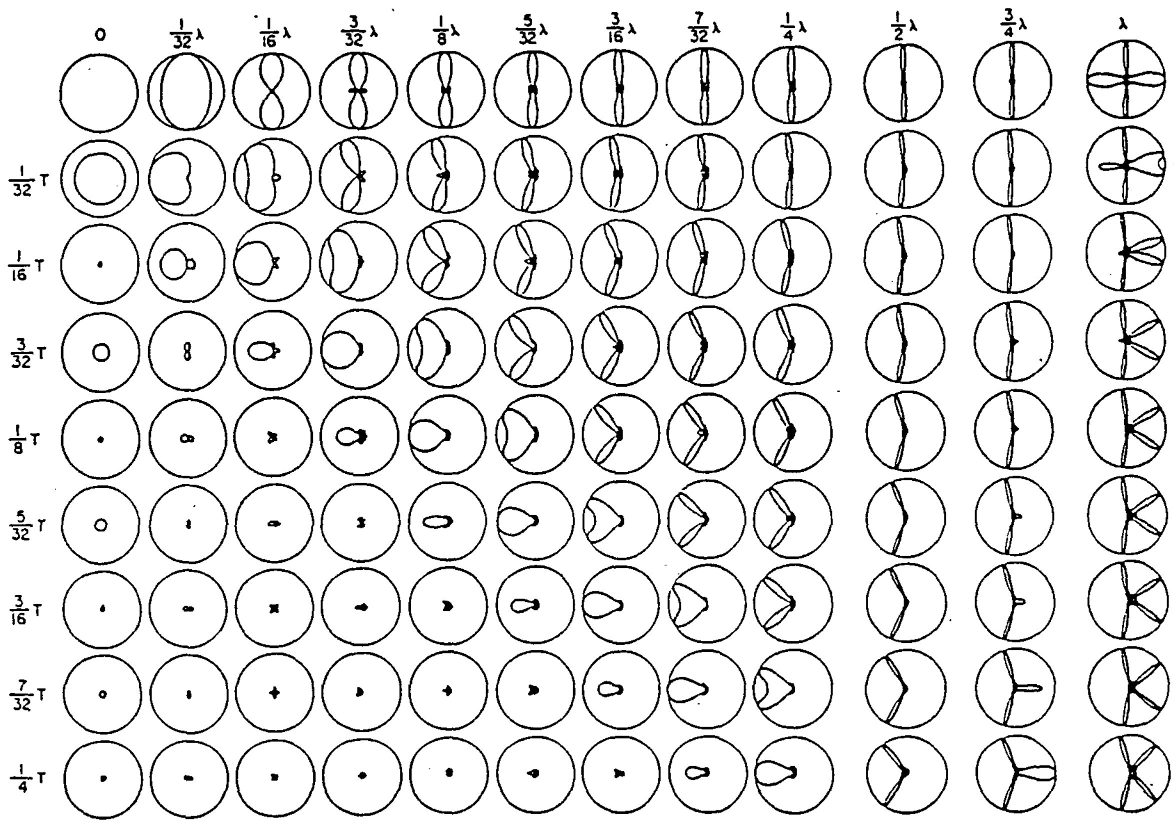
Fig 2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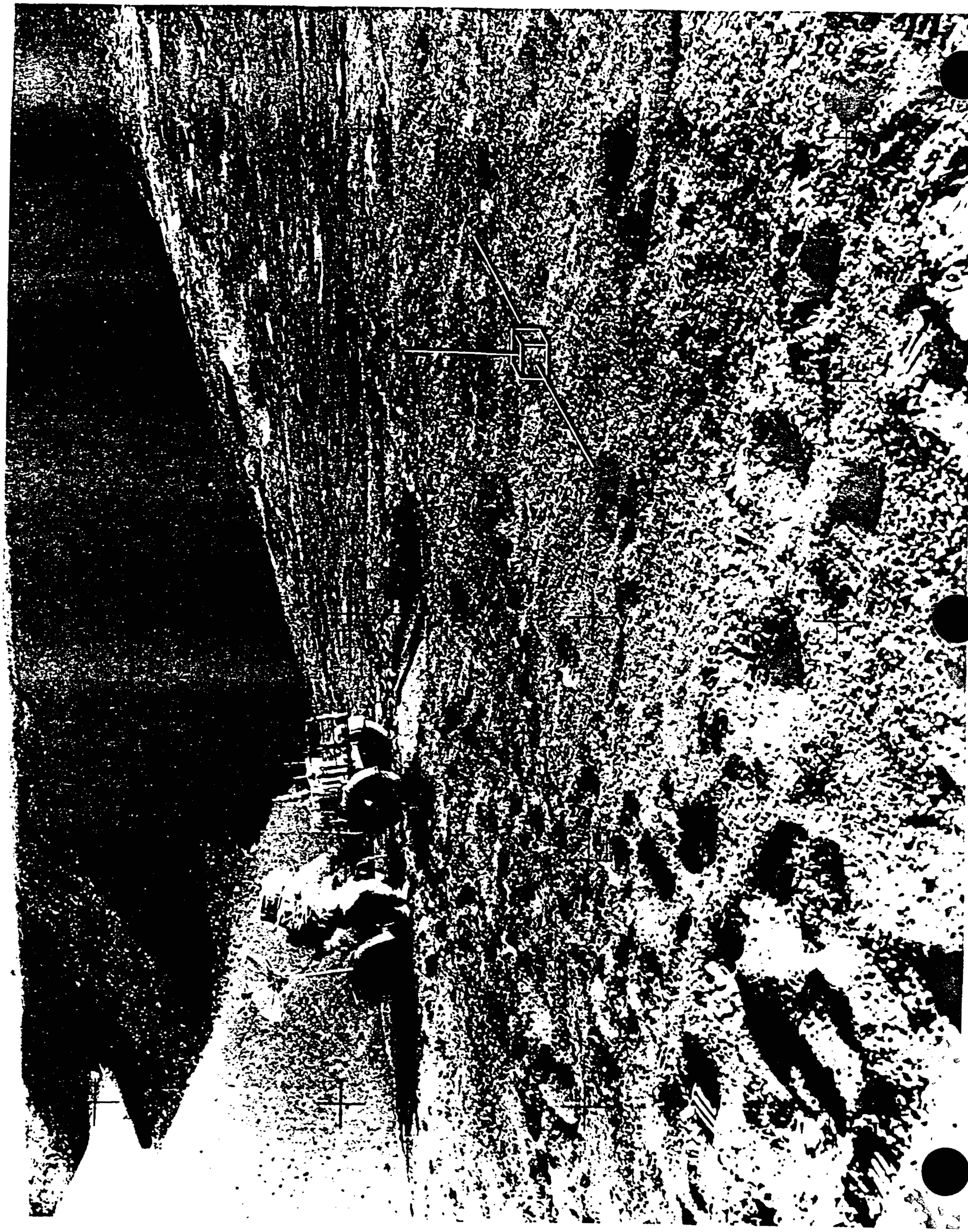

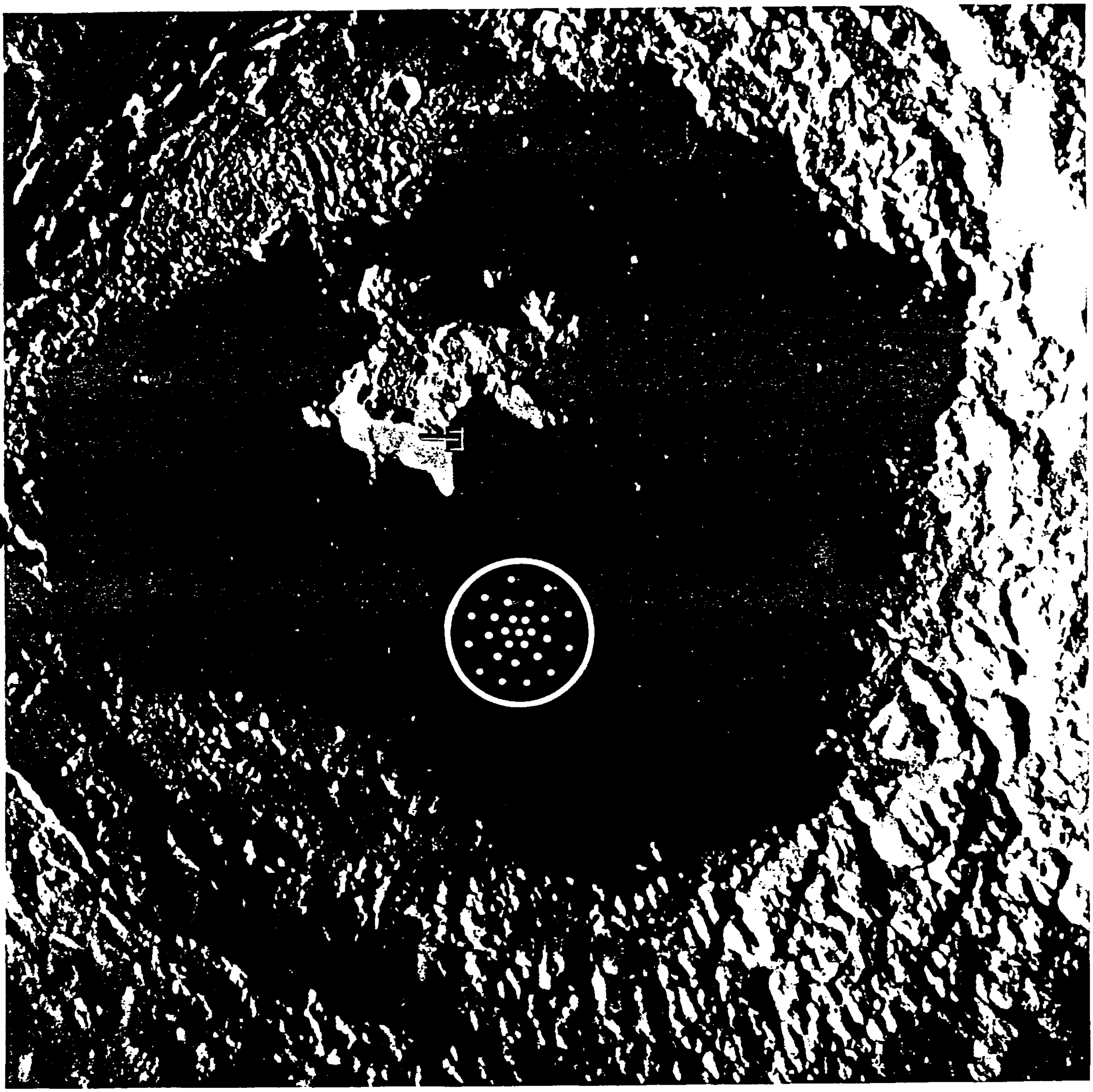

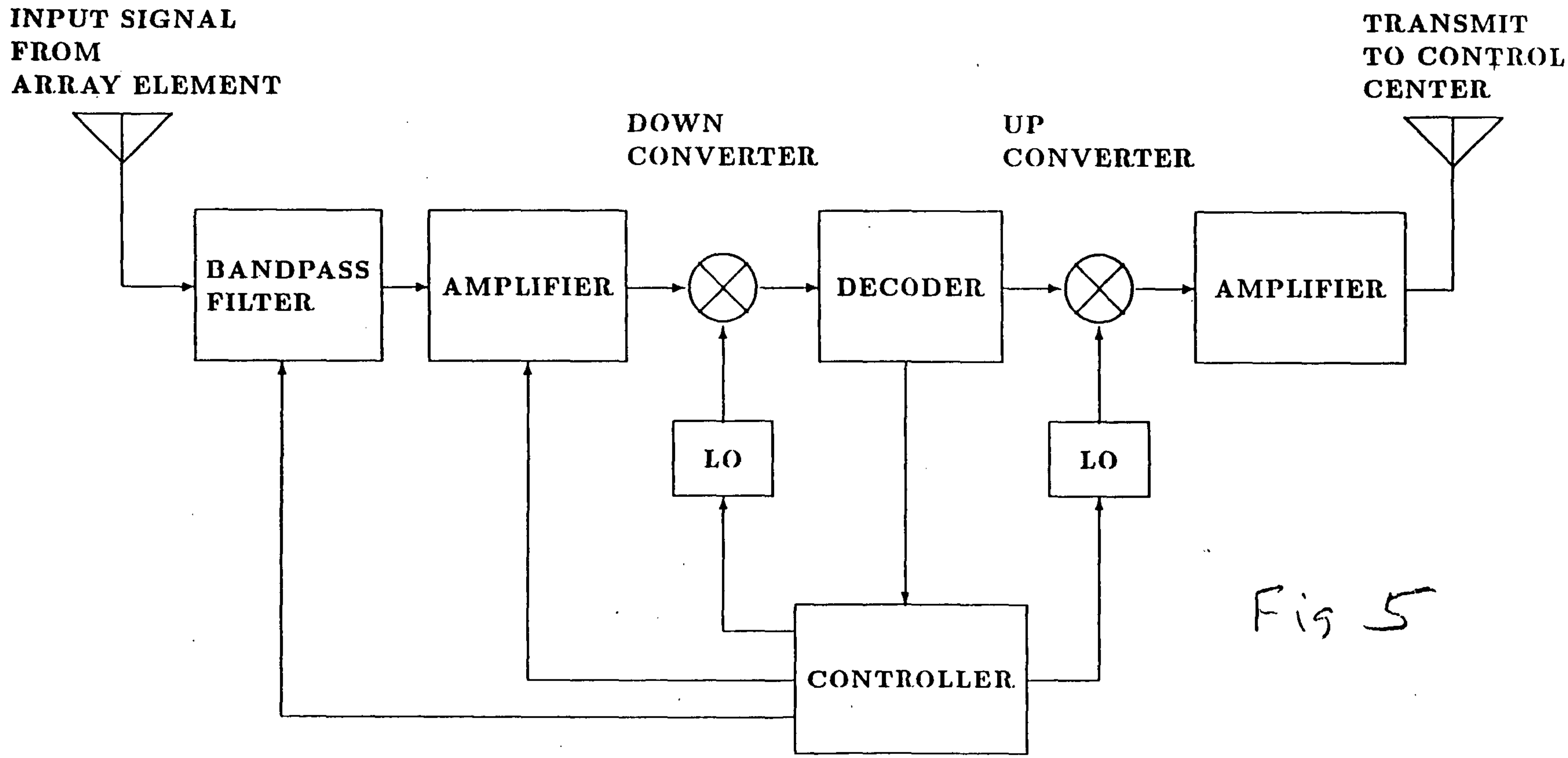
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LO
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Fig 5

PART V - POTENTIAL LUNAR VLFA SITE

SITE SELECTION CRITERIA

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We have identified several criteria that must be considered when determining where to locate an astronomical observatory on the Moon. These are the following: longitude and latitude, topography, distance from a lunar base, value of the site to lunar geoscience, and value as a materials resource.

Longitude and latitude

The high background of very-low frequency (<10 MHz) radio waves emanating from Earth requires that a VLF array be located on the lunar farside. Because of librations of the Moon, only sites with longitudes >98^{o} (East and West) are permanently shielded from Earth. However, because of growing radio-frequency interference on Earth, it is in general desireable to place all radio telescopes on the farside. An exception is the Moon-Earth Radio Interferometer (MERI), which employs one or more radio antennas on Earth, hence must tolerate radio interference. On the other hand, even this would benefit by location on the farside because it might afford a way to distinguish interference from the signals of interest.

To view the entire sky, telescopes must be deployed over a wide range of latitudes. However, a complex VLF (or optical) array is almost certain to be a unique facility, so an optimum latitude must be chosen. Because objects of interest occur in both Northern and Southern skies, it seems sensible to locate a VLF array within 20^{o} of the lunar equator. Also, polar sites are weak for viewing the planets in our solar system as all the objects of interest would be at the horizon.

Topography

The Moon's surface is divided into two distinct terrains, the highland and maria. The highlands compose the oldest lunar crust and are densely cratered. Relief differences are large over relatively short distances. For example, central peaks and walls of large craters can rise 3-4 km above their floors. Some large basins (craters >100 km across) have floors that are relatively smooth and light-colored; the floor materials represent either volcanic flows different in composition from darker mare flows or are impact-generated, fluidized materials (which is the case of the smooth plain on which Apollo 16 landed). Because of the highlands' great age, they are covered with a thick regolith of impact-generated debris, hence tend to contain fewer large blocks of rocks. The maria are younger than the highlands and formed when lavas erupted onto the lunar surface and filled low-lying regions. Mare surfaces tend to be much smoother than highland surfaces and are much less cratered. However, they also have thinner regoliths, so crater ejecta blankets tend to contain numerous blocks of rocks.

Topography enters into the selection of a site for an observatory more for ease of deployment and operation of the facility than for scientific reasons. The rugged terrain in the highlands makes it difficult for elements of an array to communicate by line-of-sight with a single central processing station. Also, deployment vehicles would need to manuver around many hills and valleys created by old, degraded craters. On the other hand, blocks of rocks would be less of a hazard than in the maria. Overall, the optimum site would be a relatively old (>3.5 billion years) mare surface. The old age would permit a relatively thick, unblocky regolith and the presence of mare basalt flows would create relatively low relief across a large region.

Distance from a lunar base

An observatory needs to be isolated from an active lunar base, especially if the base is the site of extensive mining operations. Several factors must be taken into account when estimating how far an observatory needs to be located from a lunar base. These are the distance from a lunar base located on a limb (90^{o} longitude), seismic noise, atmospheric contamination, and dust.

Distance from a limb site. It might be desirable to locate a lunar base close to a nearside limb. For example, the Mare Smmthii region holds great promise for lunar geoscience investigations and for lunar resource extraction (P. Spudis, oral presentation at AIAA meeting, Reno, 1988). To keep Earth in view (for both psychological and operational reasons), the base could be no farther than about 90^{o}E. However, lunar librations cause sites up to 98^{o} to be sometimes in view of Earth. Consequently, a radio array would need to be at least 240 km east of a lunar base located at 90^{o}E longitude (1^{o} equals 30 km at the lunar equator). Furthermore, radio waves from Earth would be diffracted. Assuming a perfectly spherical Moon, the diffraction region for very-low frequency radio waves (300m wavelength) is 75 km (see, e.g., Jackson, 1975, p.447). Thus, this distance must be added to that caused by librations: a radio telescope must be located at least 315 km from 90^{o} longitude.

Seismic noise. Lunar base activities will increase the general seismic background on the Moon. This might affect radio telescope antennas, especially dishes, and would almost certainly affect an array of optical telescopes. Using data from the signal strengths generated by charges placed on the lunar surface by astronauts and from impacts of the Apollo 17 lunar module, Cooper and Kovach (1975) developed an empirical relation between ground motion and seismic energy, $A = kE^{0.5}/r$, where A is the amplitude (nm), E is the energy (ergs), and r is the distance (km). K is an empirical constant, 2×10^{-5}. To estimate the effect of lunar base activity, let us assume that surface mining takes place continuously and calculate the ground motion (amplitude) generated by dropping 1 m^3 of soil from a height of 2 meters. This generates about 6 x 10^{10} ergs, assuming soil density of 2 x 10^3 kg/m^3. This produces the following ground motions:

distance (km)	ground motion (nm)
1	5
10	0.5
100	0.05

The lunar seismic background produces ground motions on the order of 1 nm, so it is clear from the above that even an optical-telescope array will not be affected if it is located more than 10 km from a mining operation. This analysis does include the additive effects of each mining scoop. This would seem to be important because seismic waves are not attenuated rapidly on the Moon; for example, a signal damped out in minutes on Earth lasts hours on the Moon (Lammlein et al., 1974). However, it is unlikely for the signals from successive scoops to be in phase, so they will not simply add to one another. The above analysis also does not consider more potent sources of energy such as blasting operations. We are looking into the effects of these sources. Nevertheless, we can conclude confidently that antificial seismic disturbances will not affect radio observations on the Moon.

<u>Artificial atmosphere</u>. The Moon's tenuous atmosphere makes it ideal for astronomical observation. However, lunar base operations could lead to a significant increase in atmospheric density, as was first pointed out by Vondrak (1974). This problem has been addressed recently by Burns et al. (1988). Even considering the worst case, mining for ^{3}He (which might contribute as much as 1 kg/sec into the lunar atmosphere), Burns et al. (1988) concluded that no significant growth of the atmosphere occurs beyond 10-100 km from a lunar base, roughly the range at which seismic pollution becomes negligible. However, if lunar base activities contributed > 10 kg/sec, significant damage to the environment could occur.

<u>Dust contamination</u>. In principle, this could be a serious problem located within 1-10 km of a lunar base because of dust accelerated by rocket landings and lift offs. However, this could be mitigated by construction of landing pads, so we do not consider it to be a serious problesm, but a quantitative analysis needs to be made. It is almost certainly of little concern for radio telescopes.

<u>Value to lunar geoscience</u>

The site for any astronomical observatory on the Moon ought to be chosen for its suitability for that purpose. Nevertheless, other factors, including operational considerations such as communications, being equal, it seems reasonable to propose choosing the site that has the greatest interest to lunar geoscientists. If this is done, any visit by a crew to repair or expand the facility could include geologic sampling as well. Even during deployment by automated vehicles, geophysical instruments could be deployed as well, although this might add to the cost and complexity of the deployment.

Value as a materials resource

Mining and astronomy are probably incompatible, so sites that hold obvious resource potential ought to be avoided. An alternative would be to designate areas for astronomy (and other sciences) within regions possessing resources, keeping in mind the criteria for distance from a lunar base.

Candidate site for a VLF array: Tsiolkovsky

The large crater Tsiolkovsky (Fig. 1) is an excellent candidate for the site of a VLF radio array. The crater is 180 km across, rim to rim, and its floor is 113 km across, providing ample space for even an advanced array. It is located on the lunar farside at 20°S latitude and 130°E longitude. Thus, Tsiolkovsky is in the equatorial zone and far from any base established on the nearside: even a base on the eastern limb at 90°E would be 1200 km away.

The crater's floor is covered by high-Ti mare basalt (Wilbur, 1978) with an age similar to those of the Apollo 11 landing site, ~ 3.6 billion years. The floor is smooth, except where punctuated by craters. Based on its age, a thick regolith ought to be present, thereby lessening hazards from boulder field near small craters. The central peak rises 3 km above the smooth plains. A central station located on the highest point could receive signals from anywhere on the floor: on a sphere with the Moon's radius, the horizon would be 102 km away when viewed from a mountain top 3 km high.

Tsiolkovsky is also interesting geologically (Guest and Murry, 1969; Guest, 1971; Wilber, 1978). It represents an opportunity to study a relatively well-preserved large crater. The central peak probably represents uplifted, deep-seated materials, an ideal place to study the field relations of highland crustal rocks. It also provides an opportunity to study eruption mechanisms and post-volcanic tectonic processes.

There are two drawbacks to Tsiolkovsky as the site for the VLF array, although neither is a fatal flaw. One is that the walls rise 4 km above the floor, thus limiting the view of the horizon to > 6° above the horizontal (if the array is centered 40 km from the crater wall). The second problem is that the mare basalts that help make the floor smooth are of the high-Ti variety. This makes the regolith in the crater a potential source of ^{3}He, which is found in greater abundance in high-Ti materials. However, development of ^{3}He-based fusion reactors for commercial power production is far in the future and Tsiolkovsky represents only a few percent of the total amount of high-Ti basalt on the Moon, so it would not need to be exploited. Furthermore, although high-Ti basalts are the richest source of He, all lunar soils, mare and highland, contain He in extractable quantities. If Tsiolkovsky turns out to be the best site for the VLF array, it ought to be declared a scientific preserve, closed to resource exploitation.

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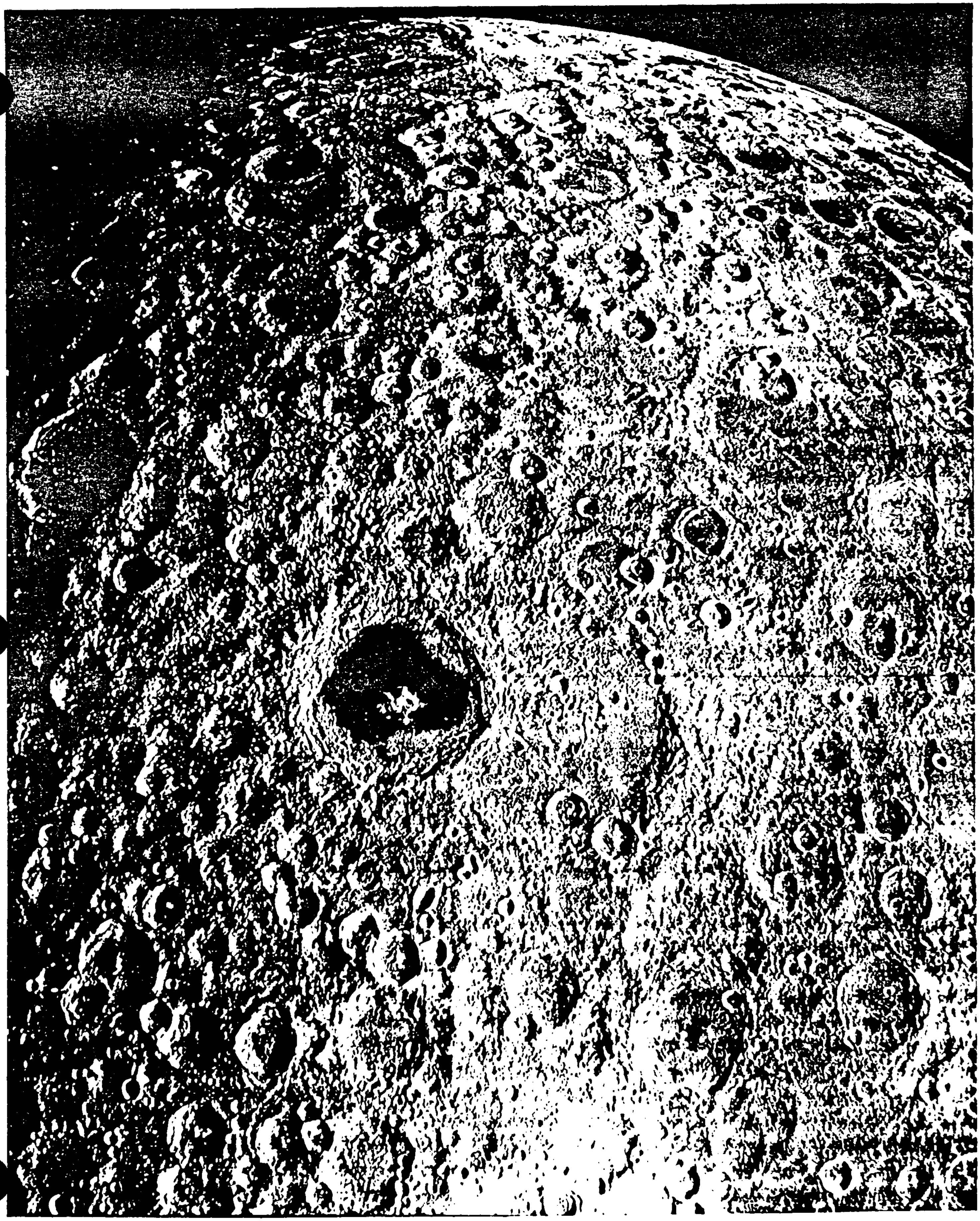

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PART VI - PROPOSAL FOR PRELIMINARY STUDIES

PRECURSOR MISSIONS

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The University of New Mexico

VLF radio observations from the Moon could be made prior to installation of an array. These could be made by including appropriate antennas on missions planned primarily for other scientific purposes, such as deployment of a geophysical package, for engineering measurements, or as dedicated astronomy missions.

I. Precursor Missions

The only lunar mission currently planned by NASA is the Lunar Observer. This mission has been detailed by the Lunar Geoscience Observer Science Workshop Members (1986). It might be launched as early as 1994. Although its primary goals are to gather geochemical, mineralogical, and geophysical data about the Moon, it could carry an antenna to detect very low frequency radio waves. The mission is scheduled to last a year, so the spacecraft could have about six months of observing time shielded from the Earth during the portion of its orbit on the far-side. Such an experiment was proposed for the Lunar Observer's original incarnation, the Lunar Polar Orbiter.

Automated landers have been proposed by various groups, most recently by the Lunar and Planetary Sample Team (1988, report in preparation). They fall into two categories. One class of missions is designed to establish a global network stations (seismometers, heat flow probes, magnetometers, atmospheric monitors). There out to be at least eight of these, approximately half of which would be located on the far-side. These might be deployed by penetrators dropped from orbit, or perhaps by soft landers. The other category of missions is a series of Luna-type sample-return missions. Their purpose would be to perform geological reconnaissance of areas of interest. Geophysical and sample-return missions could be combined. Most importantly, any of them could carry VLF radio antennas, which might be used to establish a long baseline, but low density, array. Such additions to geoscience missions could, besides making astronomical observations, provide valuable engineering data to help design the best possible antennas for the VLF array and would provide a crucial test of the concept.

The above discussion envisions that VLF experiments are additions to missions whose main goals are to better understand the nature and origin of the Moon. However, one can also envision missions dedicated to astronomy, carrying payloads such as a dipole antenna, a radio dish, and an optical telescope for photometric measurements (Zeilik, 1988). Such missions need to be planned in detail.

II. Scientific Objectives of Precursor Missions

There are a number of important scientific objectives for very low frequency radio astronomy that might be met using relatively simple instrumentation on precursor missions. These include:

1. Detailed measurements of the lunar ionosphere. Our knowledge of the density and scale height of the ionosphere on the Moon is very poor at present. Yet this information is of critical importance in developing a design for a far-side VLFA. Is there a 100-m thick layer of ionized gas that hugs the lunar surface on the day side as suggested by Vondrak (1988)? If so, lunar dipole antennas would have to be placed on 100-m poles to rise above this layer of attenuating atmosphere. Short, tunable dipole antennas that transmit and receive low frequency radio waves can be used to probe the lunar ionosphere either from orbit or on the surface. From orbit, this radio frequency system could be used in conjunction with a laser ranging device to accurately determine the scale height of the lunar plasma.

2. All-sky survey at low frequencies. We have very limited knowledge of what the sky looks like below 30 MHz. We desparately need accurate, reliable maps of the sky down to 0.5 MHz for the purpose of detailed planning for the VLFA. Source counts could provide information on the numbers of sources we might expect to observe with the VLFA, the required sensitivity, and could contribute to observational cosmology. This survey would be the natural follow-up mission to the RAE. V-shaped antennas on an orbiting vehicle or on a far-side lander could perform this survey. Either spacecraft could be shielded from the Earth's interfering signals thus allowing sensitive observations free from the strong sidelobes produced by the Earth.

3. Study of signal propagation effects through the ISM. One of the more interesting (and complicating) effects at very low frequencies is that due to scattering by turbulent cells in the interstellar medium (ISM). Images are smeared and attenuated by this plasma process. We have only cursory observational data at present with which one can compare with models. To understand the effects that the ISM will have on lunar VLFA observations and how the VLFA might constrain models of turbulence in the ISM, we need observations at a variety of low frequencies, at a variety of galactic latitudes, and over reasonable time baselines. Such observations could be performd with simple dipole antennas on precursor missions.

4. Monitor variable sources. Flickering of extragalactic sources (i.e., rapid time variability) is believed to be due to scintillations of the ISM especially at low frequencies. High time resolution observations could, once again, constrain the nature of the ISM turbulence using a third dimension of information. Intrinsic variations of active galaxies and quasars are believed to be caused by accretion processes near the central black hole. Variations at low frequencies will sample a somewhat different environment in the outer corona of the black hole accretion disk. Dipole antennas on board long-lived precursor missions could begin sampling the extent and measuring the positions of very low frequency variable sources.

III. Possible Configurations for Low Frequency Observations on Precursor Missions

A. Lunar Orbiter

A V-shaped, short (about 10 meter), low gain antenna would be most useful on an orbiting spacecraft. It would operate over at least a few frequencies possibly centered on a few MHz. The receiver should have a short time constant to allow monitoring of flickering of radio sources. Occultations of radio sources would be an important experimental capability. Therefore, the orbit of the spacecraft should be fairly elliptical with a major axis of about 36,000 km from Earth. The antenna plus receiver would probably have a mass of a few kilograms and occupy a volume of about 0.01 m^3.

B. Lander: Impacter or Soft-Lander

If impacters are used to survey the far-side, then simple short dipoles deployed from the impacters could perform some important observations for both astronomy and geoscience. Such dipoles could be deployed in a manner similar to the telemetry antenna. If several impacters are used, then a long-baseline (hundreds to thousands of km) interferometric array can be established. Such an interferometer could be used to accurately determine the positions of new low frequency sources. Similarly, the array could use known sources to accurately determine the relative positions of the landers (thus providing an important position reference frame for lunar geographic surveys). Such an array might be particularly useful for long-term monitoring of magnetospheric activity of the planets and solar flares.

B. Landers: Rovers

Semi-autonomous rovers could carry a few dipole antennas plus receiver packages and "plant" them during their traverse. Once again, the dipoles could then be linked together as a simple interferometer with baselines over a few tens of kilometers.

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PART VII - SUMMARIES, CONCLUSIONS, AND PROPOSALS FOR FUTURE WORK

~~LUNAR VLFA MEETING~~

Summary Remarks

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The best summary of this meeting and of this subject was offered by a visitor, who remarked that we "had a solution looking for a problem". It is clear that with a modest projection of extant technology a lunar-farside low-frequency telescope could be deployed. It is equally clear that none of the participants in the meeting could offer an astronomical probelm of sufficiently transcendent interest and urgency to justify the investment, although a large number of routine tasks are obvious and, it could be argued, produce a cumulative justificaton. But such arguments are difficult to make, as is the argument for serendipity. So the first job for those interested in a lunar far-side array is to focus the case for low-frequency astronomy, and let such arguments drive the design and deployment strategy.

I. The Case for Low Frequency Astronomy

1. The move the high frequencies

Radio astronomy began with observations at long wavelengths. Nature provided abundant and interesting steep-spectrum radio sources which were readily detected by the relatively low frequency systems which were dictated by the receiver technology of the time and by antenna economics. Observations at these wavelengths were fundamentally limited by the optically active terrestrial ionosphere, which restricted the field coherence length and thus the resolution obtainable, and to add insuslt to injury, the same ionosphere at longer wavelengths propagated abundant and frustrating terrestrial interference into the systems at strengths fully capable of driving them into non-linear behavior.

Radio astronomers accepted these limitations and got on with a first-order inspection of the sky, uncovering phenomena of great importance. But many a radio astronomer of the day, in contemplating the design for a new generation telescope, wished for an observatory on the far side of the moon, or alternatively, for lower noise high-frequency receivers and the funding to put up big dishes instead of wires.

The move to higher frequency is of course what has happened. The benefits of higher resolution (first anticipated and emphasized by Grote Reber in the 1930's) have been obtained, although in practice this was just as much a result of reduced ionospheric phase fluctuations permitting synthesis over very long baselines and of reduced radio frequency interference permitting continuous reliable observations as of the shorter wavelength itself. Steep-spectrum sources are of course weaker at these higher frequencies, but the sensivity of modern systems more than makes up for the weaker flux. Furthermore, new populations of flat or inverted-spectrum sources were uncovered, in numbers hitherto unsuspected. Increased understanding of the emission processes responsible for discrete radio sources pointed to the highest frequencies as being produced most directly by the underlying energy source, with the low frequencies being produced by old electrons. Not only was the observing better (and easier) at high frequencies, but the science seemed more promising as well.

Thus, the centroid frequency of observational radio astronomy has moved from 50 MHz (1930s and 1940s) to 100 MHz(1950s) to 400 MHz(1960s) to 5000 MHz (1970s and 1980s). A data point at 1400 MHz is now considered to be low-frequency, and one at 400 MHz is very low frequency and probably suspect (and often with good reason!). And the frequency range below 100 MHz has been ignored by most astronomers since the 1950s.

2. The terrestrial low-frequency funding cutoff

The only general purpose synthesis telescope capable of operation below 100 MHz – Bill Erickson's TPT at Clark Lake-- was recently cut off from NSF funding and shut down. A similar fate has also befallen the Culgoora array. Although steps to instrument the VLA at 75 MHz are now being undertaken, this recent history has discouraged designers of ground-based low frequency arrays, and should be studied by those who propose low-frequency astronomy in space or on the moon. What causes the lack of interest by the general astronomical community in frequencies below 100 MHz, and particularly in frequencies below 30 MHz?

i. Resolution

The primary reason, as suggested by Bill Erickson during this workshop, has to do with the low resolution of even the most modern low frequency telescopes. To an astronomical community accustomed to seeing maps of sources with arc-second resolution, data obtained at lower frequencies with resolution of a few arcminutes -- ten thousand times worse -- is hardly worth considering, unless driven to it by scientific necessity. But comparing the maps of sources sufficiently nearby to be adequately resolved at low frequencies with their high frequency maps shows only second order differences; thus far nature has not provided a compelling argument for low-frequency source mapping. To be sure, spatially averaged spectral information is obtained for comparison with models, but such information is of ever poorer quality as the wavelength becomes longer, as the resolution becomes worse, and as the precision obtainable becomes limited by confusion and by ionospheric and calibration problems.

ii. Low frequency phenomena play a supporting role

There are a number of phenomena which can best be studied at low frequency, such as galactic synchrotron emission, absorption by HII regions, and some domains of interstellar and interplanetary scattering. While all of these areas have their interested investigators, none have generated widespread interest and excitement; and the data obtained is more of a supplementary nature than a sole or even primary avenue of direct investigation.

iii. Unique phenomena are of specialized interest

Some phenomena can only be studied at low frequency, such as planetary and solar non-thermal emission. These phenomena are of enormous complexity, and are of considerable importance as their understanding involves an understanding of plasma instabilities in configurations which are impossible to produce in the laboratory. The theoretical insights potentially obtainable are of direct relevance both to laboratory plasmas and to the larger cosmic plasmas which are of great interest to astronomers. However, astronomers have generally abandoned the study of these phenomena to solar physicists and to space physicists, and support is diffused across several disciplines. This could be a position of strength, but historically it has been one of weakness -- at least from the perspective of the astronomers.

3. Making the case for low frequencies

An outstanding but ultimately unsuccessful effort was made by Bill Erickson and his colleagues to make the case for ground-based low frequency astronomy as practiced at the Clark Lake Radio Observatory. What additional weight can be added to the arguments in the context of a lunar farside VLFA?

i. What can we do with high resolution?

The resolution obtainable with ground-based low frequency telescopes is limited by the ionosphere at all frequencies below a few hundred MHz, and less fundamentally but equally importantly by the omnipresence of RFI, making continuous operation -- so necessary to many synthesis schemes -- virtually impossible. Even when mapping is attempted by techniques such as hybrid mapping and other restoration schemes, dynamic range is limited by the ionosphere, as is knowledge of position and flux density.

On the lunar farside, absolutely calibrated arc-second resolution maps with good dynamic ranges could be accomplished at frequencies as low as 100 MHz and possibly as low as 30 MHz. The absolute position calibration of such farside maps solves otherwise difficult position registration problems, and the flux scale of the maps would be known. Extending the spectral base of precision mapping in this way down a decade or even two in frequency from 1.4 GHz dramatically expands the catalogue of interesting investigations that could be undertaken. For example, HII regions and SNRs would be resolvable in many more distant galaxies; extremely small knots of absorption could be noticed in our own galaxy. The quantitative results obtainable from the spectrum of such objects could open new possibilities of understanding the nature of the interstellar medium in a large sample of other galaxies than our own. These investigations will also produce information on the integrated emission along hundreds of thousands of lines of sight through our own galaxy, yielding new knowledge of the distribution and homogeneity of the Milky Way plasma.

A concerted effort should be made by interested astronomers (and particularly by theoreticians) both to consider these possibilities and to suggest and work out others.
The questions is: what can we do with VLA-resolution maps obtained at 10 meters wavelength?

ii. Extending the low frequency window

The lunar farside will have an ionosphere -- at least one and possibly two orders of magnitude less dense than that of the earth. This implies farside observations could be made down to 1 MHz or possibly to 100 kHz; but even at 1 MHz other phenomena are beginning to limit performance. Interstellar and interplanetary scintillation limit resolution to about one degree at 1 MHz, and absorption by interstellar ionized hydrogen produces optical depth unity within a hundred parsecs or so. Still, the appearance of the radio sky at one degree resolution below 10 MHz is still unknown, and benefits (other than the obvious but still useful one of extending the spectra of sources known from higher frequency work) can be expected. Quite strong but very steep spectrum sources could exist at 1MHz whose presence would be quite unsuspected from any work done to date. We have no reason to expect such sources -- but suppose brown dwarfs were highly non-thermal emitters and also cosmically abundant (a suggestion due to John Wheeler)? Other obvious lines of inquiry include the distribution of local HII and synchrotron plasma and the behavior of interstellar and interplanetary scintillatiomechanisms.

A coherent program of first-order exploration of this currently unknown territory should be organized, for implementation during an intermediate phase in the deployment of a VLFA on the lunar farside.

iii. Solar system non-thermal emission

Ground-based observations of meter and decameter nonthermal emission from the sun have been made since the 1940s, and of the decameter radiation from Jupiter since 1955. The observational problems in these cases have always been different from galactic and extra-galactic work: the phenomena are quite reasonably strong, so that modest antenna systems suffice for adequate signal (although not of course for mapping!), but the radiation is also quite incredibly complex, leading to dynamic spectral observations on very short time scales in multiple polarizations in attempts to unravel things.

It has also been plain from the beginning that much of interest is just beginning to be visible at the longest wavelengths accessible from the ground, and the early radio astronomy from space, such as the RAE missions and the more recent Voyager missions have uncovered an amazing amount of detailed phenomenology -- and not only from the Sun and Jupiter, but from Saturn, Uranus, and from the Earth as well.

From the point of view of planning low frequency radio observations in space, perhaps the most dramatic result of the early missions was the impressive demonstration by the lunar-orbiting RAE II that earth was an impossibly noisy object at all frequencies, producing antenna powers one to two orders of magnitude higher than background noise at all frequencies, but that when the RAE II was on the farside, the earth disappeared (as expected).

Thus, developing a compelling case for further study -- and particularly for synoptic study -- of solar system nonthermal radiation automatically provides a strong argument for the lunar farside VLFA. As indicated earlier, this is a cross-field subject, without an influential general constituency among astronomers. It is up to planners of the VLFA to organize this as a strength rather than as a weakness, and to involved interested observers and particularly theorists from other disciplines.

Solar system non-thermal radiation is an important and unique observing opportunity for the lunar array, as was cogently argued by Mike Kaiser and Mike Desch at this workshop, and is suitable both for early stages of a lunar observatory, when copies of RAE or Voyager hardware could be deployed with only slight modification, and for succeeding stages, when increased sensitivity and mapping capability could support an ever more complete characterization of the phenomena.

II. The Lunar Farside VLFA

The lunar farside has appealed to daydreaming radio astronomers for decades, as it has been presumed to supply the cure for two very real limitations to telescope performance: man-made and natural RFI(particularly important at decameter wavelengths), and irregular refraction and absorption in the atmosphere and ionosphere. (important at any wavelength). The extension of the observable radio spectrum one or two decades lower in frequency is a bonus of potential if not demonstrable great importance.

What are the limitations on performance on a lunar farside VLFA? What are some practical forms such an array might take? And what are the areas where further investigation is merited before some preliminary version is needlessly frozen into too many people's plans?

1. Performance limitations

i. Geometry

The equatorial diameter of the moon is 3476 km; the diameter of any array must be less than this if the vital benefit of shielding is to be enjoyed. Inasmuch as the various elements of an array must be capable of seeing the same bit of sky at the same time, the north-south dimension must be significantly less than this to make a reasonable range of lunar declination accesible to the array: if a range of $\pm\delta$ is desired, the extreme range of lunar latitude useable is $\pm(90-\delta)$. Librations and diffraction of RFI around the lunar limb must also be considered. A symmetric aperture of diameter of 2000 km would lead to a range in lunar declination of about ±55°. Of course in principle the entire farside could be sprinkled with elements, and appropriate and best subarrays used for various directions in the sky. For purposes of further discussion, though, let us consider a 2000 km aperture, which would correspond to a one arcsecond beam at 30 MHz and a one-half arcminute beam at 1 Mhz.

ii. Scattering

Scattering by the interplanetary medium (IPS) will affect all observations, and interstellar scattering will affect observations of galactic and extragalactic sources. Both processes probably vary as the 2.2 power of wavelength, with ISS being about 0.7 arcsec and IPS being about 2.4 arcsec at a frequency of 30 MHz (these numbers are from Douglas and Smith, and are in rough agreement with numbers presented by Dennison at this workshop). At 1 MHz, the scattering angles become about 0.4 degrees and 1.2 degrees. If these numbers are taken to be correct for the sake of argument, the lowest frequency at which one arcsec mapping would be useful would be about 45 MHz, requiring an aperture of about 1340 km, while the largest worthwhile aperture at 1 Mhz would be about 15 km.

iii. Absorption

For galactic and extragalactic observations one must take into account the increasing optical depth due to interstellar HII. The optical depth is proportional to wavelength squared, and is a few tenths at high galactic latitude at 1 MHz for lines of sight that don't intersect HII regions. At low galactic latitudes, optical depth becomes large before one gets out of the local few hundred parsecs. At 1 MHz discrete HII regions will be opaque, even if of very low emission measure. Of course, this state of affairs is one of the things that makes observations below 30 MHz and particularly below 10 MHz astronomically interesting -- but it will certainly produce a patchy view of the external universe.

iv. Lunar ionosphere

The moon certainly has an ionosphere -- the question is: how dense is it? Participants in the workshop were all mildly surprised that no one seemed to have better numbers than those quoted by Douglas and Smith, which came from lunar occultation observations in the 1960s. Those observations suggested an upper limit of $100/cm^3$, corresponding to an electron plasma frequency of 90 kHz. Apollo observations showed the presence of a neutral atmosphere of 10^{-12} torr; if fully ionized, this would correspond to around $4(10)^4/cm^3$, or a plasma frequency of 1.8 MHz. It is clearly vital for planning purposes to clear this matter up. Not only is the lower frequency limit of utility of the VLFA affected, but even at higher frequencies, resolution may be limited by the lunar ionosphere rather than by IPS.

2. Practical forms of the VLFA

i. Some basic assumptions

The entire spectral range below 50 MHz constitutes a bandwidth equal to one VLA channel -- well within the routine data handling capability of modern digital circuits. Noise figures of front ends are low, and furthermore, the sky brightness temperature goes up (as $\lambda^{2.5}$ or so) with wavelength so that antennas whose efficiency decreases with wavelength can be tolerated without reduction in system signal-to-noise. These considerations led to a consensus of participants that it was probably possible to contemplate individual self-contained short tri-pole elements which are individually in communication, via satellite or optical link, with a central processor (located possibly on earth, with daughter processors on the lunar surface or in appropritate orbit). It was also assumed that calibration could be achieved by lunar orbiting calibration transmitters, thereby removeing the usual requirement of individual interferometer baseline elements which could see enough sources to calibrate themselves. Of course, all these assumptions about technology need to be checked out.

ii. The individual array elements

Elementary considerations of the theory of short antennas and matching circuits suggest that the efficiency of impedance-matched short dipoles is inversely proportional to wavelength cubed, and inversely proportional to the fractional bandwidth of the matching circuitry. It may be, when matters are carefully investigated, that elements more nearly 10 meters in length than one meter would be required at the lowest frequencies -- even after taking the sky brightness into consideration -- if one wishes to avoid degradation of the speed of the array. But it may be that the well-known preference of low noise amplifiers to operate in a mismatched condition removes this problem. It may also prove advantageous to accept the degradation (if any) for the benefit of the shorter dipoles. This sort of investigation could be begun at any time.

The practical advantages of a synthesis array of individual and self-contained short dipole units were pointed out: the elements can be deployed by impacter, by robot vehicles which have been soft-landed, as well as by astronauts; their location need be known only approximately, and even that knowledge can come after the deployment; the elements can probably be designed so weight and power requirements are minimal, and tens or hundreds may be deployed per trip. The elements themselves are all identical, and considerable investment in their construction is possible, with custom VLSI chips housing the electronics, together with carefully designed solar power systems and deployment devices to cause the system to spring upright after having been literally tossed on the surface.

The specifications of the electronics package interacts with what is deemed to be possible technically. Ideally, the system should be broadband -- say 50 MHz to 100 kHz instantaneously -- but in all likelihood this will be impossible. Alternatively, it should be multi-channel in that range, or perhaps dual-mode -- broad-band and inefficient (for solar system studies) and multi-channel and efficient (for galactic and extra-galactic studies). Clearly, the data rate from each element is also a consideration, and probably will be the limiting factor in the context of handling hundreds or thousands of elements. Although it would be premature to attempt a final specification now, a first trial balancing of all of these factors would be highly desirable, and has not yet been done.

iii. The array configuration

Given the concept of N individual self-contained elements, it is a straightforward matter to run computer simulations of the performance to be expected (subject to assumptions about a possible lunar ionosphere). Enough such simulations have been run in other contexts to establish that the resolution of the system will be that of the aperture over which the elements are spread. The effective area of the system will be the number of elements times the effective area per element and the dynamic range will depend on the number of elements and on the accuracy of calibration. Thus,

$$(\Delta S)_{array} = \frac{2kT_{sys}}{NA_{el}\sqrt{\beta\tau}} = 1.341(10)^{-26}\,\frac{T_{sys}}{N\lambda^{1.5}\sqrt{\alpha\tau}}$$

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for short dipoles, where $\alpha=\beta/f$ is the fractional bandwidth. The system temperature T_{sys} will be set by the sky brightness temperature T_B, which increases rapidly with λ down to about 4 MHz, and then more slowly. Taking the average sky brightness from Alexander (1970), which was based on ground-based and RAE data

$$T_B = 10750\lambda^{1.324} \text{ (f<4 MHz)} \qquad T_B = 52.0\lambda^{2.53} \text{ (f>4 MHz)}$$

we have

$$(\Delta S)_{array} = \frac{5284f^{0.176}}{N\sqrt{\alpha\tau}} \text{ Janskys, for f (in MHz) < 4 MHz}$$

and

$$(\Delta S)_{array} = \frac{24820f^{-1.05}}{N\sqrt{\alpha\tau}} \text{ Janskys, for f (in MHz) > 4 MHz}$$

The brightness temperature sensitivity of a synthesis array depends on the filling factor and on the bandwidth and integration time:

$$\frac{(\Delta T_B)_{array}}{T_{sys}} = \frac{A_{array}}{NA_{el}\sqrt{\beta\tau}} = \frac{D^2}{2632N\lambda^{1.5}\sqrt{\alpha\tau}}$$

where D is the diameter of the equivlent synthesized aperture. The fractional error in brightness temperature reaches unity for $D = D_1$, where

$$D_1 = 51.31N^{0.5}(\alpha\tau)^{0.25}\lambda^{0.75}$$

The fractional error in brightness temperature at aperture diameter D is then $(D/D_1)^2$.

iv. Performance of a benchmark array

For purposes of a benchmark, let us consider a lunar array of N=100 elements, with 10% fractional bandwidth (α=0.1), and a $\tau = 10^5$-second integration. The column labelled S_{408} is the flux density at 408 MHz of a steep spectrum (spectral index = -0.8) point source which would have a flux of ten times the rms map noise at the observing frequency.

Table 1

Properties of an Array with $N\sqrt{\alpha\tau} = 10^4$

f(MHz)	$(\Delta S)_{array}$	S_{408}	D_1 (km)	$\Delta T/T_{sys}$ (for D=20 km)
1.0	0.53 Jy	0.043 Jy	369.86	.0029
3.0	0.64	0.126 Jy	162.26	.0152
10.0	0.22	0.114 Jy	65.77	.0925
30.0	0.07	0.087 Jy	28,85	.4806

The rms map flux density is independant of the diameter D of the equivalent synthesized aperture and is adequate for studies of steep-spectrum point sources to a 408 MHz flux density of about 0.1 Jy, or about two hundred thousand objects.

To attain near arc-second maps one uses a large D (say, 2000 km) at 30 MHz. Such a system could produce 100-pixel maps for sources with 408 MHz flux brighter than about 9 Jy. This performance is useful, but by no means matches the VLA. One order of magnitude improvement would be realized by ten times the number of elements, or by stacking 100 maps (each of which represents a day's observation). It could also be attained with N=100 elements, each of which has a gain of 10 relative to a short dipole, but with attendant increased deployment and control problems.

The 100-element system when deployed to produce a synthesized aperture D = 20 km would be useful for brightness temperature maps at 10 Mhz and below, with resolution of about 5 arcminutes at 10 MHz and 50 arcminutes at 1 MHz. This 10 MHz performance is similar to the Clark Lake TPT at 100 MHz and the 1 MHz performance (neglecting probable scattering limitations) is similar to that of the original 80 MHz Mills' Cross.

In summary, the 100-element VLFA would be very useful for source spectra and for extending brightness temperature maps to low frequencies, but could produce arcsecond class maps of only the brighter sources in the sky. Our ultimate goal should be centered on arrays of a thousand or more elements.

3. Further investigations

A number of areas require further investigation before serious design work can be undertaken, while many other (perhaps most) problems are probably best uncovered and addressed by undertaking a serious (if preliminary) design study.

i. Interplanetary and Insterstellar Scattering

Performance of the array at low frequencies will definitely be limited by interplanetary and interstellar scattering. Although a substantial amount of work has been carried out in both fields, which has formed the basis of extraplolations used above, much more can and should be done. In the final analysis, however, we must be prepared to find our predictions imperfect, and should arrange the deployment of early stages of the array to permit verification of predictions.

ii. Effects of the lunar ionosphere

Investigation of this problem is the most urgent of all issues: half of the case for a lunar location is based on the presumption that the ionosphere will be at least an order of magnitude less distorting than that of the earth (the other half is RFI shielding afforded by the lunar farside). What is the density and scale height and variability of the lunar ionosphere? What are the correlation scales? We need to direct the attention of ionospheric physicists to this problem immediately, and plan on experiments to verify predictions in early stages of array deploytment.

iii. Check basic assumptions

As noted above, we have been assuming many things are possible: active calibration of the elements and determination of their exact location through this process; individual communication possible without totally saturating link bandwidths; total mass per element small enough to contemplate many elements, and so on. Particularly noteworthy and worthy of careful examination is the short-dipole efficiency, and its bandwidth and stability if it needs to be matched. Although the possibility of higher gain elements was mentioned, it needs to be more carefully examined since such a direct impact on system performance would be produced by e.g. using local clusters of short dipoles as elements, rather than single units.

iv. Do a serious design study

Accept whatever uncertainties exist, and carry out a serious and complete design study, with the input assumptions carefully listed. This process will be more effective than any other method in uncovering problems, areas where tradeoffs are beneficial, and just what the ultimate practical limitations on the array may be. The danger is that such an early design might become embedded in administrative concrete; this danger could be minimized by funding more than one group to carry out the study.

III. Deployment of the Lunar Farside VLFA

This ambitious project will grow over a period of time, and the problem of setting forth a sensible sequence of stages must be addressed. Questions of scientific justification and administrative timing are of great importance and must be considered together with the purely technical ones of design checks and array performance. The following is a skeleton outline, intended to stimulate further thought on the matter.

1. Stage I -- the return to the moon .

i. Lunar orbiter

A lunar orbiter with 0.1 - 30 MHz receivers is deployed, which is also capable of transmitting on a variety of frequencies in this range on command. The receivers would be used for passivie occultation studies of solar system sources, including deliberately generated terrertrial signals; the transmitters would be used as probes for low-frequency occultation studies of the lunar ionosphere, as well as later for calibration signals for VLFA elements on the lunar surface.

ii. Early landers

A one-element system for solar system studies is deployed, capable of swept frequency and high time resolution. Several elements are deployed at a spacing of e.g. 20,200 and 2000 km, to be used interferometricallly to check on limitations posed by the lunar ionosphere and by interplanetary and interstellar scattering and to verify the effectiveness of our calibration procedures.. Although not very sensitive, even this system has the capability of detecting astronomical surprises.

iii. Tests of robot deployment

As a test of robot deployement, the beginnings of a cluster of antennas - perhaps 10 elements --are deployed over a region of 5-10 km. The elements should be linked to the central processing site by the system ultimately to be used; a data relay satellite should be in place and continuous processing begun. This is actually a useful if small system.

2. Stage II - the hundred-element array

i. Robot deployment of the 100 element VLFA

This array would be deployed over a region of about 20 km, and should yield the performance discussed as a benchmark above. Many astronomical problems can be addressed at the two to fifty arcminute resolution of this system.

ii. High-resolution interferometry

The 20-km cluster of 100 elements will be used against one or more outlying elements hundreds and thousands of km away, producing a synthesized high-sensitivity interfermeter. In addition to providing information necessary to the ultimate expansion of the system, preliminary information on the decameter sky at this resolution is obtained.

3. Stage III -- the arcsecond mapping instrument

i. Modified element design

Earlier stages will have suggested modifications in element design, which can be incorporated at this stage. In particular, to reach the 1000-element performance level needed, the competing possibilities of 100 ten-element clusters soft-landed and robot deployed or thousands of individual elements which are hard-landed must be decided.

ii. Deployment and incorporation of elements

Deployment would presumably occur over a period of years, with new elements brought on line as they are available; the system performance would gradually grow -- and could be biassed in directions of new targets of opportunity discovered at earlier stages of the array.

IV Summary

The lunar farside VLFA appeals to the observational radio astronomer as an achievable instrument of great potential. It will never be deployed unless a focussed and convincing argument for its utility can be made. These remarks do not constitute such an argument but hopefully we will in due course find or assemble a compelling case. It is important to include as primary partners in this endeavor those solar and space physicists whose interest in the solar system non-thermal radiation processes can form part of the bedrock on which the structure is erected.

SUMMARY AND CONCLUSIONS

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The workshop concluded with a general consensus on the scientific goals and preliminary design for a lunar far-side very low frequency array. Our major conclusions and recommendations are as follows:

(1) The lunar far-side is the only viable location within the inner solar system from which to conduct sensitive, very low frequency (0.5 to 30 MHz) astronomical observations. Ground-based observations are severely limited by a generally opaque ionosphere, electrical discharges in thunderstorm activity, and man-made interference. Earth-orbit observations are similarly constrained by terrestrial atmospherics, leakage of radio and television transmissions through the ionosphere, and auroral kilometric radiation produced by plasma processes in the Earth's magnetotail. A far-side VLFA will be shielded from these interfering signals by the Moon.

(2) A lunar VLFA is technically feasible. The dipole and receiver components are simple, off-the-shelf technology. The data at the anticipated rates can be processed with current specialized correlators, such as those present at the Very Large Array radio telescope, and computers. Data transmission back to Earth can be accomplished with a communications satellite in lunar orbit. Although the thermal and cosmic ray environments are harsh on the Moon, they do not present any substantial engineering or maintenance problems for the array. The major new technology that must be developed for this observatory is an automated robot for remote deployment of the dipoles.

(3) Unlike radio telescopes operating at higher frequencies, the system temperature (i.e. noise characteristics) will not be limited by receiver or internal electronics noise, but by the brightness of the sky. The astronomical signals that we anticipate will be strong. Therefore, impedance matching of the dipoles to the system electronics is not a crucial issue. So, short dipoles of about 1 meter length (much smaller than a wavelength) will be adequate. This will greatly simplify deployment and reduce the mass of the array.

(4) The beam size and shape, and the directivity for a short dipole are all relatively poor. To improve upon these characteristics, we propose to group the dipole antennas into mini-arrays consisting of two by two elements. The elements within a group could be phased to produce a small amount of beam shaping and improve the pointing.

(5) The initial, phase I array would consist of roughly several hundred dipoles spread over a circle with diameter 17 km. Since interstellar scintillation limits the resolution at 1 MHz to about 1^{o}, there is no reason to build an array with longer baselines at this frequency. This would also produce several arcminute resolution at 30 MHz. For the higher frequencies, one would like to extend the baselines to diameters of at least 1000 km in phase II. The phase I

array could be deployed in a spiral-like pattern with a high density of antennas near the center and lower density near the periphery. We suggest antenna spacings that increase with radius roughly as a power-law (i.e., r^n) where the index of the power-law (n) would be determined by computer simulations. In analogy to the VLA, we expect this pattern to produce good instantaneous u-v coverage (i.e., a good synthetic aperture). The emplacement of individual dipoles is not at all crucial with variations of tens of meters around the nominal pre-selected positions possible due to terrain considerations.

(6) The preferred mode of operation for the array is aperture synthesis. This will produce the best beams with lowest sidelobes and the least complications in communication between elements. The post-processing of data in this mode is more complicated than in a scanning mode but no more so than the current VLA. Operationally, aperture synthesis is simpler and more reliable.

(7) The scientific motivation for a lunar VLFA is potentially very strong. One could map the propagation of electron streams through the corona of the Sun produced by solar flare activity. Magnetospheric plasma processes near Mercury, Jupiter, and Saturn tend to produce low frequency radiation that can be monitored and analyzed with the VLFA. The galactic thermal and nonthermal backgrounds can be mapped to study the properties of the interstellar medium, and the origin and propagation of cosmic rays. Measuring the low frequency spectrum of extragalactic sources would be useful in understanding the process by which radio emission is generated, and how relativistic particles are accelerated and evolve with time. One might also find evidence of coherent radiation processes in extragalactic sources that are common in solar system magnetoactive plasmas. These exciting scientific goals require further study to tailor their applications to the VLFA.

SUGGESTIONS FOR FUTURE WORK

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Not unexpectedly, the workshop generated many new questions about the lunar VLFA that must be addressed before any serious design studies can be undertaken. We divide these questions into two categories: Science and Engineering.

I. Further Science Investigations

(1) The general consensus of the workshop participants was that the scientific justification for low frequency radio astronomy in general is not well focused or detailed. There is a great deal of potentially exciting and important astronomy that might be conducted with the VLFA, but at present the scientific goals are too vague and general. In particular, more study must be undertaken of the potential observations of the planets and the Sun. Within the general topics, such as the interstellar medium and extragalactic radio sources, we must develop stronger quantitative, theoretical arguments for observations with the VLFA. Predictions with specific observational signatures would be useful. We suggest that theoreticians be asked to join the working group on the VLFA. The scientific goals must be well established before detailed design work can begin.

(2) Some effort must be made to "sell" the VLFA to the astronomical community once the scientific justification is better focused. We will likely be in competition with other branches of astronomy for funding. Thus, our arguments for the VLFA must be at least as intriguing as those for x-ray and infrared astronomy. Unlike these other fields, however, low frequency radio astronomy has yet to fly its first survey instrument. We are effectively in the same position as x-ray astronomy in the early 1970's when the UHURU satellite was launched. These other fields have had an opportunity to discover many new exciting sources of radiation and, thus, mature. Low frequency radio astronomy is still in its infancy. Serendipity will play a large role in a lunar VLFA since we are not really sure what the sky looks like in this uncharted window. This puts us at a distinct disadvantage with respect to other, more developed areas of astronomy.

(3) Some very useful survey work at low frequencies could be performed on precursor missions that might be launched in anticipation of the establishment of a permanent lunar base. A simple dipole or V-shaped antenna could be placed on board a lunar geoscience orbiter. This would be the successor of the RAE satellite. This instrument could survey the sky at low resolution and low background levels when the spacecraft orbit takes it above the lunar far-side. Similarly, if robotic spacecraft were landed on the far-side for geological survey work, dipole antennas could operate from these vehicles. If several were landed, simple interferometry at low frequencies could be performed to survey and locate radio sources at higher resolution. Such precursor missions would help to address the scientific questions described in (1) and (2) above.

(4) More theoretical study is needed on the limitations that will be placed on the lunar VLFA by scattering and refraction from the interplanetary and interstellar media. These plasma effects ultimately limit the resolutions and frequencies of the VLFA. Therefore, the effects must be well understood so that they can be incorporated into the design of the VLFA.

(5) A better investigation of the ionosphere near the surface of the Moon is critically needed. At present, good global and near-surface values of the electron density do not exist. These are crucial to the design of the array. We must know, for example, if a 10^4 electron/cm^3 layer hugs the day side of the Moon as recently suggested by Vondrak (1988). These investigations could be carried out by a combination of remote, ground-based observations, reanalysis of RAE and Apollo data, and new measurements made from the lunar surface by precursor and/or early lunar base missions.

(6) In an effort to better understand the scientific limits of the VLFA, we suggest that some initial computer modeling be undertaken to investigate the type of radio source structure that one might be able to map with the VLFA. For example, one could "observe" an extragalactic radio source (particularly, the expected low-frequency, extended components) via the computer as it would be seen by the VLFA. Beginning with a model, we can convolve the hypothetical source with the response of the array (i.e. u-v coverage) and add noise at the appropriate level to examine what structures one might expect to observe. This will constrain both the science and the design.

(7) We propose that consideration should be given to precursor missions before the establishment of a lunar VLFA. These missions would be used to accurately determine the density and extent of the lunar ionosphere, to perform a preliminary low-resolution all-sky survey at very low frequencies, to study the signal propagation effects through the interstellar medium, and to begin monitoring low frequency variable sources. These missions could be conducted from lunar orbit and from the surface as part of a survey of the lunar far-side.

II. Engineering Studies

(1) There should be continued trade studies of various options for the array elements: long versus short dipoles, dipoles versus tripoles, dipoles versus Beverage antennas (J. Kierein, private communication).

(2) The frequency range must be further investigated. One is limited to frequencies above about 1 MHz for galactic and extragalactic observations. However, some of the more interesting planetary emissions occurs between 100 kHz and 1 MHz. What should be the lower bound on the frequency of the array?

(3) Computer studies should be undertaken to design the optimum number and configuration of dipoles. Studies of the instantaneous and

longer term u-v coverage (i.e., synthetic aperture) can be performed with currently available software and computers.

(4) More thought must be given to problems involving noisy data that are anticipated from the VLFA. Questions of self-calibration, phase retrieval, refractive errors, and three-dimensional Fourier transforms must be considered.

(5) Consideration of observing strategy would be useful at this early phase. How does one maximize the range of objects to be studied in detail and minimize the observing time with the proposed VLFA configuration?

(6) One should study the design of a VLSI chip that could perform most of the receiver functions needed at each dipole. This would simplify the electronics, reduce the weight requirements, and potentially make the antennas more reliable.

(7) The power requirements and power storage are still uncertain. We expect that each element will need less than a watt of continuous power during operation. However, how will this power be supplied, especially during the 14-day nights? Will solar cells with battery storage be adequate? How will this impact on the expected limitations on the mass of the array and the deployment?

(8) An intelligent, robotic vehicle must be designed to remotely deploy the antennas. Given the delay time between the far-side and Earth and between the far and near sides, the robot will most likely have to operate in a semi-autonomous mode. Thus, questions of telepresence and artificial intelligence enter into the design. We see this as the most difficult engineering hurtle for the VLFA.

(9) We suggest that a ground-based model of a mini-VLFA be built to test observing and deployment strategies, and to debug the electronics and communications for remote operation. This model would operate at 30 MHz, a window that is accessible (but often polluted by interference) from the ground. The array would be relatively inexpensive since it can be built from currently available components. We would plan to simply deploy these antennas, as on the Moon, with each dipole having an electronics and communication package. The data would be transmitted to a central location via a radio link and recorded on magnetic tape. We may be able to use currently available correlators (e.g., the VLBA correlator) to produce source fringes and visibilities that can be made into sky maps. This model would test the feasibility of the design and operational mode for a future lunar far-side observatory.

Summary of Work on Assembly of Phobos Mission Spacecraft in Low Earth Orbit

Prepared for the NASA Johnson Space Center
by Eagle Engineering under Subcontract
to Lockheed Engineering and Management
Services Company

NASA Contract No. NAS 17900
Eagle Engineering Report No. 88-198

September 30, 1988

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4.3.1.2 ASSEMBLY STRATEGY IN LEO

OBJECTIVE

The objective of this study was to evaluate the assembly of the Phobos mission vehicle in LEO. On-orbit mating of multiple elements of a vehicle becomes more and more practical as the size of the pieces being lifted to LEO increases and the number decreases.

METHODOLOGY

Background

A variety of previous studies of transportation node Space Stations have concentrated on the problems of assembling, refurbishing, and maintaining fully or partially reusable transportation systems for trans-lunar or trans-Martian manned flight. This previous work has concentrated on long-term scenarios that assume a substantial human presence in LEO. On-orbit assembly and test of spacecraft, as well as cryogenic propellant transfer and storage are generally assumed. Recently proposed piloted Phobos and Mars missions have assumed this capability.

On the other hand, the infrastructure and many of the technologies needed to assemble, test, and launch large spacecraft from LEO do not exist at present and represent an obstacle for proposed missions carrying humans to Mars with Earth departure dates on or around the year 2000. The Space Station program, as currently designed and projected, cannot assemble a manned Mars stack which will mass more than 1,000 metric tons.

Studies of the assembly of Mars stacks have usually called for a separate facility, either co-orbiting or totally separate from the currently proposed Space Station. A Mars stack will greatly disturb the micro-g environment, contaminate the vacuum, and in general require the pressurized volume, crew time, power, etc., currently planned for other uses in the present program.

The various looks at the problem indicate that, unless the current Space Station program changes emphasis, an additional Mars stack assembly space station will be required. Given that the currently proposed phase 1 Station will not be in place and operational until the mid to late 1990s, construction of an additional large facility for assembly of Mars stacks seems unlikely before 2000, making a Mars mission desiring to depart in that time frame and requiring such facilities not possible.

One solution to the problem is to assemble the Mars stack without a space station or other space based infrastructure of significance. The Phobos vehicle concepts discussed in general assume no space station will be available.

Key Assumptions

1. The Martin Phobos configuration (Ref. 1) was used for assessment.

2. The ETO capability assumed for delivery of the Phobos vehicles (cargo and lander) was an ALS class (90 mt) launcher.

APPROACH

The approach employed for this activity was to evaluate the on-orbit assembly concept of the Martin Phobos vehicle configuration, to identify issues and/or areas of concern, and to recommend alternatives to be considered for incorporation into the FY89 Expedition to Phobos case study requirements (Ref. 2).

FINDINGS

Figure 1 shows the Martin Phobos vehicle to be assembled. Numerous large fluid connections are required. Eleven large tanks come together to make up the TMI single stage. A summary of selected key assembly requirements for this vehicle is shown in Table 1. EVA was estimated by determining each task to be performed for each tank or stage brought up. The total number arrived at was then multiplied by a 1.5 factor as a margin for things forgotten or unknown. The true value is highly influenced by the level of technology used in assembly and is difficult to accurately estimate at this level of detail. The objective at this time was to develop a reference and method to enable comparisons at an order of magnitude level of accuracy rather than to estimate exact values. Better definition of the hardware interfaces will be required to improve the EVA estimates. Based on initial assessment of the explicit or inferred requirements, assembly of the reference Phobos vehicle without the use of some facility in space is, at best, questionable and cause for concern; at worst, assembly of this vehicle as proposed is not possible.

The shuttle remote manipulator is only rated for a third of the 90 m ton tank mass that must be moved around. In addition, the shuttle must dock at a variety of locations on the vehicle in order to use the manipulator to place the tanks, which though possible, seems impractical. Twenty-five months are required to assemble the cargo and piloted vehicles using an estimated 187 EVAs of six hour duration each or approximately two EVAs per week for each week of the twenty-five month process. The orbiter fleet cannot support this length of stay or number of EVAs. It may be possible to use the habitation module for the piloted vehicle to support the EVAs however, with some weight penalty on the whole mission. The orbiter and RMS must be on hand to place each of the 24 payloads, however. Additionally, it is not viewed as being practical to fly a 90 m ton tank into a slot between other tanks and position it with sufficient accuracy to make up eight fluid connections, four structural, and at least one electrical connection. The tank must be positioned with a manipulator or other device rigidly connected to the vehicle. Thus it appears that a space facility will be necessary if assembly is to be constrained to use of current support system capabilities and concepts.

The preceding concerns led to the concept for assembly shown in Figure 3 of Appendix A with a remote manipulator (RMS) capable of reaching any point on the vehicle requiring placement of a tank. The Figure 3 concept shows a manipulator and truss structure only. Power, thermal control, attitude control, habitation, and EVA/airlock/ spacesuit support are all assumed to come from another source, most likely a space station rigidly attached to the structure.

The assembled (Figure 1) vehicle is essentially put together piece by piece. An ALS class launcher capable of placing the 90 m ton tanks in LEO is assumed. The RMS travels on a strongback. The vehicle is assumed to be docked to some rotating fixture that will allow the single RMS on a strongback access to it all. EVA or a capable robotic equivalent is required for numerous fluid, electrical, and structural interconnects. The biggest challenge is the 128 fluid connections that must be made in space, including 34 large line (20 inch or so) connections. Bolted connections and leak tests were assumed to be required for these large lines. Quick connects similar to the shuttle/external tank interface may also be possible.

Figure 2 shows the launch schedules for assembly of the Phobos vehicle and associated shuttle support launches. Shuttle launches are required to replace assembly crews every six months or so. A minimum launch rate of one ALS class stage per month is required to assemble the piloted vehicle between the time the cargo vehicle departs and the piloted vehicle departure date, roughly 18 months. Shuttle launches concurrent with the ALS launches are also required.

During the course of this study and evaluation, the question arose as to whether assembly can be simplified and what options may be available for consideration. Conceptually, it was judged that docking stages together without fluid connections might be a viable candidate configuration option to be levied as a study requirements on the vehicle. To confirm the potential merit of this idea, the Phobos vehicle TMI stage was scaled parametrically in order to estimate the effect of multiple stages on the mass of such a configuration as well as estimates of the effect on the other assembly parameters (Tables 2 and 3). It is recommended that ETO capabilities up to 230 mt, a Phobos vehicle configuration that includes both the addition of an aerobrake and multiple TMI stage options be study requirements for FY89.

For more detail about the Martin vehicle analysis refer to Appendix A, and to Appendix B for the docked vehicle. The Appendices contain information about the manifest, EVA/IVA requirements for on-orbit assembly and other related issues.

ISSUES

The following issues that need to be considered for assembly surfaced in this analysis:

1. How will the ALS payload and upper stage be brought to the assembly point? Should they fly themselves up to a docking structure or should an orbital maneuvering type device dock with them and bring them to the assembly point?

2. What is the optimum altitude assembly orbit? Are there requirements on the departure inclination? Is the phase 1 Space Station orbit adequate?

3. What is the micrometeoroid/orbital debris shielding requirement for these vehicles? Are multi-wall shields required on the tankage?

4. What are the penalties associated with boil-off? Can it simply be vented or should it be captured and used for attitude control and orbit makeup or should it be reliquified and placed back in the tank? Is the capability to top off the tanks required?

5. How can the vehicles accommodate other launch opportunities with different delta V requirements?

6. Current launch vehicles (manned and unmanned) have an ascent success rate of roughly 91 percent over 447 flights (includes all major U.S. and foreign launchers). This is simple ascent reliability. On-time performance is much worse - not even measured. Given this one in ten failure rate and poor schedule performance history, should a vehicle requiring multiple launches prefer a few large launches or many small ones? How can we make the system insensitive to a launch vehicle failure?

7. What is the maximum time a crew can work on-orbit, supporting two EVAs per week or more?

8. Can the assembled vehicle with over 100 fluid connections be launched from LEO without a hot fire engine test for the TMI stage?

9. Can the assembled vehicle be adequately vibration tested on the ground? Is a vibration test in space required?

10. Is it possible to build a large diameter (20 inch) quick connect for cryogenic fluids that requires no leak testing with cryogens in it?

11. How can low level leak testing be done in space? Gas sniffers will not work in a vacuum. Is it practical to run a spectrometer device over the surface of all the plumbing or each connection? Are leaks in the TMI stage plumbing really important in space?

12. What is the level of complexity of the plumbing for the TMI stage made up of twenty-two tanks? What is the minimum number of fluid connects required per cryogenic tank?

13. Is there a docking concept that would allow the assembled vehicle to be put together without a manipulator to move the tanks around?

14. What mass of facilities and consumables are required to support two EVAs per week over a two year period?

15. When is it reasonable to establish assembly parameter allocations on the transportation and node allocations and is there a path by which to converge to reasonable allocations?

REFERENCES

1. Human Expedition to Phobos, Case Study 1, Transportation, Viewgraph Package, Martin Marietta, CS-1.MMSS-1, 7/10/88.

2. Assembly of Phobos Mission Spacecraft in Low Earth Orbit, W. Stump et. al., Eagle Engineering/LEMSCO, Contract No. NAS 17900, Eagle Report No 88-198, August 22, 1988.

Figure 1 Assembled (Martin) Vehicles, Cargo and Piloted

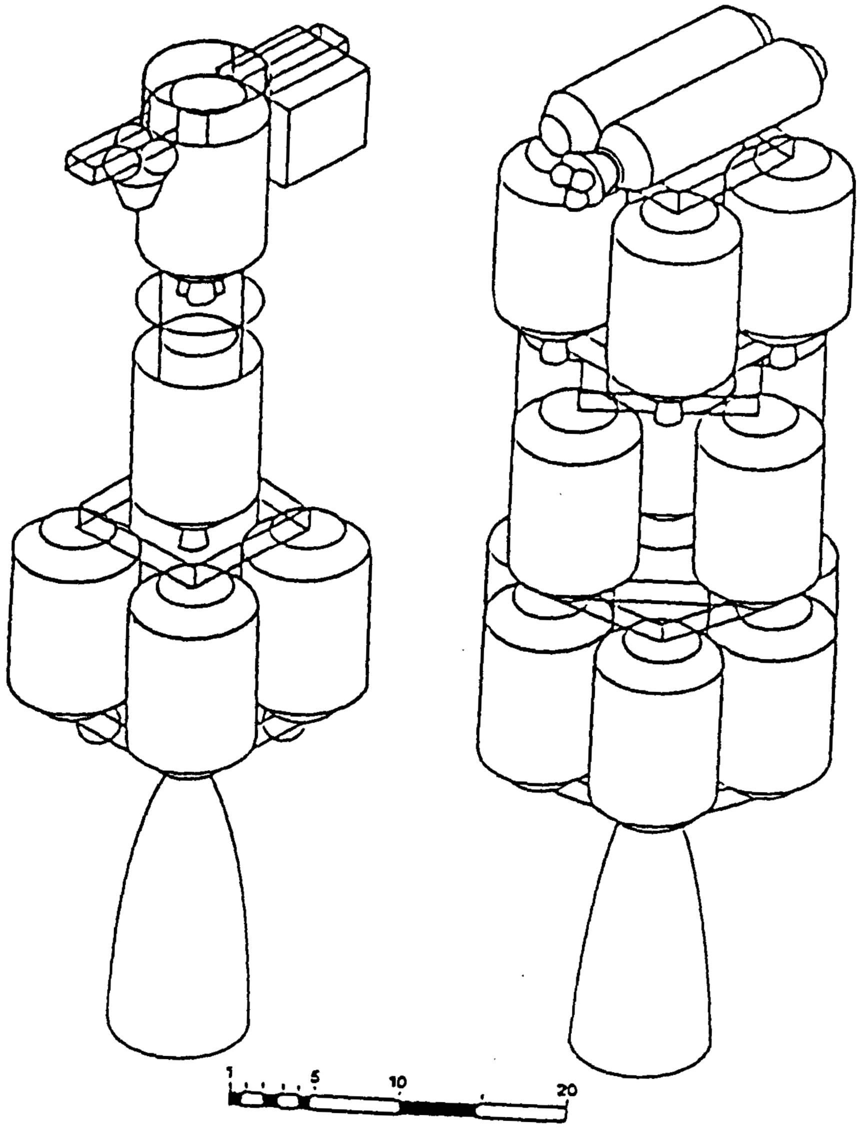


Table 1, Summary of Phobos Veh. Assembly Reqs., Martin Vehicle

Item	Assembled (Martin) Vehicle
Total LEO mass at dep., both veh., m tons	2,096
Cargo Veh. LEO Mass, m tons	467
Piloted Veh. LEO Mass,m tons *Becomes if MOO2 moved to cargo veh.	1,310* 1,180
No. of HLV launches req. for both veh.	24
No. of STS launches req. for both veh.	5
HLV max. payload to LEO required, m tons	96
HLV max. shroud dia. req., meters, (feet)	10 (33)
HLV max. shroud length req., meters, (feet)	30 (100)
EVA Req. for Cargo Veh. Assem., No. of 6 hour EVAs.	56
EVA Req. for Piloted Veh. Assem., No. of 6 hr EVAs	131
Total EVA Req., No. of 6 hr. EVAs	187
No. of req. large dia. (20 inch) fluid line connections req.	32
No. of req. small dia. fluid line connections req.	96
No. of req. structural in space connections req.	117
No of req. electrical in space connections req.	28

Figure 2. Schedule for Assembled (Martin) Vehicle

Launch Number	Launch Vehicle	Payload	2000										2001											2002								
			Mar	Apr	May	Jun	Jul	Aug	Sep	Oct	Nov	Dec	Jan	Feb	Mar	Apr	May	Jun	Jul	Aug	Sep	Oct	Nov	Dec	Jan	Feb	Mar	Apr	May	Jun	Jul	Aug
1	ALS-1	TEIS, Phev, Cargo				●																										
2	STS-1	ASSEMBLY CREW #1 (A.C. #1)				●																										
3	ALS-2	MOSC, MOOS					●																									
4	ALS-3	TMIS (tank #1)						●																								
5	ALS-4	TMIS (tank #2)							●																							
6	ALS-5	TMIS (tank #3)								●																						
7	ALS-6	TMIS (tank #4)									●																					
8	ALS-7	SSME+										●																				
9	STS-2	INSPEC. CREW #1 A.C #2												● ▨																		
10	ALS-8	HAB MODULES ECCV												●																		
11	ALS-9	MOCS (tank #1) MOO-1													●																	
12	ALS-10	MOCS (tank #2) MOO-1														●																
13	ALS-11	MOCS (tank #3) MOO-1															●															
14	ALS-12	MOCS (tank #4) MOO-1																●														

▨ Launch Cargo Mission from LEO

Figure 2. Schedule for Assembled (Martin) Vehicle, Continued

Launch Number	Launch Vehicle	Payload	2000										2001												2002							
			Mar	Apr	May	Jun	Jul	Aug	Sep	Oct	Nov	Dec	Jan	Feb	Mar	Apr	May	Jun	Jul	Aug	Sep	Oct	Nov	Dec	Jan	Feb	Mar	Apr	May	Jun	Jul	Aug
15	ALS-13	TMIS (tank #1)																	●													
16	STS-3	A.C. #3 A.C. #2 RETURNS																	●													
17	ALS-14	TMIS (tank #2)																		●												
18	ALS-15	TMIS (tank #3)																			●											
19	ALS-16	TMIS (tank #4)																				●										
20	ALS-17	TMIS (tank #5)																					●									
21	ALS-18	TMIS (tank #6)																						●								
22	ALS-19	TMIS (tank #7)																							●							
23	STS-4	A.C. #4 A.C. #3 RETURNS																							●							
24	ALS-20	TMIS (tank #8)																								●						
25	ALS-21	TMIS (tank #9)																									●					
26	ALS-22	TMIS (tank #10)																										●				
27	ALS-23	TMIS (tank #11)																											●			
28	ALS-24	SSME+																												●		
29	STS-5	FLIGHT CREW INS. CR. (on shut)																													●	▨

▨ Launch Human Mission from LEO

Table 2, TMI Stage Sizing for Staged/Docked Phobos Veh.

No. TMI Stages for piloted vehicle **	1	2	3	4	5
Piloted Vehicle:					
Total LEO mass m tons	1,180	959	925	867	865
TMI Stage wet * mass, m tons	877	352	223	156	124
Cargo Vehicle:					
No. of TMI stages for cargo veh.	1	2	2	2	3
Total LEO mass m tons	664	621	621	621	611
TMI Stage wet * mass, m tons	454	205	205	204	132

* All TMI stages are the same size, to reduce manufacturing costs and make maximum use of the launch vehicle. All TMI stages use a 10 meter maximum diameter.

** The single stage TMI vehicles assume the MOO2 burn propellant has been transferred to the cargo vehicle, as suggested by Martin, making the piloted vehicle 130 tons lighter than the Martin vehicle in reference 1.

Table 3, Summary of Phobos Veh. Assembly Reqs., Staged/Docked Veh.

(3 Stage TMI, 2 Stage MOC)

Item	Docked Vehicle
Total LEO mass at dep., both veh., m tons	1,546
Cargo Veh. LEO Mass, m tons	621
Piloted Veh. LEO Mass, m tons	925*
No. of HLV launches req. for both vehicles	8
No. of STS launches req. for both veh.	4
HLV max. payload to LEO required, m tons	226
HLV max shroud dia. req. meters, (feet)	10 (33)
HLV max shroud length req. meters, (feet)	44 (145)
EVA Req. for Cargo Veh. Assem. No. of 6 hour EVAs.	7
EVA Req. for Piloted Veh. Assem. No. of 6 hr EVAs.	10
Total EVA Req., No. of 6 hr. EVAs	17
No. of req. large dia. (20 inch) fluid line connections req.	0
No. of req. small dia. fluid line connections req.	0
No. of req. structural in space connections req.	36
No. of req. electrical in space connections req.	7

*MOO2 propel. on cargo veh.

Appendix A

Martin Phobos Vehicle

Appendix A

Figure 3 shows the assembly sequence for the piloted vehicle with manipulator and truss structure only. Power, thermal control, attitude control, habitation, and EVA/airlock/spacesuit support are all assumed to come from another source, most likely a space station rigidly attached to the structure. Table 4 shows the manifest for assembly and launch of the concept including only ALS cargos.

A rough estimation of the EVA and IVA required during the on-orbit assembly process for the Martin vehicle was generated in order to establish what kind of assembly infrastructure would be needed. Also, by performing a similar analysis on the docked concept, an idea of the EVA/IVA savings obtained by using the docked vehicle was estimated.

The technique used to evaluate the on-orbit assembly EVA/IVA required for each vehicle involved two steps. First, the vehicles were configured. The structural, electrical, and fluid interfaces between major elements of the vehicles were defined. All efforts to minimize the amount of EVA/IVA needed to assembly these elements were taken. The payload capacity of the heavy-lift launcher chosen for both vehicles was a major driver in this configuration process. The vehicle elements and their interfaces were defined with close attention to minimizing EVA/IVA and the heavy-lift vehicle's required lift capability. The elements were then manifested in as few launches as possible.

The second major step involved defining the EVA/IVA tasks required to assemble the cargo elements in the order dictated by the manifest. A list of tasks needed to assemble the cargo elements brought up on each subsequent heavy-lift launch, starting with the first, was produced. Then, each task was analyzed, broken down into subtasks if necessary, and assigned an EVA/IVA hour requirement. The different types of assembly tasks performed and their associated EVA/IVA hour costs are shown in Table 5. The list of tasks performed for each set of cargo elements in the manifest of both the Assembled and Docked configurations are shown in Tables 6 & 7 respectively. The latter set has been broken down by mission and includes each task and EVA/IVA cost, the total EVA/IVA hours and days required to complete the assembly imposed by each flight, and the total for each vehicle.

The list of tasks composed for each flight was based on conventional operations conducted on the ground and on engineering common-sense expectations (e.g., attitude control system test). No new technology was projected. For example, bolted connections were assumed to be required for large fluid lines. The list of tasks is far from complete and some of the tasks included are controversial (e.g., engine fire test for TMI stage). The amount of time allocated for each task may also be debateable (e.g., 10 hours of EVA to align, connect, and leak check a large diameter main propellant line). The total number arrived at was then multiplied by 1.5 as a contingency for things forgotten or unknown and problems that might occur. The "true value" is highly influenced by the type of technology used in assembly and is difficult to accurately estimate at this level of detail. To improve these estimates, the interface hardware must be defined in more detail.

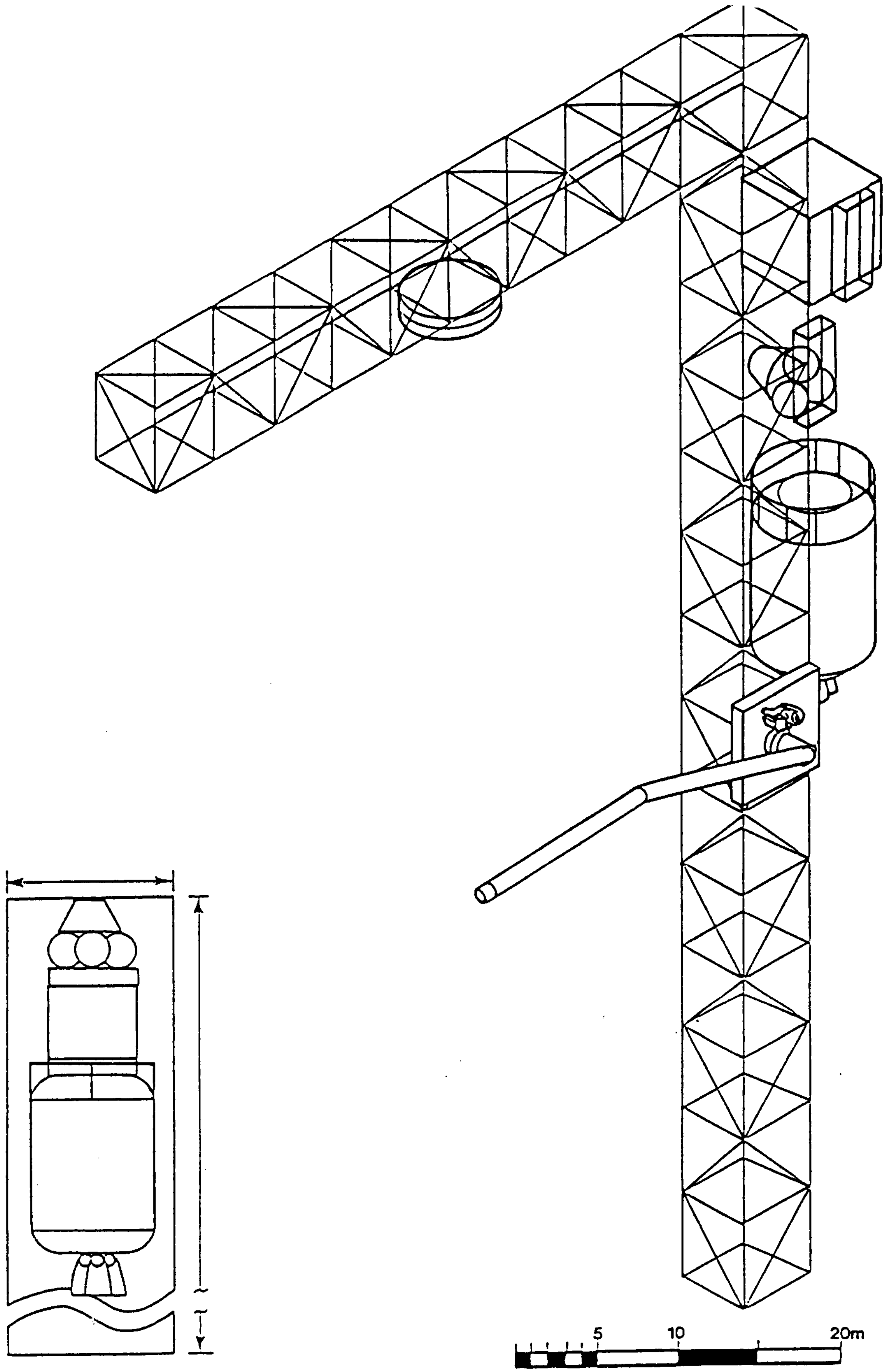
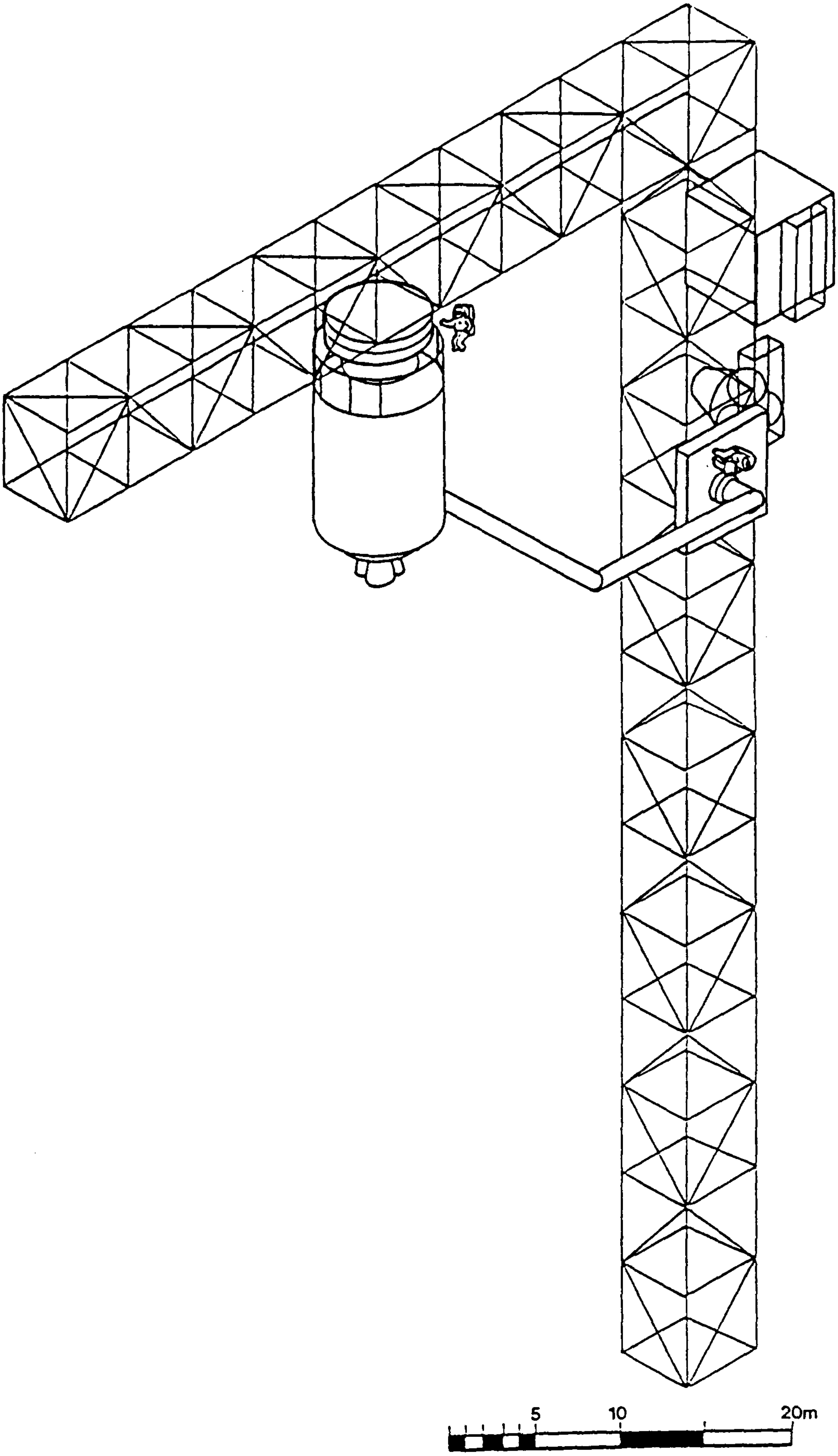
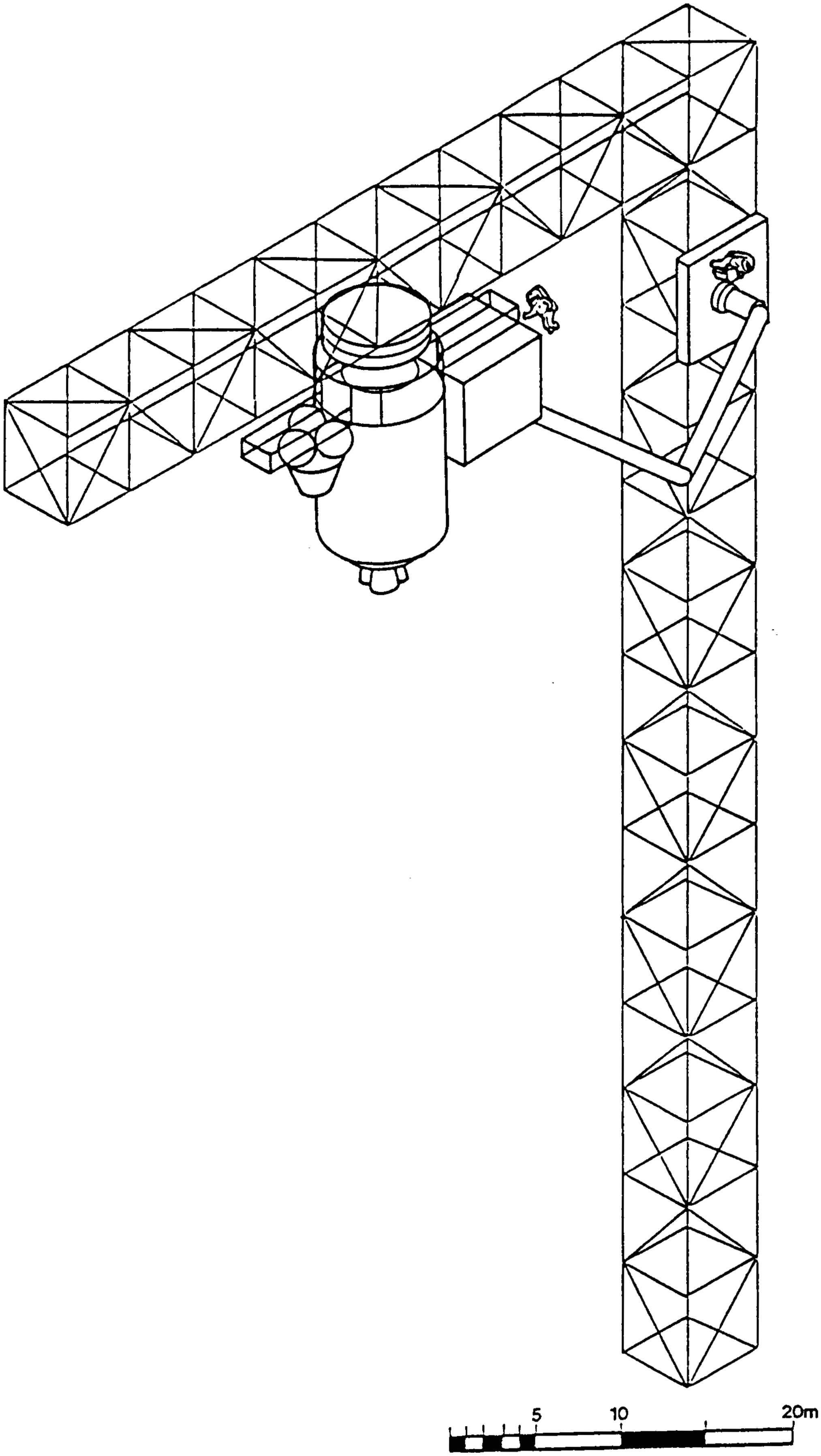


Figure 3.
Flight #1 (ALS-1)

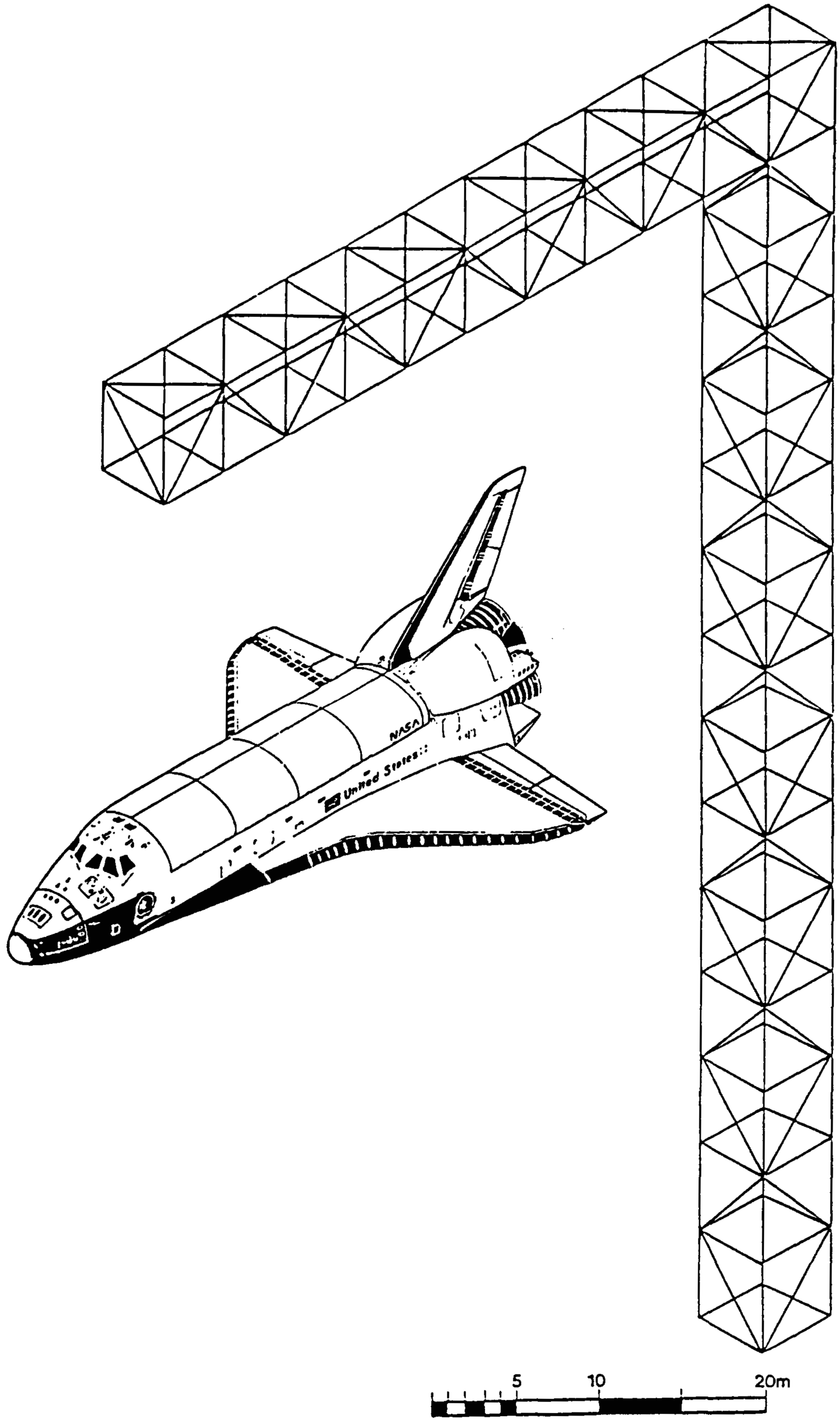
Payload elements temporarily docked to station assembly backbone



Over-sized RMS connects TEIS tank to assembly rotisserie

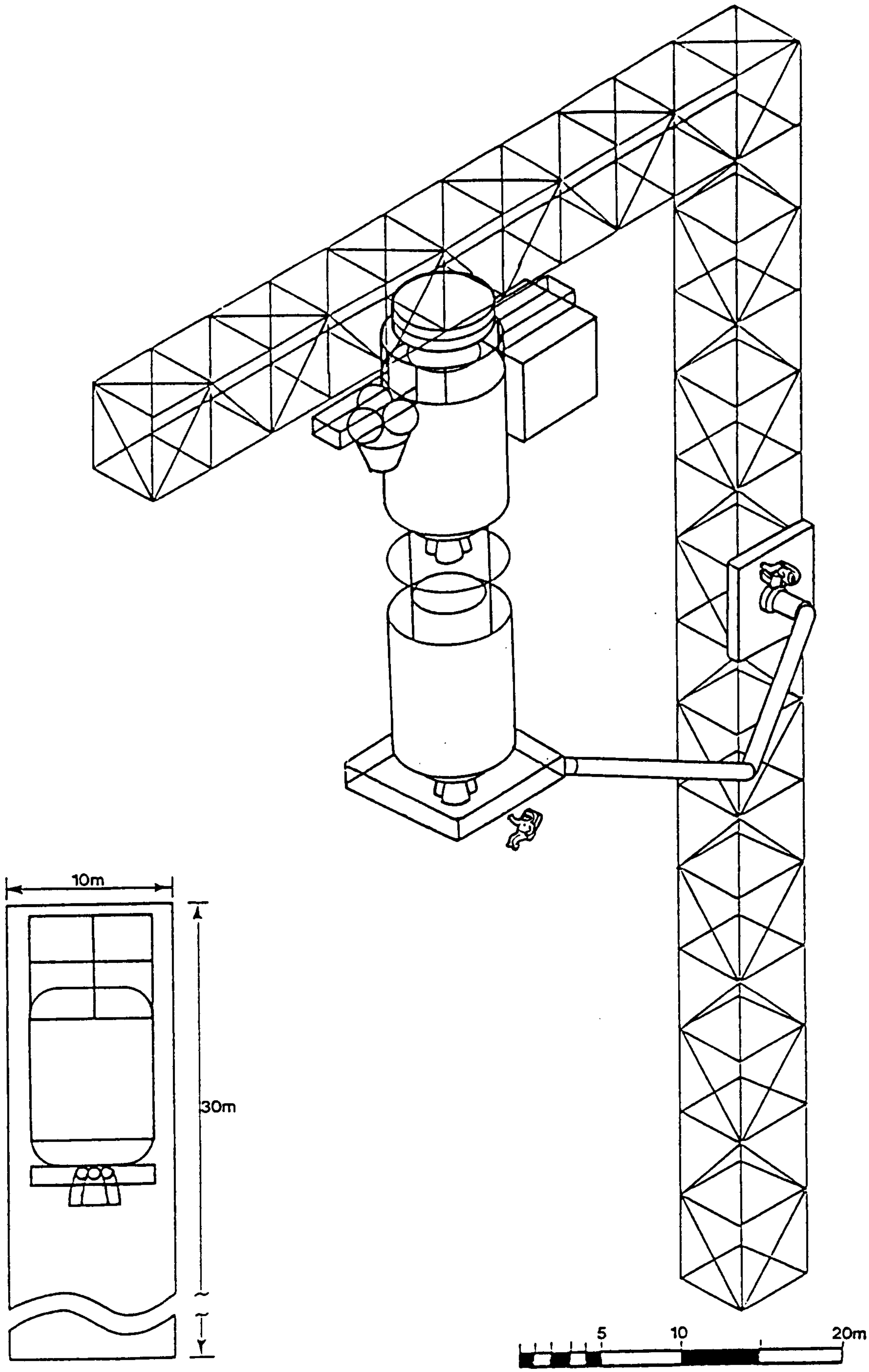


RMS attaches two truss members, Phev, and a cargo package to the TEIS tank



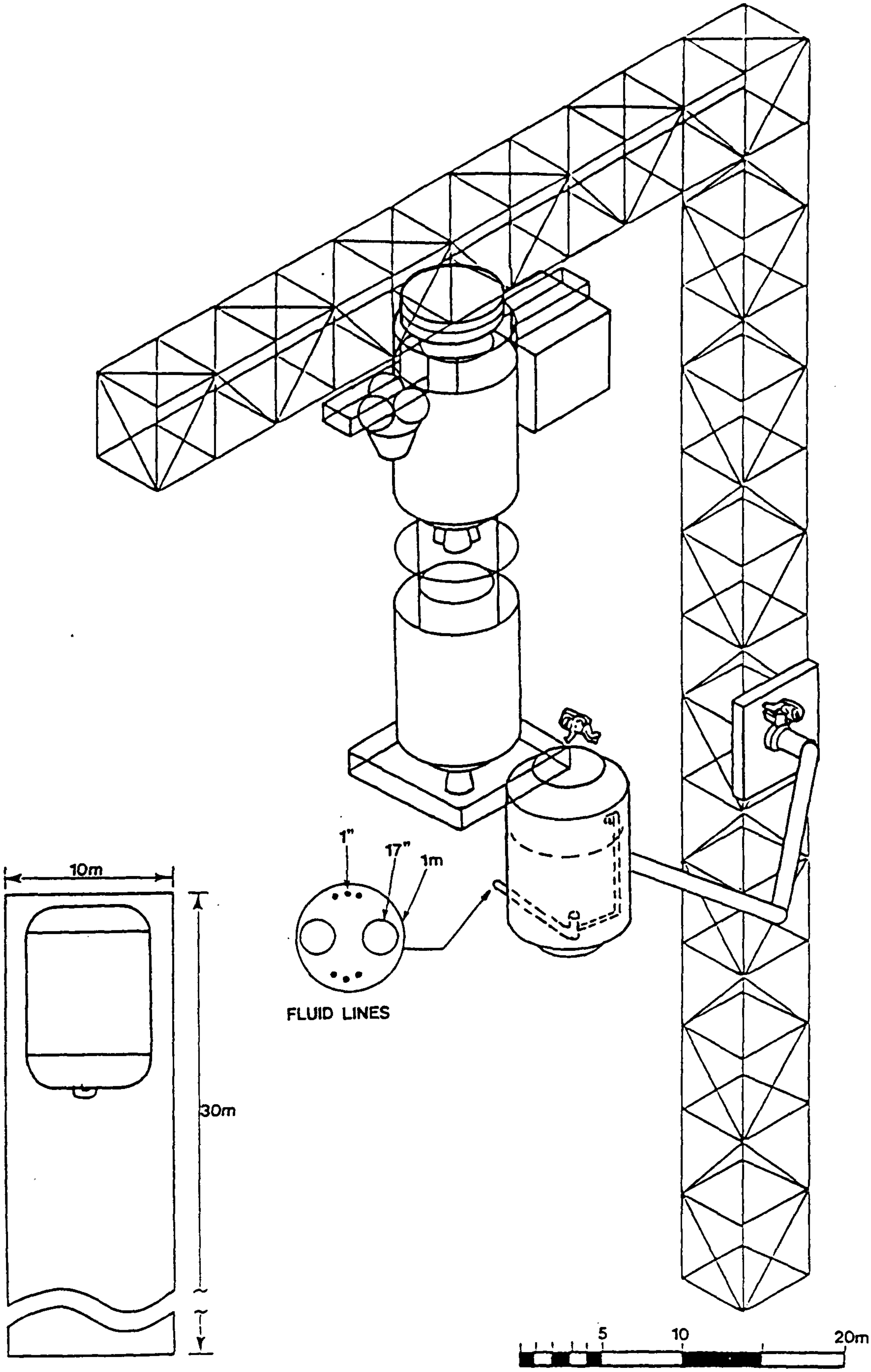
Flight #2 (STS-1)

Assembly Crew #1



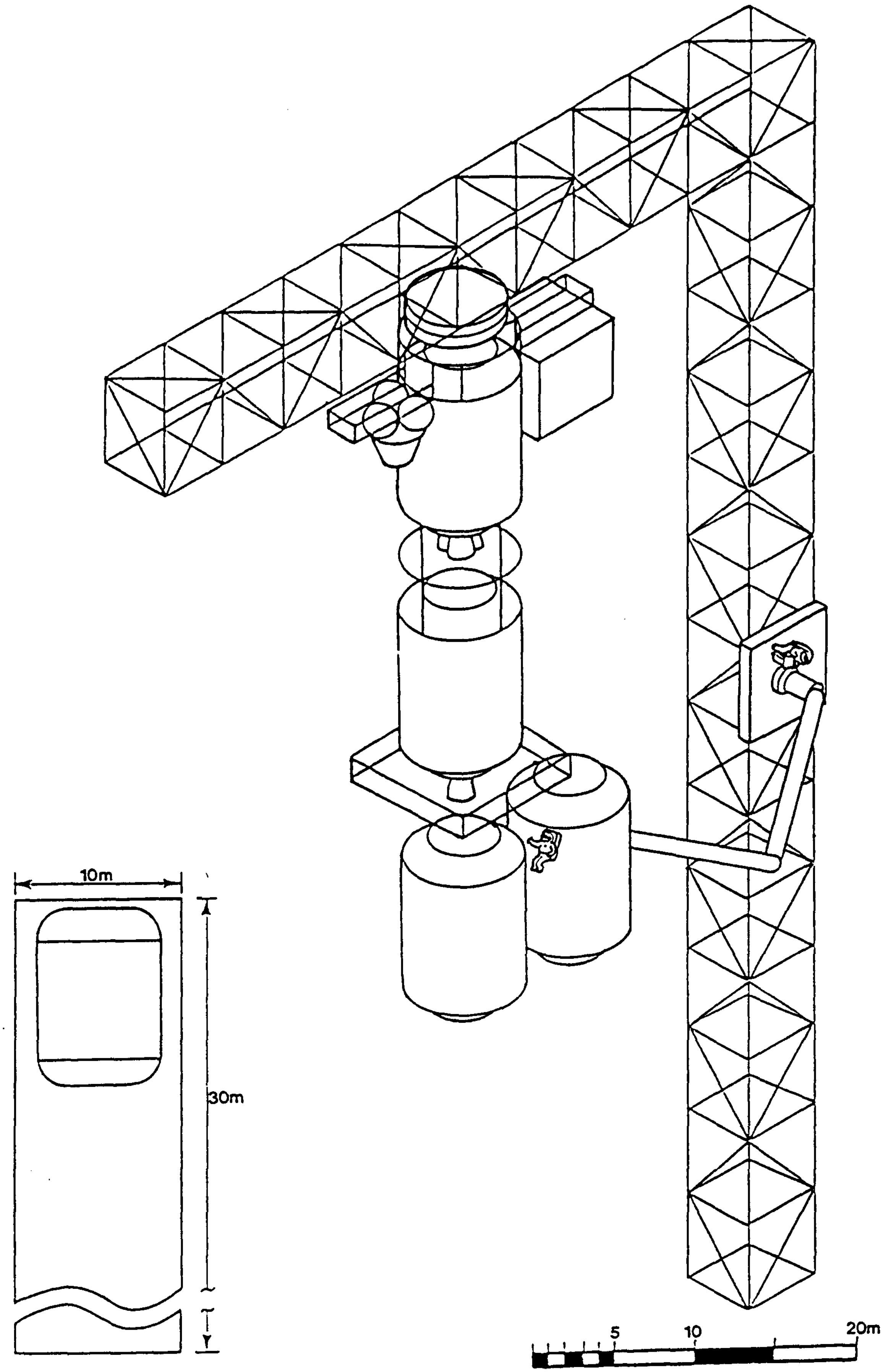
Flight #3 (ALS-2)

MOCS stage, with interstage and truss already in place, is positioned for connection to the TEIS tank



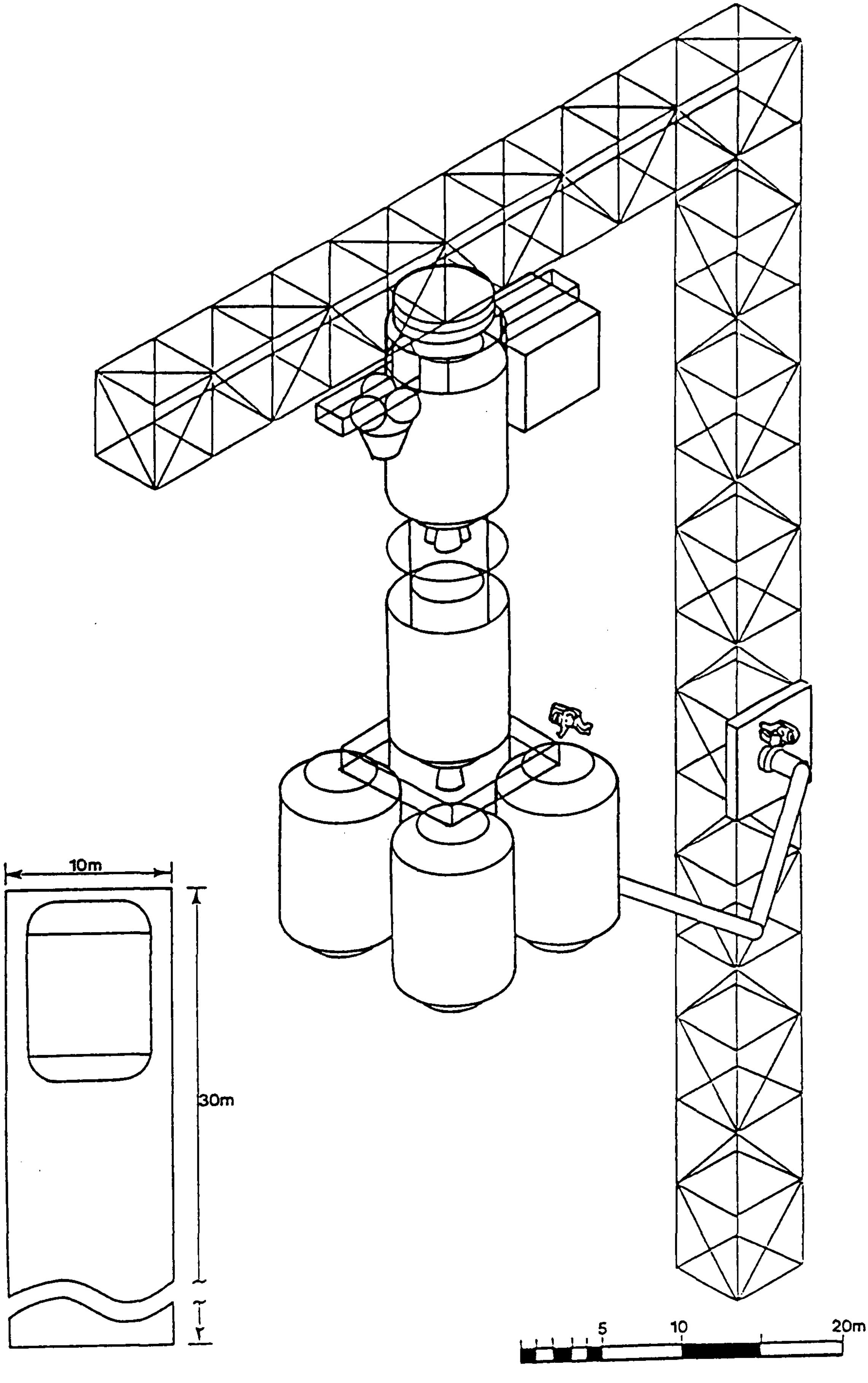
Flight #4 (ALS-3)

TMIS tank #1, with fluid line piping, is attached to square trusswork above



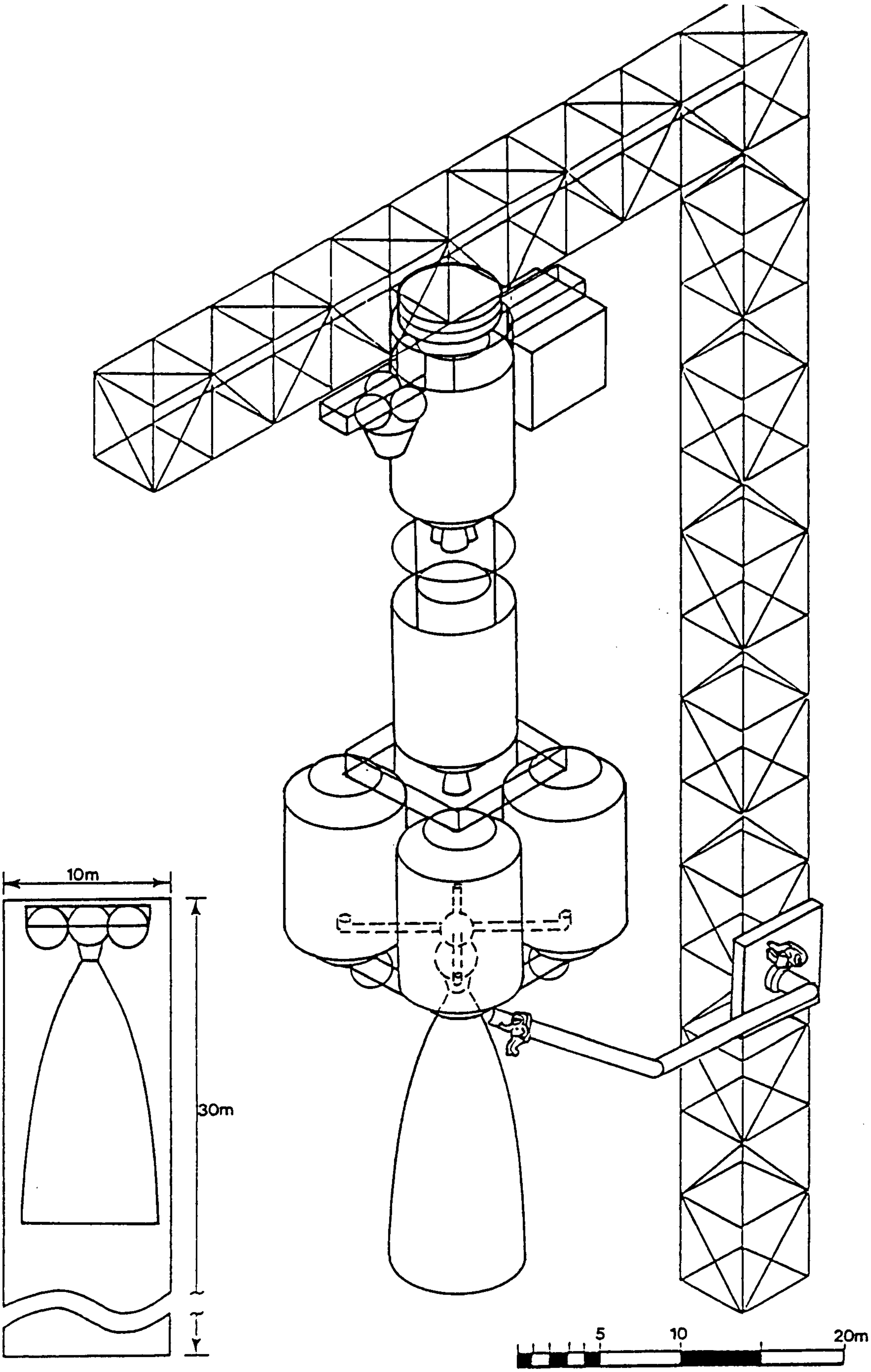
Flight #5 (ALS-4)

TMIS tank #2, with fluid line piping, is attached to square trusswork above



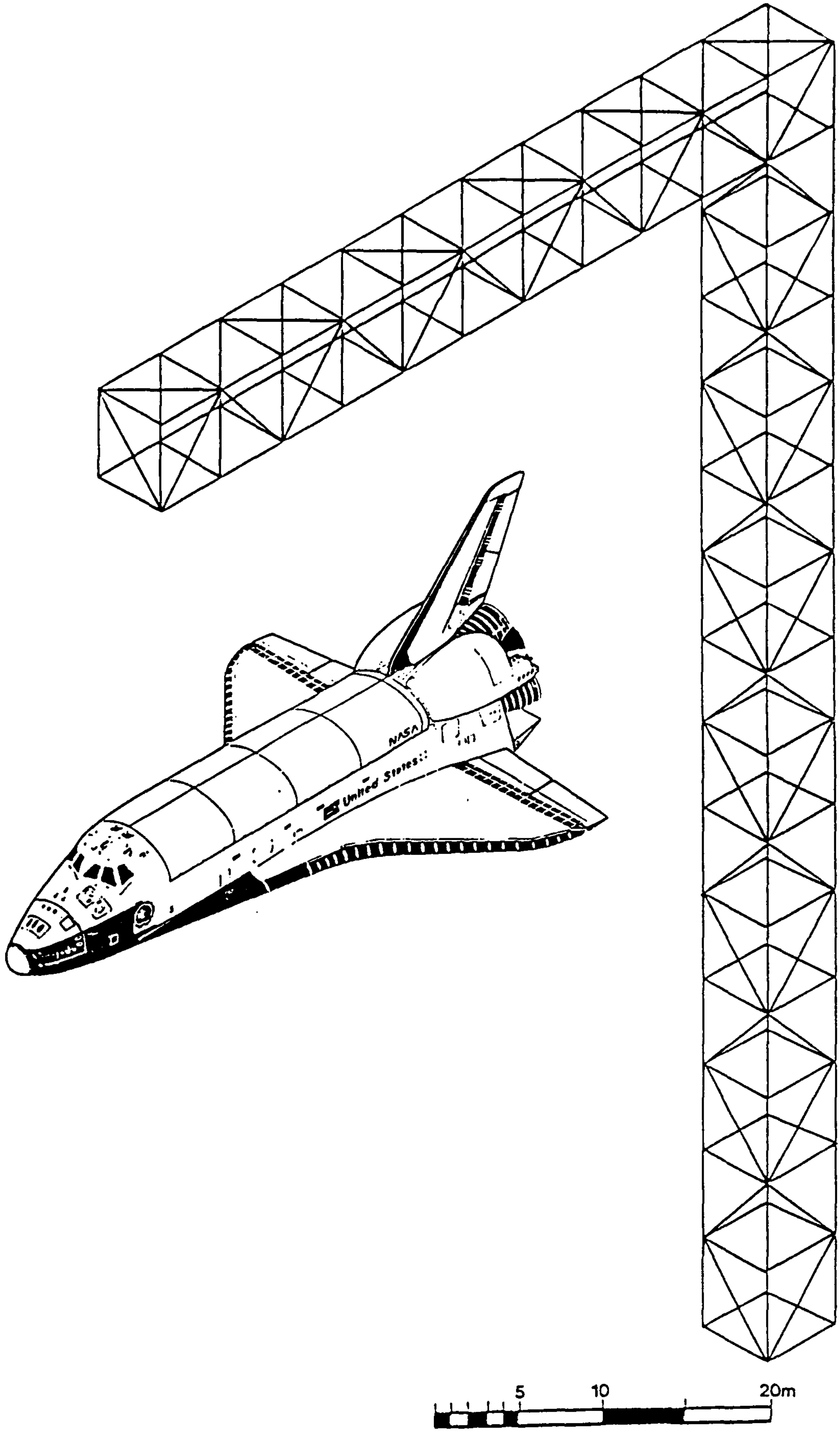
Flight #6, #7 (ALS-5, 6)

TMIS tank assembly completed



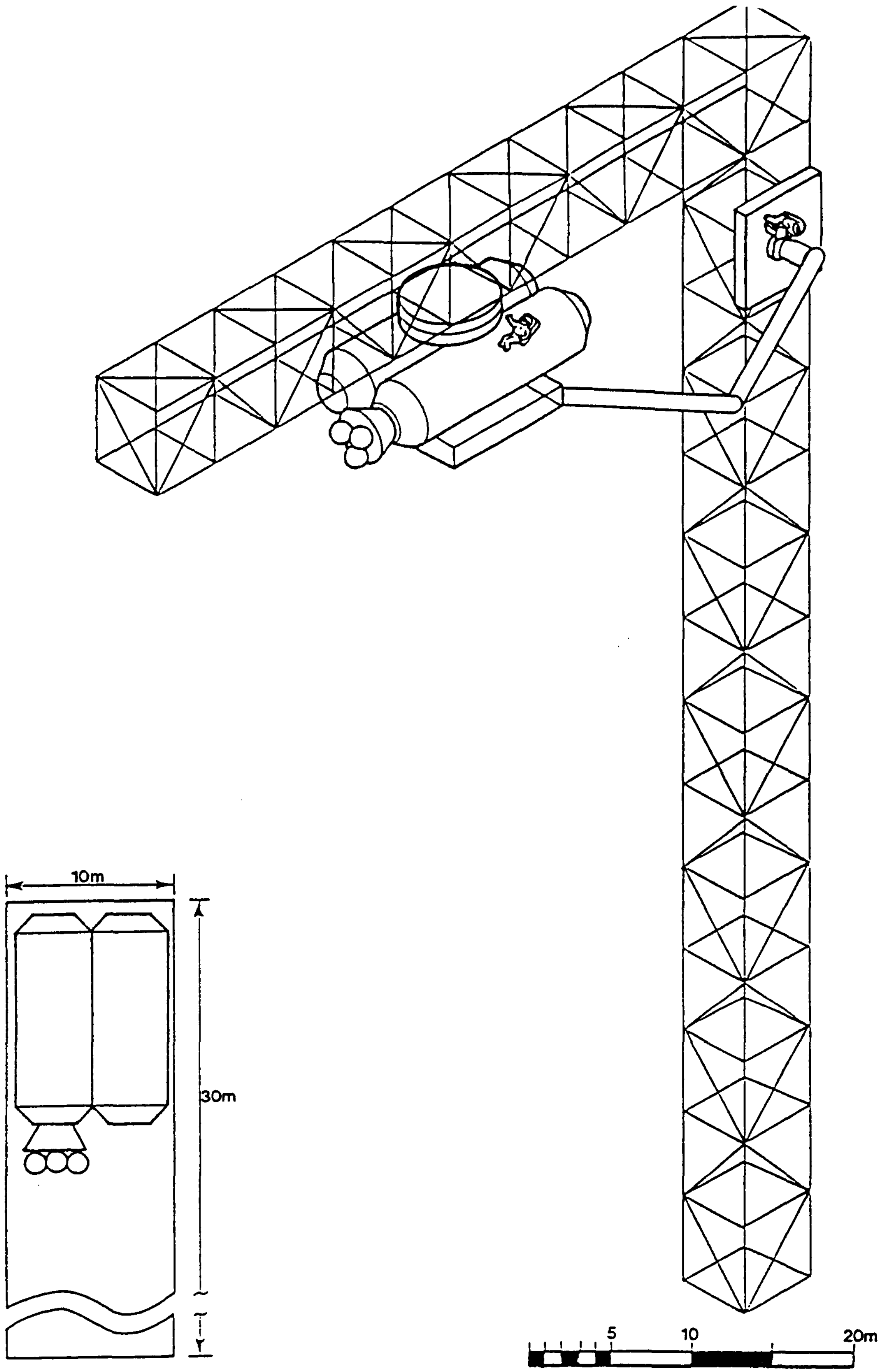
Flight #8 (ALS-7)

SSME with over-sized bell, manifold, and square truss already attached, is guided into position by RMS; all fluid lines are then connected to the manifold



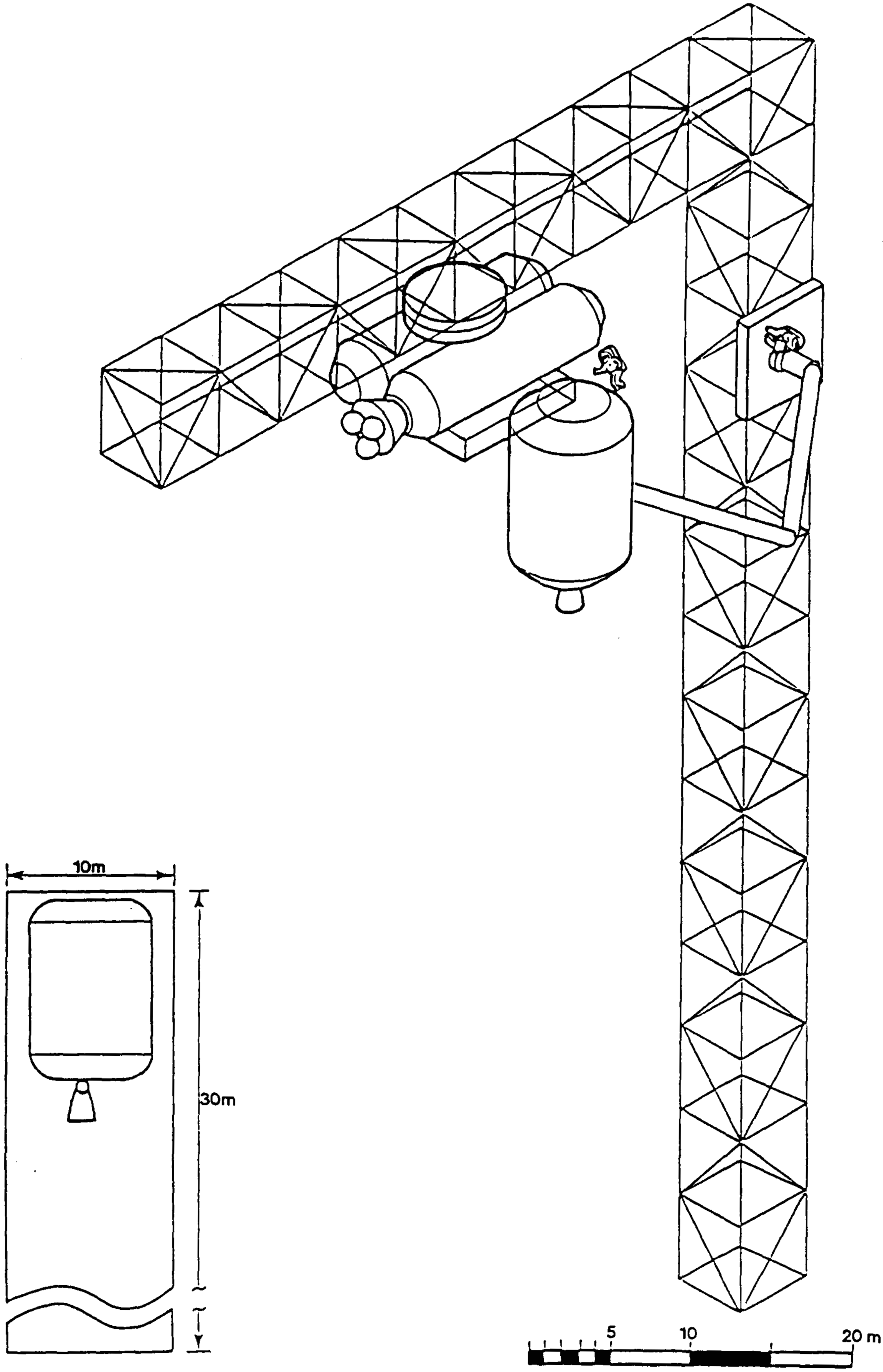
Flight #9 (STS-2)

Inspection Crew #1 (loiter on shuttle), Assembly Crew #2



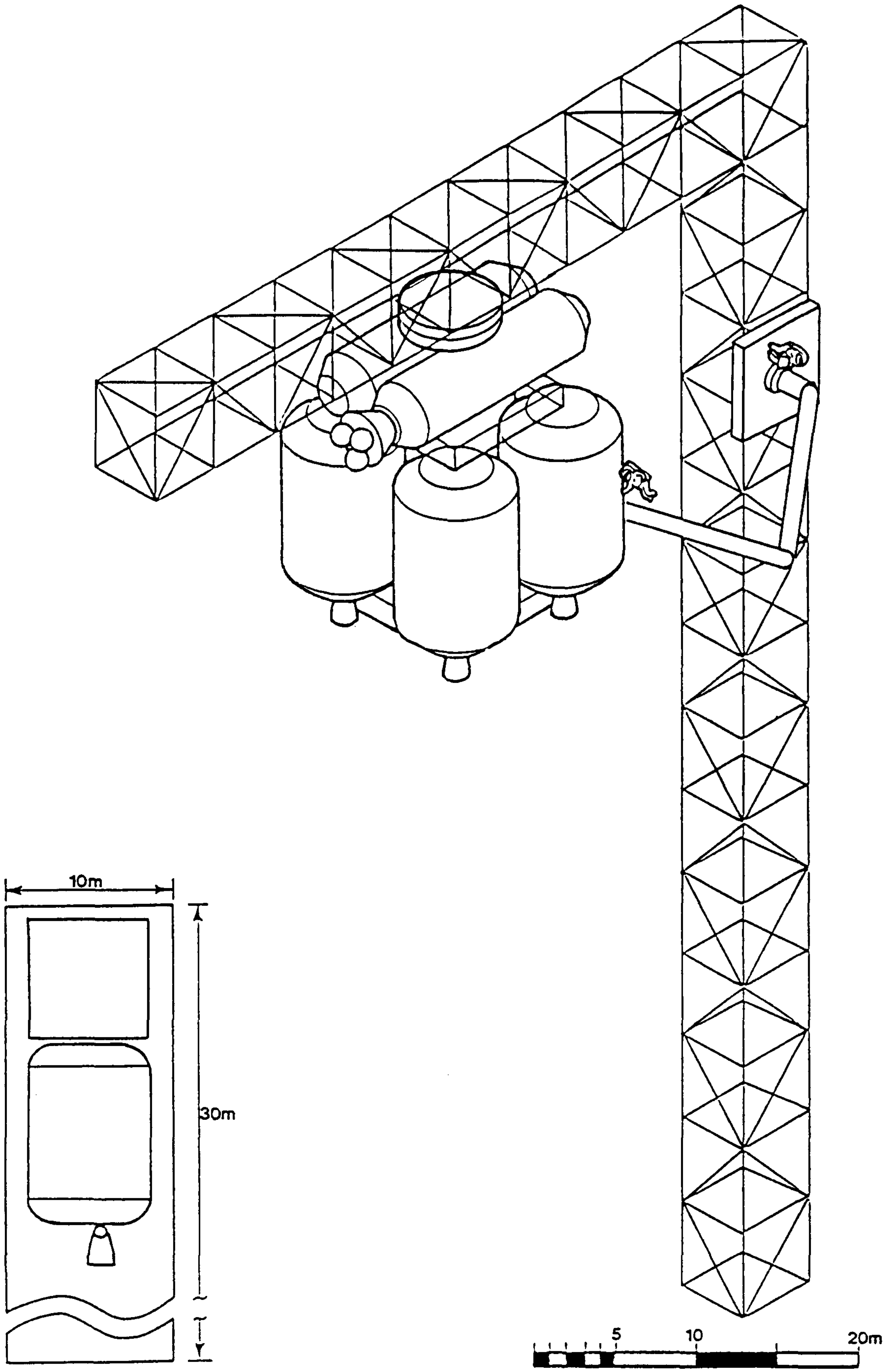
Flight #10 (ALS-8)

H module with ECCV and trusswork is attached as one cargo element to th:: rotisserie



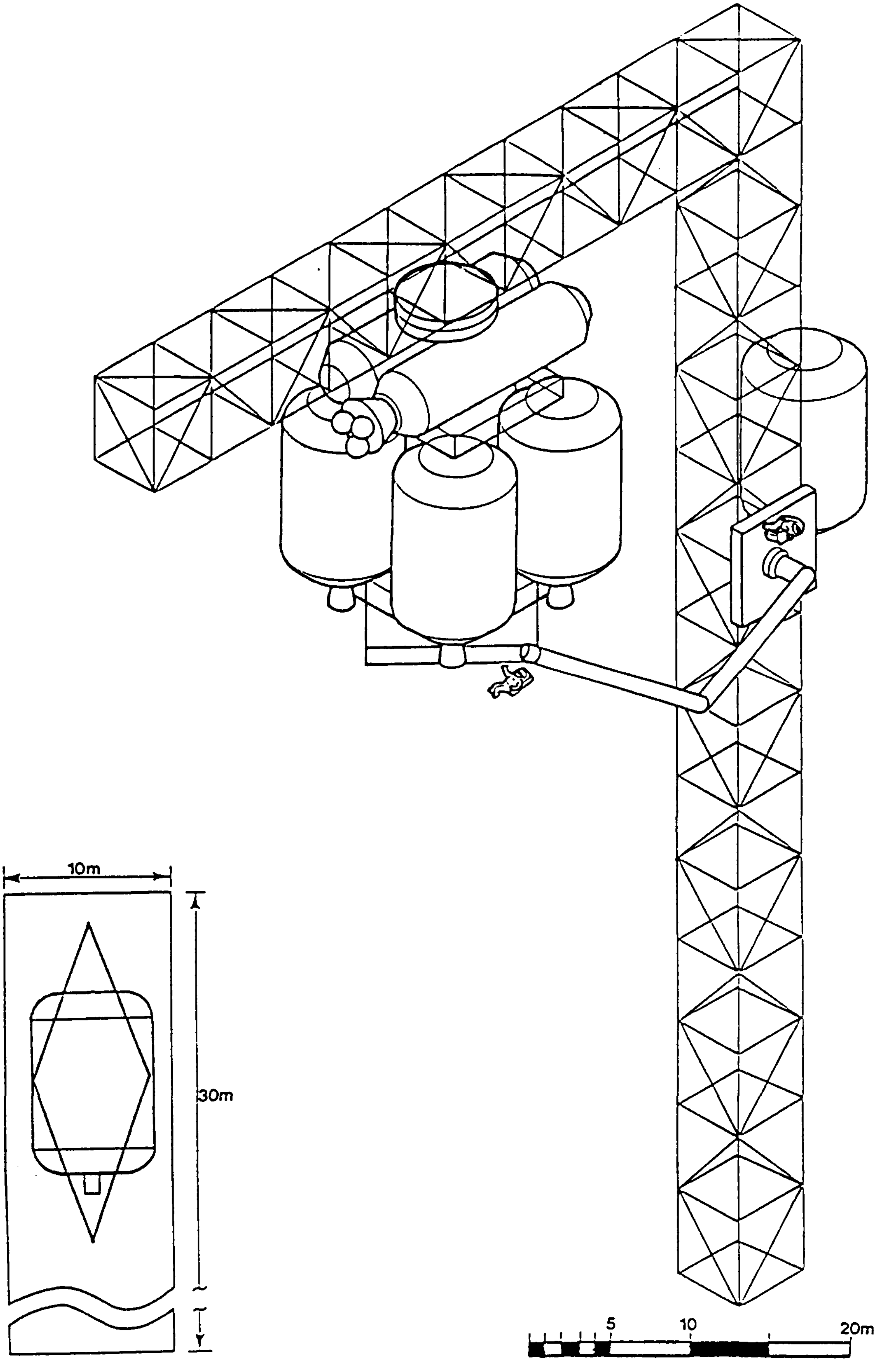
Flight #11 (ALS-9)

MOCS tank #1 is positioned under square truss



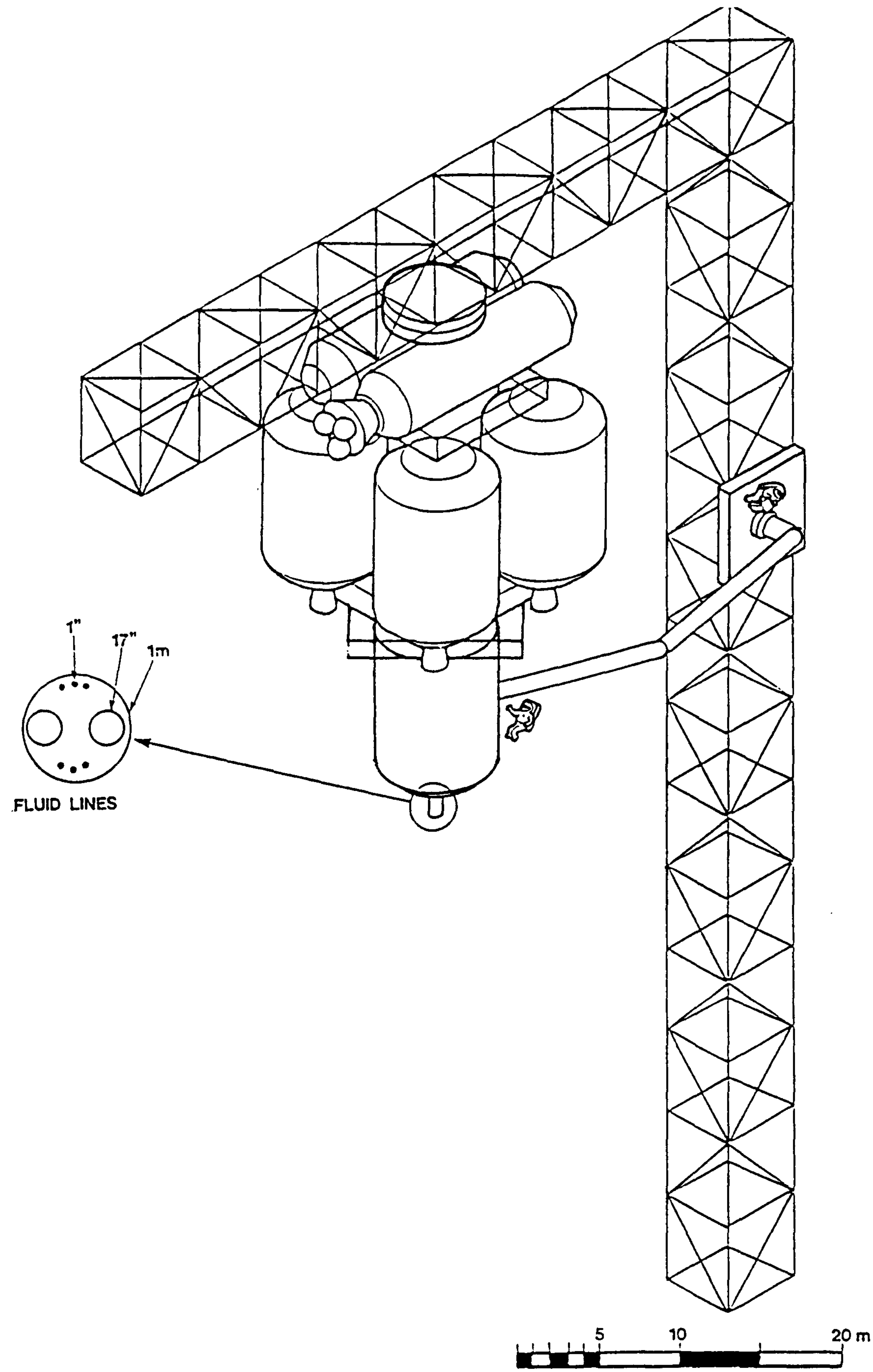
Flight #12, #13, #14 (ALS-10, 11, 12)

MOCS assembly completed including square truss

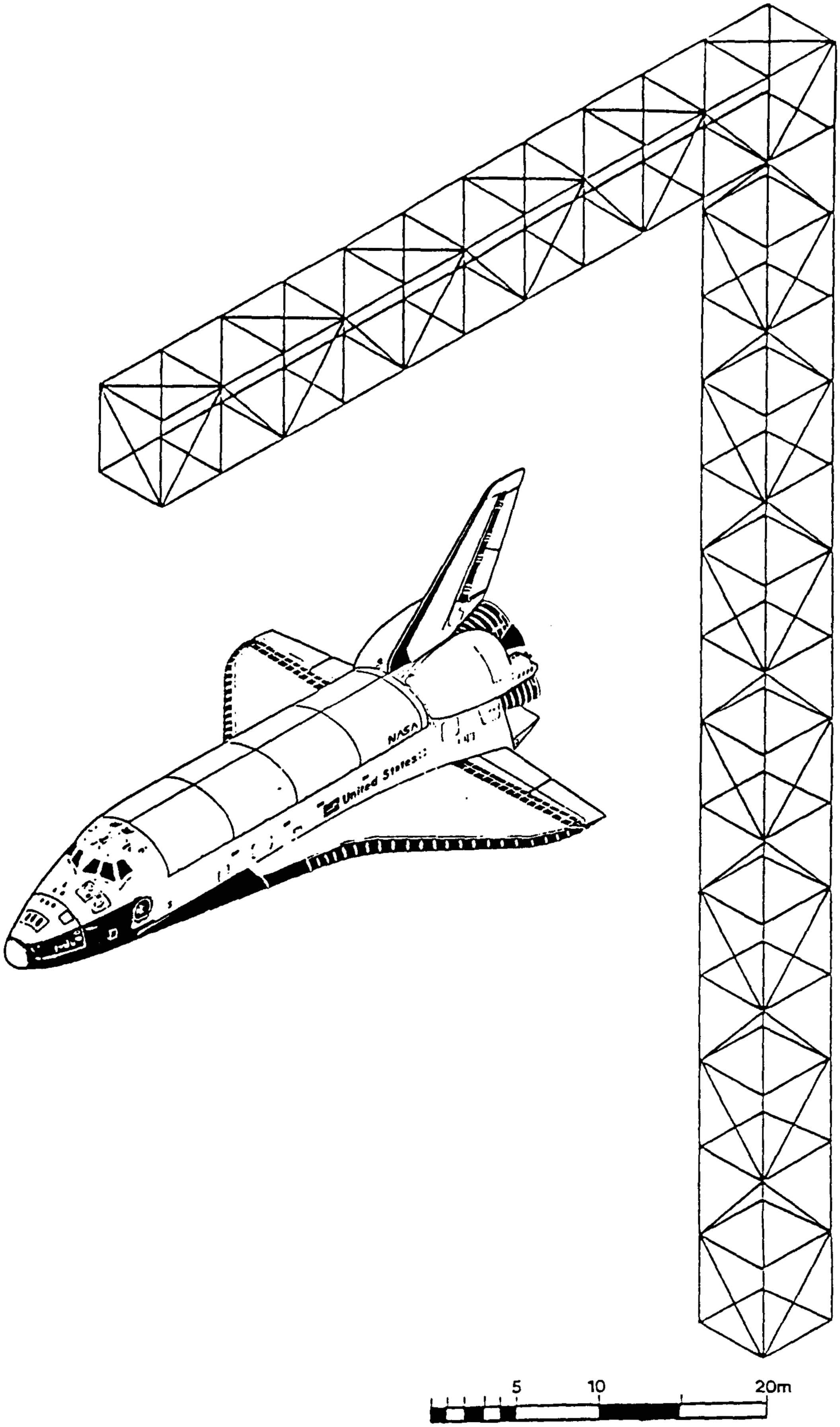


Flight #15 (ALS-13)

Square trusswork attached for TMIS tank connections

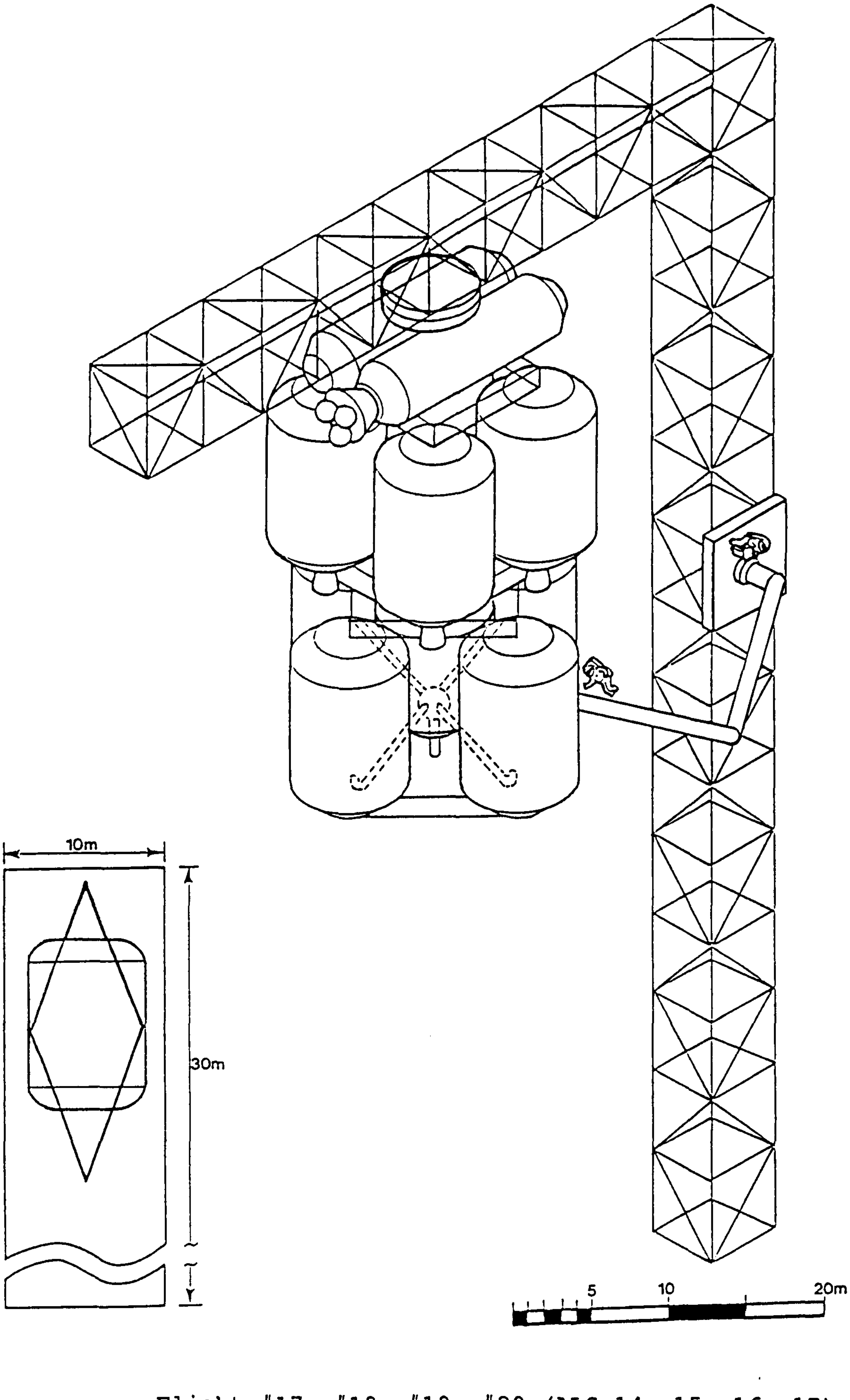


TMIS tank #1, with internal manifold, is connected to center of truss above



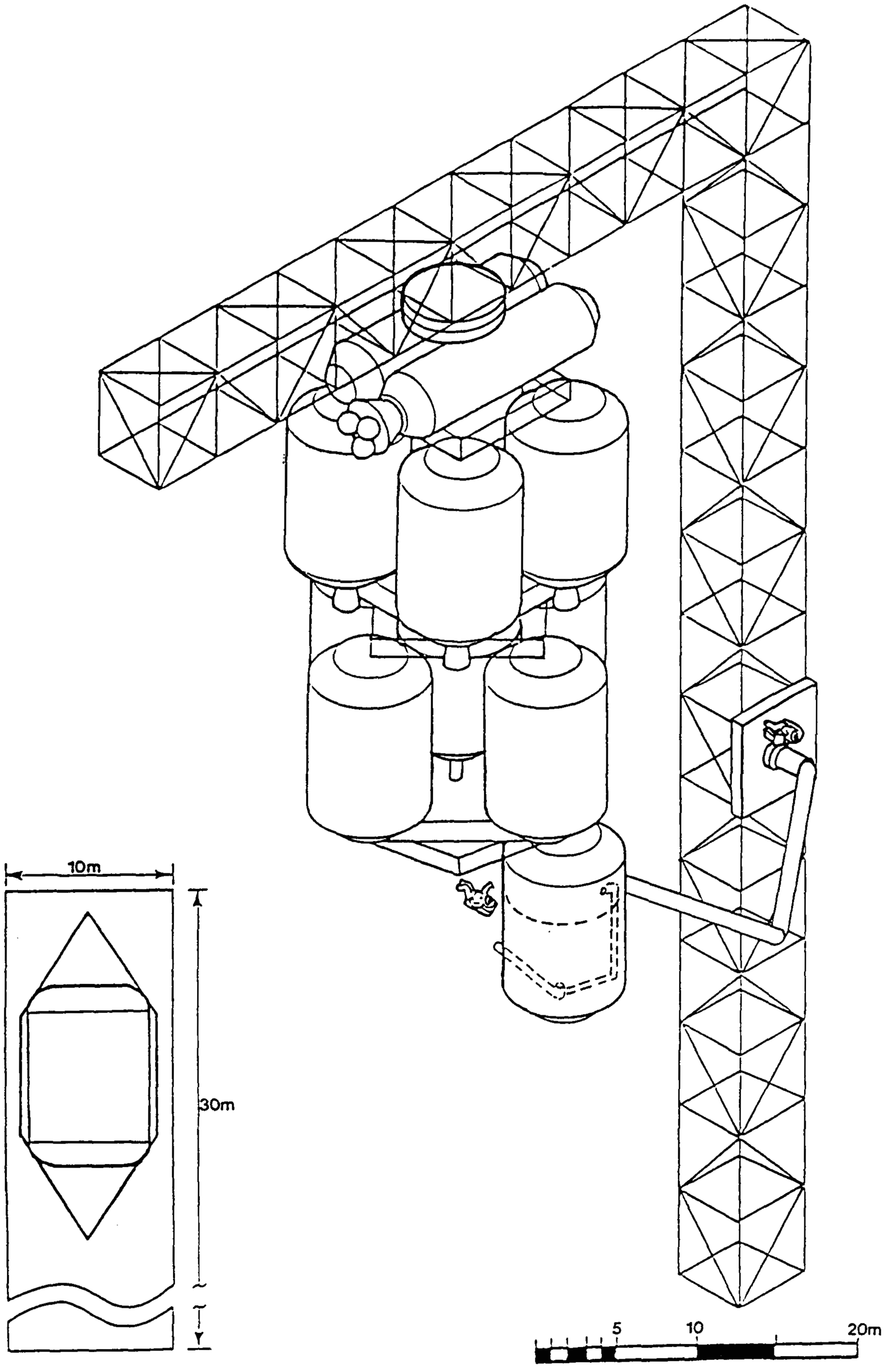
Flight #16 (STS-3)

Assembly Crew #3



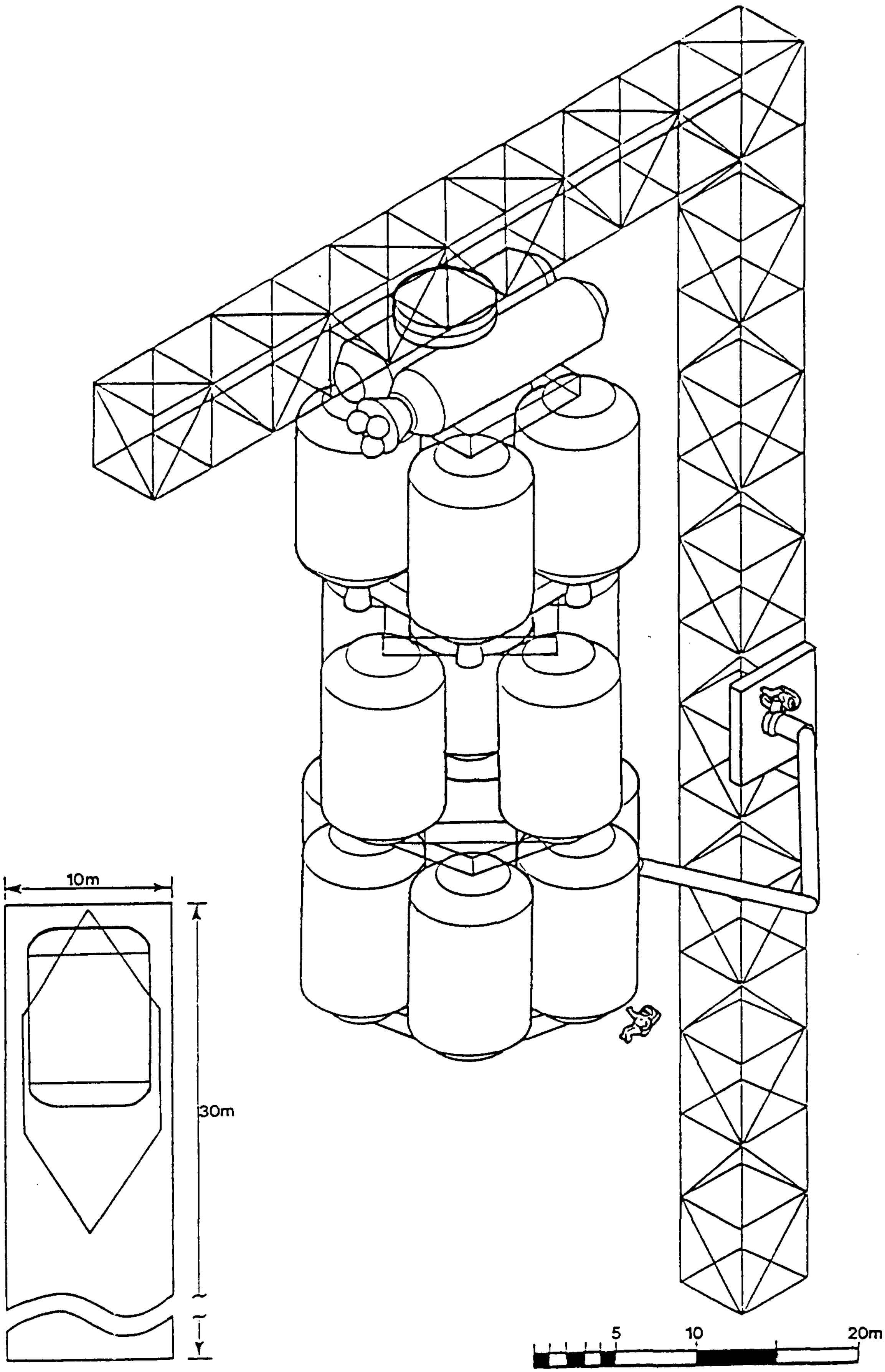
Flight #17, #18, #19, #20 (ALS-14, 15, 16, 17)

TMIS first tier tank and bottom truss assembly completed (includes piping connection from each tank to manifold in center tank)



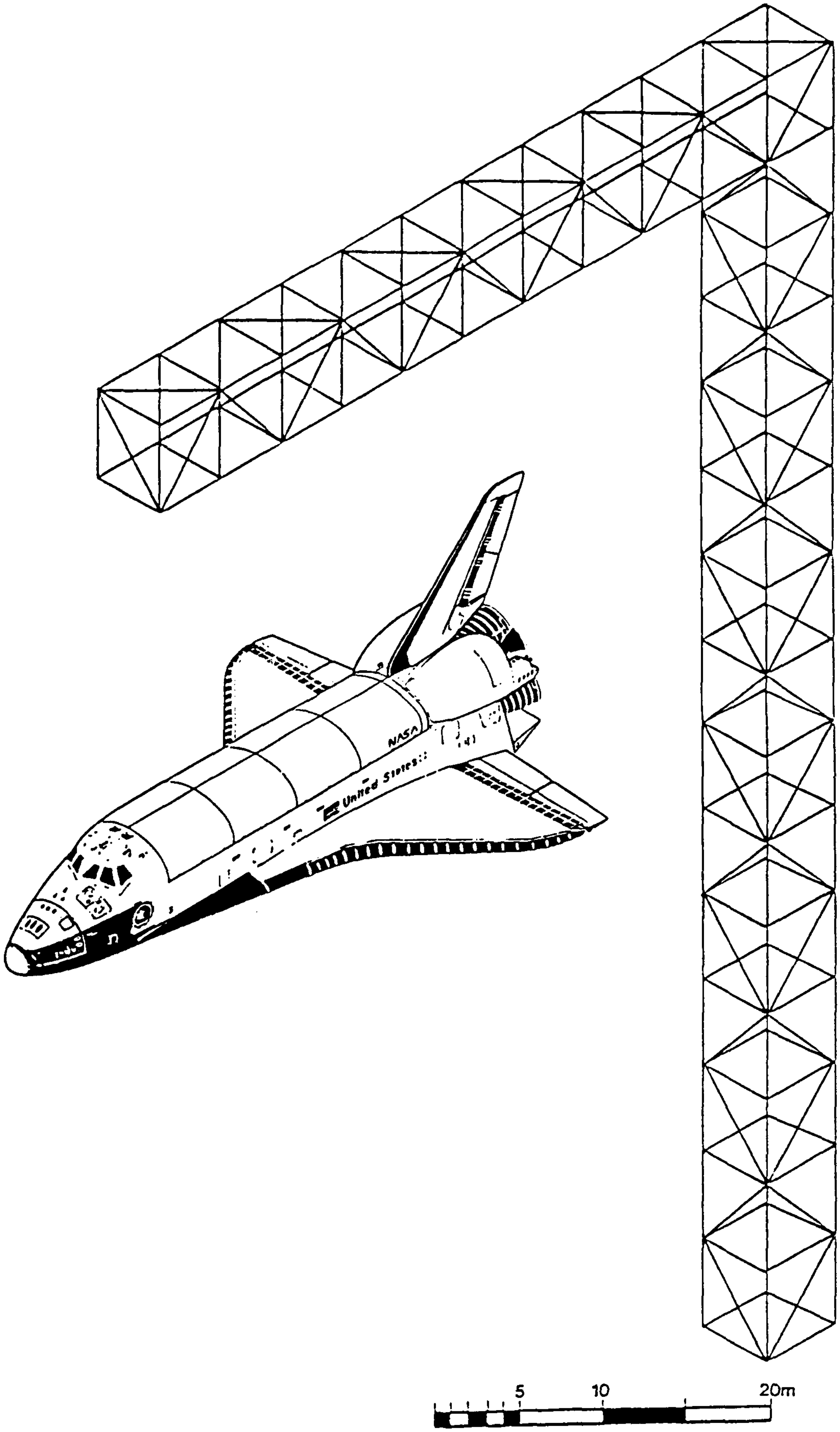
Flight #21 (ALS-18)

Hexagonal truss attached to square truss above; TMIS tank connected



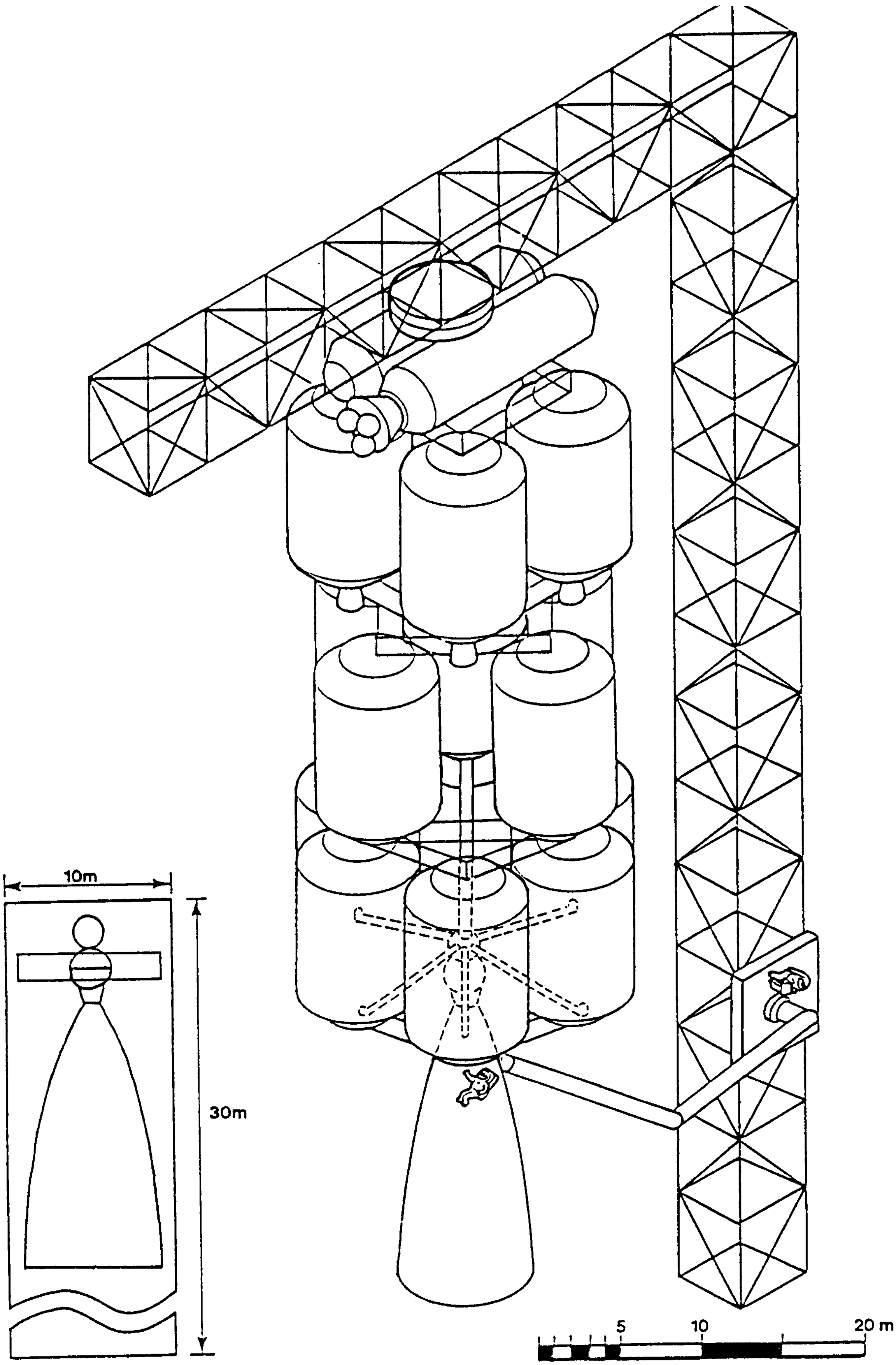
Flight #22, #24, #25, #26, #27 (ALS-19, 20, 21, 22, 23)

TMIS tank assembly completed; hexagonal truss attached to underside of tanks



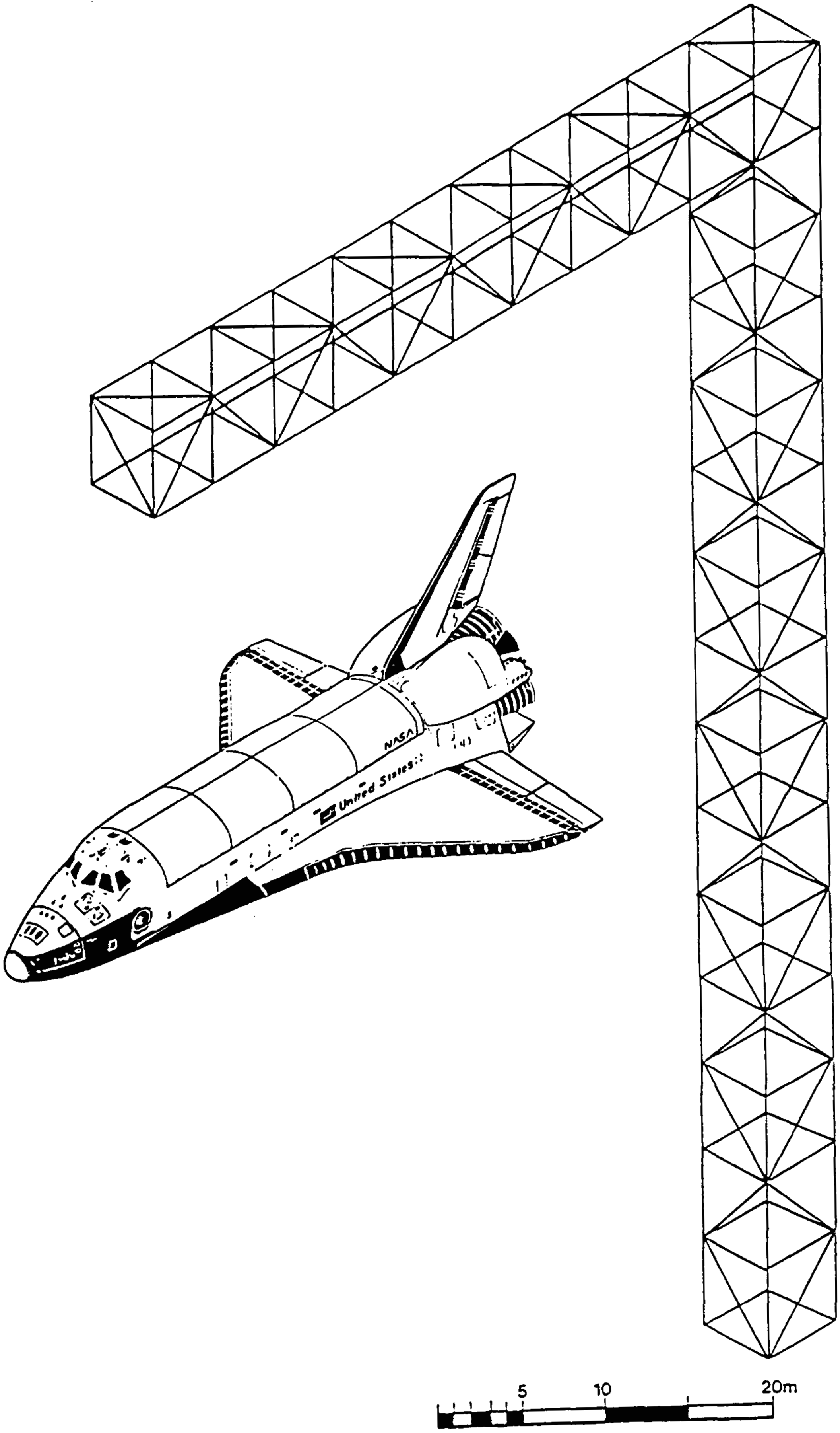
Flight #23 (STS-4)

Assembly Crew #4



Flight #28 (ALS-24)

SSME with over-sized bell, manifold, and square truss already attached, is connected to hexagonal truss above; all fluid lines are then connected to the manifold



Flight #29 (STS-5)

Flight/Inspection Crew

Table 4 Assembled (Martin) Vehicle Manifest

	Piece	Size D X L (m)	Weight (mt)
Cargo Vehicle (Feb '01)		18.7 X 54	556.1
ALS-1 (Jun '00)		7.5 X 30	84.2
	TEIS	7.5 X 13.5	60.8
	Phobos EV	4.1 X 6.0	9.8
	Cargo (MRSR, Support Services, other P/L)	4.1 X 5.5	9.8
	Truss Members		2.0
	Interstage		2.0
ALS-2 (Jul '00)		7.5 X 21.5	90.8
	MOSC	7.5 X 18.8	57.9
	MOOS		19.6
	MTX (RCS, MCC)		5.3
	Square Truss		4.0
	Interstage		4.0
ALS-3 (Aug '00)		7.5 X 11.0	91.0
	TMIS (tank #1)	7.5 X 11.0	91.0
ALS-4 (Sep '00)		7.5 X 11.0	91.0
	TMIS (tank #2)	7.5 X 11.0	91.0
ALS-5 (Oct '00)		7.5 X 11.0	91.0
	TMIS (tank #3)	7.5 X 11.0	91.0
ALS-6 (Nov '00)		7.5 X 11.0	91.0
	TMIS (tank #4)	7.5 X 11.0	91.0
ALS-7 (Dec '00)		9.0 X 20.0	17.1
	SSME+	9.0 X 20.0	13.1
	Square Truss		4.0

Table 4 (Continued)

		Piece	Size D X L (m)	Weight (mt)
Piloted Vehicle (Aug '02)			24.7 X 56	1,484.0
ALS-8	(Feb '01)		9.0 X 20.0	81.6
		HAB Module (w/equip.)	9.0 X 20.0	52.2
		ECCV		6.9
		MTX (MCC,RCS,etc.)		18.5
		Square Truss		4.0
ALS-9	(Mar '01)		7.5 X 13.5	91.0
		MOCS (tank #1)	7.5 X 13.5	91.0
		MOO-1		
ALS-10	(Apr '01)		7.5 X 13.5	91.0
		MOCS (tank #2)	7.5 X 13.5	91.0
		MOO-1		
ALS-11	(May '01)		7.5 X 13.5	91.0
		MOCS (tank #3)	7.5 X 13.5	91.0
		MOO-1		
ALS-12	(Jun '01)		7.5 X 13.5	95.0
		MOCS (tank #4)	7.5 X 13.5	91.0
		MOO-1		
		Square Truss		4.0
ALS-13	(Jul '01)		7.5 x 13.0	95.0
		TMIS (tank #1)	7.5 X 11.0	91.0
		Square Truss		4.0
ALS-14	(Aug '01)		7.5 X 11.0	91.0
		TMIS (tank #2)	7.5 X 11.0	91.0
ALS-15	(Sep '01)		7.5 X 11.0	91.0
		TMIS (tank #3)	7.5 X 11.0	91.0

Table 4 (Continued)

		Piece	Size D X L (m)	Weight (mt)
ALS-16	(Oct '01)		7.5 X 11.0	91.0
		TMIS (tank #4)	7.5 X 11.0	91.0
ALS-17	(Nov '01)		7.5 X 13.0	95.0
		TMIS (tank #5) Square Truss	7.5 X 11.0	91.0 4.0
ALS-18	(Dec '01)		7.5 X 13.0	95.0
		TMIS (tank #6) Hexagonal Truss	7.5 X 11.0	91.0 4.0
ALS-19	(Jan '02)		7.5 X 11.0	91.0
		TMIS (tank #7)	7.5 X 11.0	91.0
ALS-20	(Feb '02)		7.5 X 11.0	91.0
		TMIS (tank #8)	7.5 X 11.0	91.0
ALS-21	(Mar '02)		7.5 X 11.0	91.0
		TMIS (tank #9)	7.5 X 11.0	91.0
ALS-22	(Apr '02)		7.5 X 11.0	91.0
		TMIS (tank #10)	7.5 X 11.0	91.0
ALS-23	(May '02)		7.5 X 13.0	95.0
		TMIS (tank #11) Hexagonal Truss	7.5 X 11.0	91.0 4.0
ALS-24	(Jun '02)		9.0 X 20.0	17.1
		SSME+ Square Truss	9.0 X 20.0	13.1 4.0

Table 5 Task Description and EVA/IVA Requirements for On-Orbit Assembly

Typical Tasks for On-Orbit Assembly	Requirements (hrs) EVA		IVA	
1. Connect 17" hydrogen and oxygen main fuel lines	10		15	
- Align and connect		2		
- Insert and torque bolts to specifications		4		
- Leak check with pressurized gas		1		
- Leak check with cryogenic fluid		3		
2. Connect 1" hydrogen and oxygen recirculation lines	1		1	
- Align and connect		0.5		
- Torque connections to specifications		0.5		
3. Connect 1" hydrogen and oxygen helium press. lines	1		1	
- Align and connect		0.5		
- Torque connections to specifications		0.5		
4. Connect 1" hydrogen and oxygen vent lines	1		1	
- Align and connect		0.5		
- Torque connections to specifications		0.5		
5. Leak check for recirc., press., and vent lines	3		6	
- Leak check with pressurized gas		1		
- Leak check with cryogenic fluid		2		
- Disconnect and repair if required				
6. Make structural connection of truss to truss capable of taking 100 ton loads	1.25		1.25	
- Align and connect		0.50		
- Insert and torque bolts to specifications		0.50		
- Inspect connection		0.25		
7. Conduct TMI engine hot fire test and perform post fire inspection	8		8	
8. Conduct RCS attitude control tests			2	
9. Deploy and test power system	4		4	
10. Make and test electrical connection	1		1	
11. Conduct vibration test	24		24	
- Install shaker		8		
- Standby during test		8		
- Remove shaker		8		
12. Inspection	4		4	
13. Top off tanks	6		6	
14. Remove steel manifold plate	1		1	
15. Align and secure H module into docking position	5		5	

Table 5 (Continued)

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Typical Tasks for On-Orbit Assembly	Requirements (hrs) EVA		IVA	
(H module has ECCV and truss structure attached)				
16. Remotely manipulate (RMS) tank from docking to assembly position	6		6	
- RMS moved into retrieve pos. along backbone		1		
- RMS retrieves tank		2		
- RMS moves to assembly position		2		
- Initial alignment and attachment		1		
17. Remotely rotate structure			1	
18. Make structural connection of truss to tank capable of taking 100 ton loads	2.25		2.25	
- Align and connect		0.5		
- Insert and torque bolts to specifications		1.5		
- Inspect connection		0.25		
19. Make structural connection of tank to truss capable of taking 100 ton loads	3.25		3.25	
- Align and connect		1.5		
- Insert and torque bolts to specifications		1.5		
- Inspect connection		0.25		
20. Remotely manipulate (RMS) less than 10 tons of cargo into assembly position	4		4	
- RMS moved into retrieve pos. along backbone		1		
- RMS retrieves cargo element		1		
- RMS moves to assembly position		1		
- Initial alignment and attachment		1		
21. Make structural connection of tank to tank capable of taking 100 ton loads	7		7	
- Align and connect (includes rotations)		2		
- Insert and torque bolts to specifications		4		
- Inspect connection		1		
22. Make structural connection of tank to MSS rotisserie capable of taking 100 ton loads	8.5		8.5	
- Align and connect (includes rotations)		2.5		
- Insert and torque bolts to specifications		5		
- Inspect connection and test rotisserie		1		
23. Shuttle RMS deployment of hab module adapter			2	
- Attach arm to adapter and remove from cargo bay				1
- Deploy adapter for attachment				1
24. Remotely manipulate (RMS) SSME+/truss from docking to assembly position	5		5	
- RMS moved into retrieve pos. along backbone		1		
- RMS retrieves SSME+		2		

Table 5 (Continued)

Typical Tasks for On-Orbit Assembly	Requirements (hrs) EVA		IVA	
- RMS moves to assembly position		1		
- Initial alignment and attachment		1		
25. Make structural connection of SSME+/truss to one tank capable of taking 100 ton loads	2.25		2.25	
- Align and connect		0.5		
- Insert and torque bolts to specifications		1.5		
- Inspect connection		0.25		
26. Make structural connection between adapter and two hab modules	5		5	
- Align and connect		1		
- Insert and torque bolts to specifications		3		
- Inspect connection		1		
27. Detach from station			1	
28. Remotely controlled moving of vehicle for testing			6	
- OMV retrieval				1
- OMV tug				4
- OMV detachment				1
29. Remotely control stage docking			6	
- OMV rendezvous				2
- OMV final docking				4

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Table 6 Task Sequence and EVA/IVA Requirements for Assembly of Martin Vehicle

Heavy-Lift Flight	Task Number	Requirement (hrs) EVA	IVA	Brief Description of Task
Cargo Vehicle				
1	29	0.00	6.00	dock
	16	6.00	6.00	get tank
	22	8.50	8.50	fit tank
	20	4.00	4.00	get truss
	18	2.25	2.25	fit truss
	20	4.00	4.00	get cargo
	6	1.25	1.25	fit cargo
	17	0.00	1.00	rotate
	20	4.00	4.00	get truss
	18	2.25	2.25	fit truss
	20	4.00	4.00	get Phev
	6	1.25	1.25	fit Phev
	9	4.00	4.00	power test
	10	1.00	1.00	elec. for cargo
	10	1.00	1.00	elec. for Phev
	10	1.00	1.00	elec. for tank
		44.50	51.50	
2	29	0.00	6.00	dock
	16	6.00	6.00	get tank/truss
	21	7.00	7.00	fit tank/truss
	10	1.00	1.00	elec. for tank
		14.00	20.00	
3	29	0.00	6.00	dock
	16	6.00	6.00	get tank
	19	3.25	3.25	fit tank
	10	1.00	1.00	elec. for tank
		10.25	16.25	
4	17	0.00	1.00	rotate
	29	0.00	6.00	dock
	16	6.00	6.00	get tank
	19	3.25	3.25	fit tank
	10	1.00	1.00	elec. for tank
		10.25	17.25	
5	17	0.00	1.00	rotate
	29	0.00	6.00	dock
	16	6.00	6.00	get tank
	19	3.25	3.25	fit tank
	10	1.00	1.00	elec. for tank
		10.25	17.25	

Table 6 (Continued)

Heavy-Lift Flight	Task Number	Requirement (hrs) EVA	IVA	Brief Description of Task
6	17	0.00	1.00	rotate
	29	0.00	6.00	dock
	16	6.00	6.00	get tank
	19	3.25	3.25	fit tank
	10	1.00	1.00	elec. for tank
		10.25	17.25	
7	29	0.00	6.00	dock
	24	5.00	5.00	get SSME/truss
	25	2.25	2.25	SSME/truss to tank
	25	2.25	2.25	SSME/truss to tank
	25	2.25	2.25	SSME/truss to tank
	25	2.25	2.25	SSME/truss to tank
	10	1.00	1.00	elec. for SSME
	14	1.00	1.00	remove plate on manifold
	1	10.00	15.00	install and check main
	2	1.00	1.00	install recirc
	3	1.00	1.00	install press
	4	1.00	1.00	install vent
	5	3.00	6.00	check minor lines
	14	1.00	1.00	remove plate
	1	10.00	15.00	install and check main
	2	1.00	1.00	install recirc
	3	1.00	1.00	install press
	4	1.00	1.00	install vent
	5	3.00	6.00	check minor lines
	14	1.00	1.00	remove plate
	1	10.00	15.00	install and check main
	2	1.00	1.00	install recirc
	3	1.00	1.00	install press
	4	1.00	1.00	install vent
	5	3.00	6.00	check minor lines
	14	1.00	1.00	remove plate
	1	10.00	15.00	install and check main
	2	1.00	1.00	install recirc
	3	1.00	1.00	install press
	4	1.00	1.00	install vent
	5	3.00	6.00	check minor lines
		83.00	121.00	
SUBTOTAL		182.50	260.50	30.42
Post Assembly	12	4.00	4.00	inspection of vehicle
	27	0.00	1.00	detach
	28	0.00	6.00	move away
	8	0.00	2.00	attitude test
	7	8.00	8.00	engine test

Table 6 (Continued)

Heavy-Lift Flight	Task Number	Requirement (hrs) EVA	IVA	Brief Description of Task
	11	24.00	24.00	vibration test
	13	6.00	6.00	top-off tanks
TOTAL		224.50	311.50	
	*** LAUNCH CARGO VEHICLE ***			
Piloted Vehicle				
8	29	0.00	6.00	dock
	16	6.00	6.00	get module/truss
	15	5.00	5.00	fit module/truss
	9	4.00	4.00	power test
	10	1.00	1.00	elec. for H module
	10	1.00	1.00	elec. for H module
	10	1.00	1.00	elec. for ECCV
		18.00	24.00	
9	29	0.00	6.00	dock
	16	6.00	6.00	get tank
	19	3.25	3.25	fit tank
	10	1.00	1.00	elec. for tank
		10.25	16.25	
10	17	0.00	1.00	rotate
	29	0.00	6.00	dock
	16	6.00	6.00	get tank
	19	3.25	3.25	fit tank
	10	1.00	1.00	elec. for tank
		10.25	17.25	
11	17	0.00	1.00	rotate
	29	0.00	6.00	dock
	16	6.00	6.00	get tank
	19	3.25	3.25	fit tank
	10	1.00	1.00	elec. for tank
		10.25	17.25	
12	17	0.00	1.00	rotate
	29	0.00	6.00	dock
	16	6.00	6.00	get tank
	19	3.25	3.25	fit tank
	10	1.00	1.00	elec. for tank
	20	4.00	4.00	get truss
	6	1.25	1.25	fit truss
	6	1.25	1.25	fit truss

Table 6 (Continued)

Heavy-Lift Flight	Task Number	Requirement (hrs) EVA	IVA	Brief Description of Task
	6	1.25	1.25	fit truss
	6	1.25	1.25	fit truss
	18	2.25	2.25	fit truss
	18	2.25	2.25	fit truss
	18	2.25	2.25	fit truss
	18	2.25	2.25	fit truss
		28.25	35.25	
13	29	0.00	6.00	dock
	20	4.00	4.00	get truss
	6	1.25	1.25	fit truss
	6	1.25	1.25	fit truss
	6	1.25	1.25	fit truss
	6	1.25	1.25	fit truss
	6	1.25	1.25	fit truss
	6	1.25	1.25	fit truss
	6	1.25	1.25	fit truss
	6	1.25	1.25	fit truss
	16	6.00	6.00	get tank
	19	3.25	3.25	fit tank
	10	1.00	1.00	elec. for tank
		24.25	30.25	
14	17	0.00	1.00	rotate
	29	0.00	6.00	dock
	16	6.00	6.00	get tank
	19	3.25	3.25	fit tank
	14	1.00	1.00	remove plate
	1	10.00	15.00	install and check main
	2	1.00	1.00	install recirc
	3	1.00	1.00	install press
	4	1.00	1.00	install vent
	5	3.00	6.00	check minor lines
	10	1.00	1.00	elec. for tank
		27.25	42.25	
15	17	0.00	1.00	rotate
	29	0.00	6.00	dock
	16	6.00	6.00	get tank
	19	3.25	3.25	fit tank
	14	1.00	1.00	remove plate
	1	10.00	15.00	install and check main
	2	1.00	1.00	install recirc
	3	1.00	1.00	install press
	4	1.00	1.00	install vent
	5	3.00	6.00	check minor lines
	10	1.00	1.00	elec. for tank

Table 6 (Continued)

Heavy-Lift Flight	Task Number	Requirement (hrs) EVA	IVA	Brief Description of Task
		27.25	42.25	
16	17	0.00	1.00	rotate
	29	0.00	6.00	dock
	16	6.00	6.00	get tank
	19	3.25	3.25	fit tank
	14	1.00	1.00	remove plate
	1	10.00	15.00	install and check main
	2	1.00	1.00	install recirc
	3	1.00	1.00	install press
	4	1.00	1.00	install vent
	5	3.00	6.00	check minor lines
	10	1.00	1.00	elec. for tank
		27.25	42.25	
17	17	0.00	1.00	rotate
	29	0.00	6.00	dock
	16	6.00	6.00	get tank
	19	3.25	3.25	fit tank
	14	1.00	1.00	remove plate
	1	10.00	15.00	install and check main
	2	1.00	1.00	install recirc
	3	1.00	1.00	install press
	4	1.00	1.00	install vent
	5	3.00	6.00	check minor lines
	10	1.00	1.00	elec. for tank
	20	4.00	4.00	get truss
	6	1.25	1.25	fit truss (square)
	6	1.25	1.25	fit truss (square)
	6	1.25	1.25	fit truss (square)
	6	1.25	1.25	fit truss (square)
	20	4.00	4.00	get truss
	6	1.25	1.25	fit truss (hex)
	6	1.25	1.25	fit truss (hex)
	6	1.25	1.25	fit truss (hex)
	6	1.25	1.25	fit truss (hex)
	6	1.25	1.25	fit truss (hex)
	6	1.25	1.25	fit truss (hex)
	6	1.25	1.25	fit truss (hex & square)
	6	1.25	1.25	fit truss (hex & square)
	6	1.25	1.25	fit truss (hex & square)
	6	1.25	1.25	fit truss (hex & square)
	18	2.25	2.25	fit truss
	18	2.25	2.25	fit truss
	18	2.25	2.25	fit truss
	18	2.25	2.25	fit truss
		61.75	76.75	

Table 6. (Continued)

Heavy-Lift Flight	Task Number	Requirement (hrs) EVA	IVA	Brief Description of Task
18	29	0.00	6.00	dock
	16	6.00	6.00	get tank
	19	3.25	3.25	fit tank
	10	1.00	1.00	elec. for tank
		10.25	16.25	
19	17	0.00	1.00	rotate
	29	0.00	6.00	dock
	16	6.00	6.00	get tank
	19	3.25	3.25	fit tank
	10	1.00	1.00	elec. for tank
		10.25	17.25	
20	17	0.00	1.00	rotate
	29	0.00	6.00	dock
	16	6.00	6.00	get tank
	19	3.25	3.25	fit tank
	10	1.00	1.00	elec. for tank
		10.25	17.25	
21	17	0.00	1.00	rotate
	29	0.00	6.00	dock
	16	6.00	6.00	get tank
	19	3.25	3.25	fit tank
	10	1.00	1.00	elec. for tank
		10.25	17.25	
22	17	0.00	1.00	rotate
	29	0.00	6.00	dock
	16	6.00	6.00	get tank
	19	3.25	3.25	fit tank
	10	1.00	1.00	elec. for tank
		10.25	17.25	
23	17	0.00	1.00	rotate
	29	0.00	6.00	dock
	16	6.00	6.00	get tank
	19	3.25	3.25	fit tank
	10	1.00	1.00	elec. for tank
	20	4.00	4.00	get truss
	6	1.25	1.25	fit truss (hex)
	6	1.25	1.25	fit truss (hex)
	6	1.25	1.25	fit truss (hex)
	6	1.25	1.25	fit truss (hex)

<u>Table 6 (Continued)</u>

Heavy-Lift Flight	Task Number	Requirement (hrs) EVA	IVA	Brief Description of Task
	6	1.25	1.25	fit truss (hex)
	6	1.25	1.25	fit truss (hex)
	18	2.25	2.25	fit truss
	18	2.25	2.25	fit truss
	18	2.25	2.25	fit truss
	18	2.25	2.25	fit truss
	18	2.25	2.25	fit truss
	18	2.25	2.25	fit truss
		35.25	42.25	
24	29	0.00	6.00	dock
	24	5.00	5.00	get SSME/truss
	25	2.25	2.25	SSME/truss to truss
	25	2.25	2.25	SSME/truss to truss
	25	2.25	2.25	SSME/truss to truss
	25	2.25	2.25	SSME/truss to truss
	10	1.00	1.00	elec. for SSME
	14	1.00	1.00	remove plate (extra link)
	1	10.00	15.00	install and check main (ext
	2	1.00	1.00	install recirc (extra link)
	3	1.00	1.00	install press (extra link)
	4	1.00	1.00	install vent (extra link)
	5	3.00	6.00	check minor lines (extra li
	14	1.00	1.00	remove plate (extra link)
	1	10.00	15.00	install and check main (ext
	2	1.00	1.00	install recirc (extra link)
	3	1.00	1.00	install press (extra link)
	4	1.00	1.00	install vent (extra link)
	5	3.00	6.00	check minor lines (extra li
	14	1.00	1.00	remove plate (#1)
	1	10.00	15.00	install and check main
	2	1.00	1.00	install recirc
	3	1.00	1.00	install press
	4	1.00	1.00	install vent
	5	3.00	6.00	check minor lines
	14	1.00	1.00	remove plate (#2)
	1	10.00	15.00	install and check main
	2	1.00	1.00	install recirc
	3	1.00	1.00	install press
	4	1.00	1.00	install vent
	5	3.00	6.00	check minor lines
	14	1.00	1.00	remove plate (#3)
	1	10.00	15.00	install and check main
	2	1.00	1.00	install recirc
	3	1.00	1.00	install press
	4	1.00	1.00	install vent
	5	3.00	6.00	check minor lines
	14	1.00	1.00	remove plate (#4)
	1	10.00	15.00	install and check main

Table 6 (Continued)

Heavy-Lift Flight	Task Number	Requirement (hrs) EVA	IVA	Brief Description of Task
	2	1.00	1.00	install recirc
	3	1.00	1.00	install press
	4	1.00	1.00	install vent
	5	3.00	6.00	check minor lines
	14	1.00	1.00	remove plate (#5)
	1	10.00	15.00	install and check main
	2	1.00	1.00	install recirc
	3	1.00	1.00	install press
	4	1.00	1.00	install vent
	5	3.00	6.00	check minor lines
	14	1.00	1.00	remove plate (#6)
	1	10.00	15.00	install and check main
	2	1.00	1.00	install recirc
	3	1.00	1.00	install press
	4	1.00	1.00	install vent
	5	3.00	6.00	check minor lines
		151.00	221.00	
SUBTOTAL		482.25	692.25	80.38
Post Assembly	12	4.00	4.00	inspection of vehicle
	27	0.00	1.00	detach
	28	0.00	6.00	move away
	8	0.00	2.00	attitude test
	7	8.00	8.00	engine test
	11	24.00	24.00	vibration test
	13	6.00	6.00	top-off tanks
TOTAL		524.25	743.25	
*** LAUNCH PILOTED VEHICLE ***				

Appendix B

Docked Phobos Vehicle

Appendix B

Table 8 shows the manifest for assembly and launch of the concept including associated shuttle launches. Shuttle launches are required to replace assembly crews every six months or so. The docked vehicle requires a minimum launch rate of one HLV every 4 months and could conceivably be assembled in a short time if one launch per month rates are possible for the HLV (see Figure 4 for one launch/two month scenario).

The Gross Lift-Off Weight (GLOW), lift capability to LEO, a short description, and an illustration of all possible launch vehicles are shown in Figure 5. Data was obtained on historical or existing (Saturn V, Energia), and proposed launchers (e.g., ALS) varying in lift capability from around 40 to over 500 metric tons to LEO. The description at the bottom of this figure includes data necessary to analyze the impact of each launcher on ground handling operations (e.g., core and strapon sizes and weights). Figure 5 has the launchers in order from lowest payload capability to highest.

The staged or docked concept involves redesign of the vehicle so that assembly and support (EVA or IVA) is minimized. A sizing program was developed to investigate different docked vehicle configurations for a Phobos mission that might be easier to assemble. Each docked vehicle configuration studied partitioned the propellant into different numbers of stages that a certain preselected launch vehicle could handle. This was accomplished by segregating the total delta V requirements for a burn so that individual stage weights remained under a ceiling value. Figure 6 shows the detailed calculations.

From existing knowledge of the assembled (Martin) vehicle, a spreadsheet (Figure 6) was formulated to match previous results with docked vehicle calculations. Once confidence was gained in how these previous results were obtained, the vehicle configuration was modified to a staged TMI configuration. This ultimately reduced the mass requirements for the mission, while promoting modularity for LEO vehicle assembly.

Tankage, engine, and nozzle weight estimates were approximated using percentages of the total fuel. This level of detail was assumed appropriate for this initial study. The values of 15% and 18% were used for normal stage propulsion systems with the 18% reserved for the extended bell nozzles of the modified SSMEs (used in TMI stages). The MCC and RCS stage propellant systems used 50% and 100% factors respectively. Some performance parameters were adjusted from the baseline configuration in order to keep thrust to weight and burn times reasonable.

There are several factors contributing to the dissimilarities in the masses of both the Martin and docked vehicle configurations. The mass savings for the selected docked piloted vehicle (3 TMI stages) result from the following factors (seen in Figure 6). First, the piloted vehicle's MOO2 propellant and staging is now transported to a Martian high circular orbit by the cargo vehicle (direct savings ~ 25,000 kg). It should be noted that this deviation affects all the previous staging systems. Primarily, it affects the MOC, since a large delta V is associated with this maneuver. With a large delta V, a small relative drop in mass after burn results in a dramatic drop in propellant requirements (total savings ~78,000 kg). The opposite occurs for the cargo vehicle which acquires a new mass. The second fact is that staging for a large delta V maneuver like TMI produces considerable savings compared to no staging (inert weights are dropped after each stage burn for a savings of ~ 146,000 kg). There is an additional 130,000 kg saved on the TMI system due simply to shifting the MOO2 propellant to the cargo vehicle.

Table 7 Task Sequence and EVA/IVA Requirements for Assembly of Docked Vehicle

Heavy-Lift Flight	Task Number	Requirement (hrs) EVA	IVA	Brief Description of Task
Cargo Vehicle				
3	29	0.00	6.00	dock
	21	7.00	7.00	fit tank
	10	1.00	1.00	elec. for tank
4	29	0.00	6.00	dock
	21	7.00	7.00	fit tank
	10	1.00	1.00	elec. for tank
5	29	0.00	6.00	dock
	21	7.00	7.00	fit tank
	10	1.00	1.00	elec. for tank
Post Assembly	12	4.00	4.00	inspection
TOTAL		28.00	46.00	
*** LAUNCH CARGO VEHICLE ***				
Piloted Vehicle				
2	23	0.00	2.00	deploy adapter
	26	5.00	5.00	fit adapter
7	29	0.00	6.00	dock
	21	7.00	7.00	fit tank
	10	1.00	1.00	elec. for tank
9	29	0.00	6.00	dock
	21	7.00	7.00	fit tank
	10	1.00	1.00	elec. for tank
10	29	0.00	6.00	dock
	21	7.00	7.00	fit tank
	10	1.00	1.00	elec. for tank
12	29	0.00	6.00	dock
	21	7.00	7.00	fit tank
	10	1.00	1.00	elec. for tank
Post Assembly	12	4.00	4.00	inspection
TOTAL		41.00	67.00	
*** LAUNCH PILOTED VEHICLE ***				

Table 8. Docked Vehicle Manifest

Items (Payload)	Launch#	Date	Mass *	Mass *	Mass *	Dimensions
1. CREW :IMM+Ext Srvcs, & other (20000kg)	HLV-1	June '00	133,488			
CARGO: MOO1 CREW: TEIS						
CREW: IMM				43,250		DIA. - 4.5m / LEN. - 13m
Cylinderical Module(s)					31,000	
Disk Module(s)					0	
Tunnel(s)					900	
Resource Nodes (docking, prox ops)					3,000	DIA. 4.4m / LEN. 11m
Airlock(s)					650	
Radiation Shelter Shielding					2,000	
Life Support Systems (LSS)					2,800	
Data Management System (DMS)					300	
Internal Com/EPS/TCS					2,600	
CREW: External Services				1,050		
Electrical Power System, external					400	
Thermal Control System, external					500	
Communications System, external					150	
Other				20,000		DIA.-5.5m/ LEN.-4m
1 fully loaded OMV & other support systems for assembly					20,000	
CARGO: MOO1 CREW: TEIS				69,188		
Total TEI propellant (+FPR,Unusable prop.)					49,832	
TEI Prop Sys+Interstage					9,085	Ox tank-4.52m dia; H tank-6.3m dia; Engine-1m; Nozzel- 2.5m
Structure					500	
Payload						DIA.-5.5m/ LEN.-4m
MOSE					150	
Satellites					2,000	
Teleoperated MRSR					7,000	
Support Services					320	
Solar/SPE monit. (ISE)					200	
Astro/Planetary (ISE)					100	
2. ASSEMBLY CREW & SUPPLIES	STS-1	June '00	14,613			
Assembly Crew #1 (6)				450		
Consumables				6,000		
1 fully loaded OMV				8,163		

* All masses are in kilograms

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Table 8. (Continued)

Items (Payload)	Launch#	Date	Mass	Mass	Mass	Dimensions
3. CARGO: MOO1, MOC-A CREW: MOO2, PhEV	HLV-2	Aug '00	136,023			
CARGO: MOO1				26,262		
MOO1 Interstage					300	
MOO1 Prop Sys/Stages					4,375	Ox tank-3.14m dia; H tank-4.38m dia; Engine-1m; Nozzel- 2.5m
MOO1 RCS prop.					1,490	
MOO1 FPR prop					577	
MOO1 Unusable prop					288	
MOO1 propellant					19,232	
CARGO: MOC-A				74,632		
MOC-A Interstage					2,000	
MOC-A Prop Sys/Stages					9,117	Ox tank-4.52m dia; H tank-6.3m dia; Engine-1m; Nozzel- 2.5m
MOC-A FPR Prop (3%)					1,823	
MOC-A Unusable Prop (1.5%)					912	
MOC-A Propellant					60,780	
CREW: MOO2				25,335		
Total CREW:MOO2 propellant (+FPR,Unusable prop.)					20,388	
CREW:MOO2 Prop Sys+Interstage					4,087	Ox tank-3.14m dia; H tank-4.38m dia; Engine-1m; Nozzel- 2.5m
CREW: MOO2 RCS					860	
PhEV				9,794	9,794	DIA.-11m/ LEN.-4m
4. CARGO: TMI-A	HLV-3	Oct '00	207,803			
TMI-A interstage					3,000	
TMI-A prop sys					32,472	Ox tank-6.54m dia; H tank-9.14m dia; Engine-2.5m; Nozzel- 18m
ETM MCC prop					2,620	
ETM RCS prop					2,660	
TMI-A FPR prop					6,334	
TMI-A Unusable prop					2,375	
TMI-A propellant					158,343	
5. CARGO: TMI-B	HLV-4	Dec '00	207,606			
TMI-B interstage					3,000	
TMI-B prop sys					29,821	Ox tank-6.54m dia; H tank-9.14m dia; Engine-2.5m; Nozzel- 18m
TMI-B FPR prop					6,627	
TMI-B Unusable prop					2,485	
TMI-B propellant					165,673	
6. INSPECTION CREW & SUPPLIES	STS-2	Feb '01	11,388			
Inspection Crew #1 (3) [Assembly Crew #1 (6) returns]				225		
Consumables				3,000		
1 fully loaded OMV				8,163		

Table 8. (Continued)

Items (Payload)	Launch#	Date	Mass	Mass	Mass	Dimensions
7. ASSEMBLY CREW & SUPPLIES	STS-3	Dec '01	14,613			
Assembly Crew #2 (6) [Inspection Crew #1 (3) returns]				450		
Consumables				6,000		
1 fully loaded OMV				8,163		
8. CREW: ECCV, ETV(remainder), M001	HLV-5	Dec '01	204,289			
CREW: MOC-A						
CREW: ECCV				6,870		
Crew+returnables+consum+suits					610	
Inert Module					5,360	DIA.-4m/ LEN.-3m
Earth entry Ab					600	
EELS					300	
CREW: ETV(remainder)				6,995		
Crew Consumables					2,730	
Spacesuits					420	
Solar/SPE monit. (ISE)					400	
Astro/Planetary (ISE)					100	
NTE MCC/RCS prop sys					1,435	
NTE MCC prop					950	
NTE RCS prop					960	
CREW: M001				15,677		
Crew Consumables					660	
M001 Interstage					300	
M001 Prop Sys/Stages					2,491	Ox tank-3.14m dia; H tank-4.38m dia; Engine-1m; Nozzel- 2.5m
M001 RCS prop.					860	
M001 FPR prop					326	
M001 Unusable prop					163	
M001 propellant					10,876	
CREW: MOC-A				174,747		
MOC-A Interstage					2,000	
MOC-A Prop Sys/Stages					21,684	Ox tank-6.54m dia; H tank-9.14m dia; Engine-2.5m; Nozzel- 2.5m
MOC-A FPR Prop (3%)					4,337	
MOC-A Unusable Prop (1.5%)					2,168	
MOC-A Propellant					144,559	
9. CREW: TMI-A	HLV-6	Feb '02	225,699			
Crew Consumables					6320	
TMI-A Interstage					3,000	
TMI-A Prop Sys/Stages					37,587	Ox tank-6.54m dia; H tank-9.14m dia; Engine-2.5m; Nozzel- 18m
TMI-A FPR Prop (3%)					6,317	
TMI-A Unusable Prop (1.5%)					2,369	
TMI-A Propellant					157,926	
NTE MCC prop					6,040	
NTE RCS prop					6,140	

Table 8. (Continued)

Items (Payload)	Launch#	Date	Mass	Mass	Mass	Dimensions
10. CREW: TMI-B	HLV-7	Apr '02	225,651			
TMI-B Interstage					3,000	
TMI-B Prop Sys/Stages					32,451	Ox tank-6.54m dia; H tank-9.14m dia; Engine-2.5m; Nozzel- 18m
TMI-B FPR Prop (3%)					7,211	
TMI-B Unusable Prop (1.5%)					2,704	
TMI-B Propellant					180,284	
11. CREW: TMI-C	HLV-8	Jun '02	225,442			
TMI-C Interstage					3,000	
TMI-C Prop Sys/Stages					27,576	Ox tank-6.54m dia; H tank-9.14m dia; Engine-2.5m; Nozzel- 18m
TMI-C FPR Prop (3%)					8,273	
TMI-C Unusable Prop (1.5%)					2,758	
TMI-C Propellant					183,837	
12. ASSEMBLY CREW & SUPPLIES	STS-4	Aug '02	14,613			
Assembly Crew #3 (6) [Assembly Crew #2 (6) returns]				450		
Consumables				6,000		
1 fully loaded OMV				8,163		
TOTAL			1,621,228			

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Figure 4. Schedule for Docked Vehicle

Launch Number	Launch Vehicle	Payload	2000										2001											2002								
			Mar	Apr	May	Jun	Jul	Aug	Sep	Oct	Nov	Dec	Jan	Feb	Mar	Apr	May	Jun	Jul	Aug	Sep	Oct	Nov	Dec	Jan	Feb	Mar	Apr	May	Jun	Jul	Aug
1	HLV-1	Crew:IMM+Ext Srvcs., M002 Cargo: M001				●																										
2	STS-1	Assembly Crew #1 Consumables and Equipment				●																										
3	HLV-2	Cargo: MOC-A						●																								
4	HLV-3	Cargo: TMI-A								●																						
5	HLV-4	Cargo: TMI-B										●																				
6	STS-2	Inspection Crew #1 Both Crews Return												● ▩																		
7	STS-3	Assembley Crew #2																						●								
8	HLV-5	Crew: ECCV, ETV(remainder), M001, MOC-A																						●								
9	HLV-6	Crew: TMI-A																								●						
10	HLV-7	Crew: TMI-B																										●				
11	HLV-8	Crew: TMI-C																												●		
12	STS-4	Inspection Crew #2 (loiter on Shuttle), Flight Crew																														● ▨

▩ Launch Cargo Mission from LEO

▨ Launch Human Mission from LEO

Figure 5 Proposed Heavy Lift Vehicles

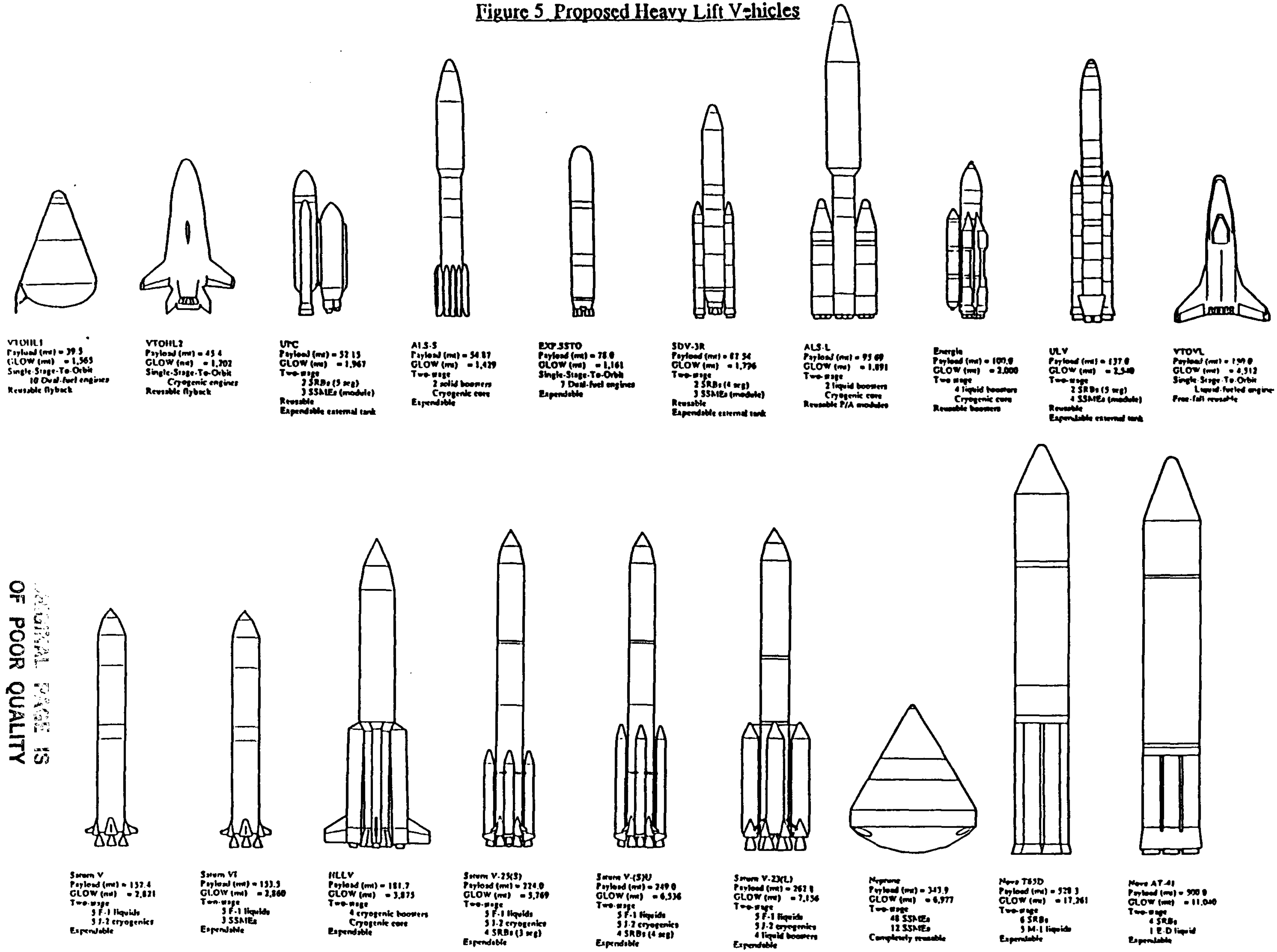

Figure 6. Comaprison of Different Mars Vehicle Configurations

Item	1 Piloted THI Stage Martin/SAIC Veh. Piloted	Cargo	1 Piloted THI Stage Eagle Vers. 1 Piloted	Cargo	5 Piloted THI Stage Eagle Vers. 2 Piloted	Cargo	4 Piloted THI Stage Eagle Vers. 3 Piloted	Cargo	3 Piloted THI Stage Eagle Vers. 4 Piloted	Cargo	2 Piloted THI Stage Eagle Vers. 5 Piloted	Cargo
Delta V Table												
THI (total)	4.352	3.739	4.352	3.739	4.352	3.739	4.352	3.739	4.352	3.739	4.352	3.739
THI-Stage E	-	-	-	-	0.579	-	-	-	-	-	-	-
THI-Stage D	-	-	-	-	0.687	-	0.741	-	-	-	-	-
THI-Stage C	-	-	-	-	0.840	0.906	0.921	-	1.043	-	-	-
THI-Stage B	-	-	-	-	1.091	1.190	1.231	2.280	1.402	1.462	1.662	1.463
THI-Stage A	-	-	-	-	1.155	1.643	1.459	1.459	1.907	2.277	2.690	2.276
MOC (total)	3.931	1.585	3.931	1.585	3.931	1.585	3.931	1.585	3.931	1.585	3.931	1.585
MOC-Stage B	-	-	-	-	1.467	-	1.467	-	-	-	-	-
MOC-Stage A	-	-	-	-	2.464	-	2.464	-	-	-	-	-
MOO1	0.719	0.719	0.719	0.719	0.719	0.719	0.719	0.719	0.719	0.719	0.719	0.719
MOO2	0.619	-	0.619	-	0.619	-	0.619	-	0.619	-	0.619	-
TEI	2.331	-	2.331	-	2.331	-	2.331	-	2.331	-	2.331	-
EOC	0	-	0	-	0	-	0	-	0	-	0	-
Weight Statement/Propellant Calculations												
Earth Crew Capture Vehicle												
Subsequent Stage Mass (none)	0		0		0		0		0		0	
Fixed Mass												
Crew+returns+consum+suits	610		610		610		610		610		610	
Inert Module	5,360		5,360		5,360		5,360		5,360		5,360	
Earth Entry Ab	600		600		600		600		600		600	
Other EELS	300		300		300		300		300		300	
Prop. Sys. Mass	0		0		0		0		0		0	
FPR Prop.	0		0		0		0		0		0	
Unusable Prop.	0		0		0		0		0		0	
Mass after burn	6,870		6,870		6,870		6,870		6,870		6,870	
Burned propellant mass	0		0		0		0		0		0	
Mass before burn	6,870		6,870		6,870		6,870		6,870		6,870	
Total Stage Mass	6,870		6,870		6,870		6,870		6,870		6,870	
Burn Isp (Kgf-s/Kgm)	330		330		330		330		330		330	
Mass Ratio	1.00		1.00		1.00		1.00		1.00		1.00	
T/W (Kgf/Kgm)	0		0		0		0		0		0	
Thrust (kN)	0		0		0		0		0		0	
Burn Time (min)	0		0		0		0		0		0	

Figure 6. (Continued)

Item	1 Piloted TMI Stage Martin/SAIC Veh. Piloted	Cargo	1 Piloted TMI Stage Eagle Vers. 1 Piloted	Cargo	5 Piloted TMI Stage Eagle Vers. 2 Piloted	Cargo	4 Piloted TMI Stage Eagle Vers. 3 Piloted	Cargo	3 Piloted TMI Stage Eagle Vers. 4 Piloted	Cargo	2 Piloted TMI Stage Eagle Vers. 5 Piloted	Cargo
Earth Transfer Vehicle												
Subsequent Stage Mass (ECCV)	6,870		6,870		6,870		6,870		6,870		6,870	
Fixed Mass												
Interplanterary Manned Module (IMM)	43,250		43,250		43,250		43,250		43,250		43,250	
External Services (EPS,TCS & Comm)	1,050		1,050		1,050		1,050		1,050		1,050	
Crew Consumables	2,730		2,730		2,730		2,730		2,730		2,730	
Spacesuits	420		420		420		420		420		420	
Solar/SPE monit.(ISE)	400		400		400		400		400		400	
Astro/Planetary (ISE)	100		100		100		100		100		100	
Interstage	-		-		2,000		2,000		2,000		2,000	
Prop. Sys. Dropped post-TEI (15%)	8,880		8,880		7,085		7,085		7,085		7,085	
Prop. Sys. Mass MCC/RCS (MCC 50%;RCC 100%)	2,340		2,340		1,435		1,435		1,435		1,435	
MCC prop.	950		950		950		950		950		950	
RCS MTE prop.	960		960		960		960		960		960	
FPR Prop. (4% TEI prop)	3,023		2,819		1,889		1,889		1,889		1,889	
Unusable Prop. (1.5% TEI prop)	-		725		709		709		709		709	
Mass after burn	70,973		71,494		69,848		69,848		69,848		69,848	
Propellant mass for TEI	48,872		48,347		47,234		47,234		47,234		47,234	
Mass before burn	119,845		119,841		117,082		117,082		117,082		117,082	
Total Stage Mass	52,200		52,200		51,295		51,295		51,295		51,295	
Burn Isp (Kgf-s/Kgm)	460		460		460		460		460		460	
Mass Ratio	1.69		1.68		1.68		1.68		1.68		1.68	
T/W (Kgf/Kgm)	0.179		0.179		0.183		0.183		0.183		0.183	
Thrust (kN)	210		210		210		210		210		210	
Burn Time (min)	17.50		17.31		16.92		16.92		16.92		16.92	
Mars Orbit Vehicle												
Stage MOO2 --------------												
Subsequent Stages Mass (ETV)	119,845		119,841		117,082		117,082		117,082		117,082	
Interstage	-		-		300		300		300		300	
PhEV	9,794		9,794		9,794		9,794		9,794		9,794	
Prop Sys Dropped post MOO2 (MOO2 20%-RCS 100%)	4,000		4,000		3,787		3,787		3,787		3,787	
RCS MOO2 prop.	860		860		860		860		860		860	
FPR Prop. (3%)	456		456		585		585		585		585	
Unusable Prop. (1.5%)	210		210		293		293		293		293	
Mass after burn	135,165		135,161		132,701		132,701		132,701		132,701	
Propellant mass for MOO2	19,873		19,872		19,510		19,510		19,510		19,510	
Mass before burn	155,038		155,033		152,211		152,211		152,211		152,211	
Total Stage Mass	25,399		25,398		*25,335*		*25,335*		*25,335*		*25,335*	
					********************* This stage is obtained from MCV *********************							
Burn Isp (Kgf-s/Kgm)	460		460		460		460		460		460	
Mass Ratio	1.15		1.15		1.15		1.15		1.15		1.15	
T/W (Kgf/Kgm)	0.138		0.138		0.141		0.141		0.141		0.141	
Thrust (kN)	210		210		210		210		210		210	

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Figure 6. (Continued)

Item	[illegible] Piloted THI Stage Marlin/SAIC Veh. Piloted	Cargo	1 Piloted THI Stage Eagle Vers. 1 Piloted	Cargo	5 Piloted THI Stage Eagle Vers. 2 Piloted	Cargo	4 Piloted THI Stage Eagle Vers. 3 Piloted	Cargo	3 Piloted THI Stage Eagle Vers. 4 Piloted	Cargo	2 Piloted THI Stage Eagle Vers. 5 Piloted	Cargo
Stage MOO1 --------------												
Subsequent Stages Mass (MOO2 or ETV)	155,038	0	155,033	0	152,211	0	152,711	0	152,211	0	152,211	0
TEIS (Received from MCV)	(60,775)	60,775	(60,771)	60,771	(58,917)	58,917	(58,917)	58,917	(58,917)	58,917	(58,917)	58,917
MOO2 Prop & Stage Sys (Received from MCV)	-	-	-	-	(25,335)	25,335	(25,335)	25,335	(25,335)	25,335	(25,335)	25,335
PhEV (Received from MCV)	(9,794)	9,794	(9,794)	9,794	(9,794)	9,794	(9,794)	9,794	(9,794)	9,794	(9,794)	9,794
Structure	0	500	0	500	0	500	0	500	0	500	0	500
MOSE	0	150	0	150	0	150	0	150	0	150	0	150
Crew Consumables	660	0	660	0	660	0	660	0	660	0	660	0
Satellites	0	2,000	0	2,000	0	2,000	0	2,000	0	2,000	0	2,000
Teleoperated MRSR	0	7,000		7,000	0	7,000	0	7,000	0	7,000	0	7,000
Support Services	0	320	0	320	0	320	0	320	0	320	0	320
Solar/SPE monit.(ISE)	0	200	0	200	0	200	0	200	0	200	0	200
Astro/Planetary (ISE)	0	100	0	100	0	100	0	100	0	100	0	100
Interstage	-	-	-	-	300	300	300	300	300	300	300	300
Prop Sys Dropped post MOO1 (MOO1 15%-RCS 100%)	2,750	3,220	2,750	3,320	2,491	4,375	2,491	4,375	2,491	4,375	2,491	4,375
RCS MOO1 prop.	860	1,490	860	1,490	860	1,490	860	1,490	860	1,490	860	1,490
FPR Prop. (3%)	324	350	324	350	326	577	326	577	326	577	326	577
Unusable Prop. (1.5%)	185	100	185	100	163	288	163	288	163	288	163	288
Mass after burn	89,240	85,999	89,247	86,095	62,966	111,346	62,966	111,346	62,966	111,346	62,966	111,346
Propellant mass for MOO1	15,415	14,854	15,415	14,871	10,876	19,232	10,876	19,232	10,876	19,232	10,876	19,232
Mass before burn	104,663	100,853	104,662	100,966	73,842	130,579	73,842	130,579	73,842	130,579	73,842	130,579
Total Stage Mass	20,194	40,078	20,194	100,966	15,677	130,579	15,677	130,579	15,677	130,579	15,677	130,579
Burn Isp (Kgf-s/Kgm)	460	460	460	460	460	460	460	460	460	460	460	460
Mass Ratio	1.17	1.17	1.17	1.37	1.17	1.17	1.17	1.17	1.17	1.17	1.17	1.37
T/W (Kgf/Kgm)	0.205	0.212	0.205	0.212	0.290	0.164	0.290	0.164	0.290	0.164	0.290	0.164
Thrust (kN)	210	210	210	210	210	210	210	210	210	210	210	210
Burn Time (min)	5.52	5.32	5.52	5.33	3.89	6.89	3.89	6.89	3.89	6.89	3.89	6.89
Stage MOC-A -------------												
Subsequent Stages Mass (ETV)	104,663	100,853	104,662	100,966	73,842	130,579	73,842	130,579	73,842	130,579	73,842	130,579
Interstage	-	-	-	-	2,000	2,000	2,000	2,000	2,000	2,000	2,000	2,000
Prop Sys Dropped post MOC-A (15%)	33,900	8,510	33,900	8,510	9,626	9,117	9,627	9,117	21,684	9,117	21,684	9,117
FPR Prop. (3%)	13,006	1,860	9,745	1,776	1,925	1,823	1,925	1,823	4,337	1,823	4,337	1,823
Unusable Prop. (1.5%)			3,157	707	963	912	963	912	2,168	912	2,168	912
Mass after burn	151,569	111,223	151,464	111,959	88,356	144,431	88,356	144,431	104,031	144,431	104,031	144,431
Propellant mass for MOC-A	205,694	47,515	210,471	47,115	64,176	60,780	64,178	60,780	144,559	60,780	144,359	60,780
Mass before burn	357,263	158,738	361,935	159,074	152,532	205,211	152,535	205,211	248,589	205,211	248,589	205,211
Total Stage Mass	313,375	57,885	257,273	58,108	78,690	74,632	78,693	74,632	174,747	74,632	174,747	74,632
Burn Isp (Kgf-s/Kgm)	460	460	460	460	460	460	460	460	460	460	460	460
Mass Ratio	2.36	1.42	2.39	1.42	1.73	1.42	1.73	1.42	2.39	1.42	2.39	1.42
T/W (Kgf/Kgm)	0.060	0.135	0.059	0.135	0.401	0.298	0.401	0.199	0.410	0.199	0.410	0.199
Thrust (kN)	210	210	210	210	600	600	600	400	1,000	400	1,000	400
Burn Time (min)	73.67	17.02	75.38	16.87	8.04	7.62	8.04	11.43	10.87	11.43	10.87	11.43

Figure 6. (Continued)

Item	1 Piloted TMI Stage Martin/SAIC Veh.		1 Piloted TMI Stage Eagle Vers. 1		5 Piloted TMI Stage Eagle Vers. 2		4 Piloted TMI Stage Eagle Vers. 3		3 Piloted TMI Stage Eagle Vers. 4		2 Piloted TMI Stage Eagle Vers. 5	
	Piloted	Cargo	Piloted	Cargo	Piloted	Cargo	Piloted	Cargo	Piloted	Cargo	Piloted	Cargo
Stage MOC-B ------------												
Subsqnt Stages Mass (MOC-A)					152,532		152,535					
Interstage					2,000		2,000					
Prop. Sys. Dropped post MOC-B (15%)					9,627		9,626					
FPR Prop. (3%)					1,925		1,925					
Unusable Prop. (1.5%)					963		963					
Mass after burn					167,047		167,049					
Propellant mass for MOC-B					64,178		64,176					
Mass before burn					231,225		231,225					
Total Stage Mass					78,693		76,690					
Burn Isp (Kgf-s/Kgm)					460		460					
Mass Ratio					1.38		1.38					
T/W (Kgf/Kgm)					0.093		0.093					
Thrust (kN)					210		210					
Burn Time (min)					22.98		22.98					

Mars Transfer Vehicle

Item	Martin/SAIC Piloted	Martin/SAIC Cargo	Eagle 1 Piloted	Eagle 1 Cargo	Eagle 2 Piloted	Eagle 2 Cargo	Eagle 3 Piloted	Eagle 3 Cargo	Eagle 4 Piloted	Eagle 4 Cargo	Eagle 5 Piloted	Eagle 5 Cargo
Stage TMI-A ------------												
Subsqnt Stages Mass (MOV)	357,263	158,738	361,935	159,074	231,225	205,211	231,225	205,211	248,589	205,211	248,589	205,211
TMI-A Interstage	0	0	0	0	3,000	3,000	3,000	3,000	3,000	3,000	3,000	3,000
Prop Sys Dropped post TMI-A (TMI 10%;MCC 50%;RCS 100%)	122,820	40,530	122,820	40,530	23,176	22,092	27,076	32,537	37,587	32,472	56,457	32,456
Crew Consumables	6,320	0	6,320	0	6,320	0	6,320	0	6,320	0	6,320	0
MCC prop.	6,040	2,620	6,040	2,620	6,040	2,620	6,040	2,620	6,040	2,620	6,040	2,620
RCS ETM prop.	6,140	2,660	6,140	2,660	6,140	2,660	6,140	2,660	6,140	2,660	6,140	2,660
FPR prop. (4%)	54,902	15,484	37,875	15,484	3,115	4,027	4,159	6,348	6,317	6,334	10,510	6,330
Unusable Prop. (1.5%)			12,625		1,168	1,510	1,560	2,381	2,369	2,375	3,941	2,374
Mass after burn	553,485	220,032	553,755	220,368	280,183	241,120	286,319	254,757	316,362	254,672	340,998	254,652
Propellant mass for TMI-A	756,588	247,346	841,664	247,346	77,866	100,677	103,976	158,706	157,926	158,343	262,762	158,258
Mass before burn	1,310,073	467,378	1,395,419	467,714	358,049	341,798	390,295	413,463	474,288	413,014	603,760	412,910
Total Stage Mass	952,810	308,640	1,033,483	308,640	126,824	136,587	159,070	208,252	225,699	207,803	355,171	207,699
Burn Isp (Kgf-s/Kgm)	480	480	480	480	480	480	480	480	480	480	480	480
Mass Ratio	2.37	2.21	2.52	2.21	1.28	1.42	1.36	1.62	1.50	1.62	1.77	1.62
T/W (Kgf/Kgm)	0.188	0.527	0.176	0.526	0.688	0.720	0.631	0.595	0.519	0.596	0.408	0.596
Thrust (kN)	2,415	2,415	2,415	2,415	2,415	2,415	2,415	2,415	2,415	2,415	2,415	2,415
Burn Time (min)	24.59	8.04	27.35	8.04	2.53	3.27	3.38	5.16	5.13	5.15	8.54	5.14

Figure 6. (Continued)

Item	1 Piloted TMI Stage Martin/SAIC Veh. Piloted	Cargo	1 Piloted TMI Stage Eagle Vers. 1 Piloted	Cargo	5 Piloted TMI Stage Eagle Vers. 2 Piloted	Cargo	4 Piloted TMI Stage Eagle Vers. 3 Piloted	Cargo	3 Piloted TMI Stage Eagle Vers. 4 Piloted	Cargo	2 Piloted TMI Stage Eagle Vers. 5 Piloted	Cargo
Stage TMI-B ------------												
Subsqnt Stages Mass (TMI-A)					358,049	341,798	390,295	413,463	474,288	413,014	603,760	412,910
Prop. Sys. Dropped post TMI-B (18%)					18,053	19,138	22,748	29,762	32,451	29,821	51,329	29,835
TMI-B interstage					3,000	3,000	3,000	3,000	3,000	3,000	3,000	3,000
Crew Consumables					0	0	0	0	0	0	0	0
FPR prop. (4%)					4,012	4,253	5,055	6,614	7,211	6,627	11,406	6,630
Unusable Prop. (1.5%)					1,504	1,595	1,896	2,480	2,704	2,485	4,277	2,486
Mass after burn					384,617	369,784	422,993	455,319	519,654	454,947	673,773	454,861
Propellant mass for TMI-B					100,292	106,324	126,377	165,342	180,284	165,673	285,159	165,750
Mass before burn					484,910	476,107	549,370	620,660	699,939	620,620	958,932	620,611
Total Stage Mass					126,861	134,310	159,075	207,197	225,651	207,606	355,171	207,701
Burn Isp (Kgf-s/Kgm)					480	480	480	480	480	480	480	480
Mass Ratio					1.26	1.29	1.30	1.36	1.35	1.36	1.42	1.36
T/W (Kgf/Kgm)					0.508	0.517	0.348	0.397	0.352	0.397	0.257	0.397
Thrust (kN)					2,415	2,415	2,415	2,415	2,415	2,415	2,415	2,415
Burn Time (min)					3.26	3.46	4.11	5.37	5.86	5.38	9.27	5.39
Stage TMI-C ------------												
Subsqnt Stages Mass (TMI-B)					484,910	476,107	549,370		699,939			
Prop. Sys. dropped post TMI-C (18%)					17,997	19,256	22,656		27,576			
TMI-C interstage					3,000	3,000	3,000		3,000			
Crew Consumables					0	0	0		0			
FPR prop. (4%)					4,499	4,279	5,664		8,273			
Unusable Prop. (1.5%)					1,500	1,605	1,888		2,758			
Mass after burn					511,906	504,247	582,578		741,544			
Propellant mass for TMI-C					99,986	106,977	125,867		183,837			
Mass before burn					611,892	611,224	708,444		925,381			
Total Stage Mass					126,982	135,116	159,075		225,442			
Burn Isp (Kgf-s/Kgm)					480	480	480		480			
Mass Ratio					1.20	1.21	1.22		1.25			
T/W (Kgf/Kgm)					0.402	0.403	0.347		0.266			
Thrust (kN)					2,415	2,415	2,415		2,415			
Burn Time (min)					3.25	3.48	4.09		5.97			

Figure 6. (Continued)

Item	1 Piloted TMI Stage Martin/SAIC Veh. Piloted	1 Piloted TMI Stage Martin/SAIC Veh. Cargo	1 Piloted TMI Stage Eagle Vers. 1 Piloted	1 Piloted TMI Stage Eagle Vers. 1 Cargo	5 Piloted TMI Stage Eagle Vers. 2 Piloted	5 Piloted TMI Stage Eagle Vers. 2 Cargo	4 Piloted TMI Stage Eagle Vers. 3 Piloted	4 Piloted TMI Stage Eagle Vers. 3 Cargo	3 Piloted TMI Stage Eagle Vers. 4 Piloted	3 Piloted TMI Stage Eagle Vers. 4 Cargo	2 Piloted TMI Stage Eagle Vers. 5 Piloted	2 Piloted TMI Stage Eagle Vers. 5 Cargo
Stage TMI-D ------------												
Subsqnt Stages Mass (TMI-C)					611,892		708,444					
Prop. Sys. dropped post TMI-D (18%)					18,050		22,742					
TMI-D interstage					3,000		3,000					
Crew Consumables					0		0					
FPR prop. (4%)					4,011		5,054					
Unusable Prop. (1.5%)					1,504		1,895					
Mass after burn					638,457		741,135					
Propellant mass for TMI-D					100,278		126,343					
Mass before burn					738,735		867,478					
Total Stage Mass					126,843		159,033					
Burn Isp (Kgf-s/Kgm)					480		480					
Mass Ratio					1.16		1.17					
T/W (Kgf/Kgm)					0.333		0.284					
Thrust (kN)					2,415		2,415					
Burn Time (min)					3.26		4.11					
Stage TMI-E ------------												
Subsqnt Stages Mass (TMI-D)					738,735							
Prop. Sys. dropped post TMI-E (18%)					18,027							
TMI-E interstage					3,000							
Crew Consumables					0							
FPR prop. (4%)					4,006							
Unusable Prop. (1.5%)					1,502							
Mass after burn					765,271							
Propellant mass for TMI-E					100,152							
Mass before burn					865,423							
Total Stage Mass					126,688							
Burn Isp (Kgf-s/Kgm)					480							
Mass Ratio					1.13							
T/W (Kgf/Kgm)					0.284							
Thrust (kN)					2,415							
Burn Time (min)					3.25							

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MARS EXPLORATION STRATEGIES

PRECURSORS/PREREQUISITES TO HUMAN EXPLORATION OF MARS

A WORKSHOP REPORT

EXECUTIVE SUMMARY

INTRODUCTION

People will land on the surface of Mars in the future, possibly the near future, and explore the planet. To ensure safety and efficiency it is essential to consider what information must be in hand before human exploration of Mars takes place. The Office of Exploration (Code Z) is making this assessment.

Office of Exploration Process

The Office of Exploration is responsible for developing initiatives that will take human beings into deep space beyond low Earth orbit. In doing so, the scope and responsibilities of other Offices and programs within NASA must be considered and woven into any initiative proposed to provide a consistent, sustainable program that utilizes all of the required capabilities within the agency. Consequently, the Office of Exploration has adopted a process in which initiatives are developed and the scientific and operational requirements of varying degrees of importance necessary to carry out the intiatives are set out. The requirements are transmitted to the various program offices and it is then their responsibility to define specific approaches to acquirong the data necessary to satisfy the requirements set out. The plans, data, studies or missions that result are than incorporated into the exploration plans and scenarios considered by the Office of Exploration and are used to more specifically define options and plans.

The process of identifying piloted exploration initiatives has focused attention on the Moon and Mars. Studies of each involve three elements, namely science, resource exploitation, and sustained human presence. These three elements are variables in the scenarios developed and result in some cases in different requirements.

The Office of Exploration has developed a set of general requirements which apply to piloted missions irrespective of objective. In addition, the Office of Exploration has developed a set of requirements that must be considered in the exploration of Mars. Both the specific and general requirements are listed in section 5.0 of Prerequisite Requirements for Exploration Class Missions, version 1.0 (draft) Feb. 1988, produced by the Office of Exploration. These requirements, in the notation used, are as follows:

5.1 General Requirements

5.1.1 Develop an approach and the appropriate systems to predict and provide onboard detection to preclude the possibility of unanticipated crew exposure to solar flares.

5.1.2 Characterize the potential of surface materials and the surface environment to disable humans or machines through toxicity, chemical reactivity, or unusual physical properties.

5.1.3 Determine the characteristics of surface materials necessary to design advanced life support systems or resource extraction processes using indigenous materials.

5.1.4 Determine the physical and compositional characteristics of the surface materials that define the nature and type of operational procedures and tools (drill, etc.) to be used by crews on the surface.

5.1.5 Determine sub-surface and surface properties for candidate outpost or base sites necessary to perform habitat and structure construction and burial for radiation shielding.

5.1.6 Determine the presence of possible, but undemonstrated, natural hazards to human operation, such as excessive natural or induced radioactivity of surface materials on Mars.

5.2 Requirements Specific to Mars Exploration Scenarios

5.2.1 Determine the properties (composition, physical state) of Martian moons to enable selection/rejection of scenarios that include Phobos/Deimos resource utilization.

5.2.2 Define the atmospheric and surface environment in sufficient detail to enable the safe entry and landing of piloted spacecraft and the extended operation of surface systems.

5.2.3 Resolve issues dealing with the forward contamination of Mars and back contamination of Earth by martian materials.

5.4 Engineering Tests

Some missions can also serve as engineering tests or demonstrations for entry and landing, surface mobility systems, orbital rendevous, power systems (e.g., photovoltaic systems), and communications systems to be used by piloted missions.

PURPOSE AND RESULTS OF THE WORKSHOP

To more fully understand the requirements for the piloted exploration of Mars, a workshop was held at the California Institute of Technology on March 7-9, 1988, under the auspices of the Office of Exploration. Participants reviewed existing data and developed assessments of what must be done before human exploration of Mars is possible. The workshop was divided into five working groups. Each working group considered an aspect of the piloted exploration of Mars and the following summary is divided into sections reflecting the responsibilities and conclusions of the five groups.

Mission Objectives:

The objectives of human exploration of Mars as defined by the workshop participants are:

- **to send people to the surface of Mars and return them safely to Earth**
- **to conduct scientific investigations uniquely enabled by a human presence**
- **to investigate the potential for sustained human activity on Mars**

The purpose of human exploration is to conduct scientific investigations which are uniquely enabled by the presence of human beings. This applies not only to activities on the surface of Mars and directly relevant to the planet but also to those investigations which are a result of the extended stay in space that such an enterprise entails. Scientific investigations that might be carried out on such missions were not defined. Viewpoints will change significantly as data from precursor missions accumulate. Humans on the surface may be best able to assess the potential of the planet (and of humans themselves) to support sustained human activity.

Exploration Strategies:

The workshop defined **Robust** and **Targeted** strategies as a discussion framework. These strategies have different longterm implications and spring from different philosophies of exploration but both have similar requirements for precursor data.

The **robust** lander strategy involves landing on a previously unexplored site. The site selection is based on an understanding of the planets characterisitics and processes derived from a series of orbital precursor missions plus extrapolation of surface characteristics based on returned samples from other locations on Mars. This strategy resembles that employed during the Apollo program, and humans replace automated devices for purposes of scientific investigations as opposed to largely robotic exploration. The mission(s) are self-contained and require no interaction with previously landed devices or materials. The scientific objectives are local reconnaissance and detailed studies with documented sample collection with limited mobility. The robust strategy does not necessarily have a long term evolutionary implication although it is not excluded.

The **targeted** strategy involves landing at a previously visited and well characterized site. The site will have navigation aids and environmental monitoring devices in place at a minimum and may also have additional components such as consumables, propellants, science support facilities, ascent and surface mobility vehicles etc. Site selection and preparation are based on characterization of the planet and the site by a series of robotic missions. The site will have been characterized in detail using robotic landers. Science and exploration objectives are focused. A fundamental difference between the two strategies is that the targeted strategy places a much greater reliance on robotic exploration. The targeted strategy allows exploration of more complex sites and the exploration could be focused so as to accelerate studies of the feasibility of human habitation. The targeted strategy fits into an evolutionary scenario leading to a permanent human presence on Mars, and probably would result in human exploration of Mars at a later date than is the case for the robust strategy. Detailed scenarios for each strategy have yet to be worked out.

An important aspect of the human exploration of Mars is the clear distinction that can be made relative to the Apollo program and the landings on the Moon. Mars has an atmosphere and its surface is much more active, both in terms of geological processes and chemistry than the moon. In addition, the extraordinary duration of the missions currently conceived results in hazards and requirements that did not come into play in the much shorter missions of the Apollo program.

Categories of Requirements for The Human Exploration of Mars:

Not all requirements for the human exploration of Mars have the same degree of importance and it is convenient to separate requirements into five categories which reflect their impact on various mission aspects. Each working group formulated requirements which fall into one or more of the following categories.

<u>Category 1. Go/No-Go decisions</u>:

Requirements influencing go/no-go decisions must be satisifed before a mission can be undertaken. Mission Rules generally fall into this category. The workshop concluded that there are such requirements in **life sciences, lander targeting, and system development and reliability** (Table 1).

Life sciences:

The physiological and psychological well-being of the crew members is of paramount concern during any mission. In the case of the human exploration of Mars, there are fundamental issues of health and safety which must be dealt with before a mission can be underaken. These arise from 1.) the **physiological effects of** extended stay times in 0-G gravity fields; 2.) the anticipated 3 year duration of the mission which imposes severe **psychological stress**; 3.) the long term cumulative effect of the interplanetary **radiation environment** (plus that imposed by shipboard devices), which has unknown consequences; and 4.) the **possible toxicity of Martian materials.**

The workshop participants recommend that it is essential to **1.) determine the feasibility of measures to counter the effects of long term microgravity exposure in the absence of an artificial G field; 2.) it is essential to determine the effectiveness and the utility of an artificial gravity field for long duration space flight** and to balance the merits of these two approaches against their effectiveness and cost.

Long duration spaceflight will impose severe psychological stresses that are not well enough understood. **It is recommended that means be devised for dealing with such stresses and that test procedures for predicting susceptibility to certain stresses must be developed.**

Radiation is a serious hazard and the workshop concluded that **the long term effects of continuous exposure to certain radiation levels are not well enough known to certify crew safety for a 3 year mission. It is recommended that on board means of detecting solar flares during the course of the mission, and of shielding the crew from both primary and secondary radiation must be developed and integrated into spacecraft design.**

The workshop concluded that it is possible that Martian materials may be toxic to humans. The composition of the Martian regolith is known only generally and the presence of small quantities of potentially toxic compounds cannot be ruled out. **The workshop recommends that the potential toxicity of Martian materials should be assessed by the collection and return of Martian samples.**

Lander Targeting

The targeted strategy requires a pinpoint landing capability to prepared sites. Landing accuracy must be satisfactorily within the mobility range of the lander, i.e., the crew with their equipment must be able to get to and utilize previously landed materials. A pinpoint landing capability minimizes the contingency mobility requirements which must be available to ensure access to the prepared site. This requirement is less severe for the robust mode, but safety considerations peculiar to a specific site may require a pinpoint landing capability. **The workshop recommends development and demonstration of a pinpoint landing capability.**

System Development and Reliability

Many new systems must be developed prior to piloted missions. Because crew members lives will depend on them, **the workshop participants recommended that systems reliability over long durations must be thoroughly proven.** Some examples of such systems include: spacecraft life support systems, technology for consumables recycling, reliable power and communications systems, low maintenance space suits, Martian surface habitats and life support systems.

Category 2. Spacecraft design; High Leverage Prerequisites

More detailed and precise data in some fields may significantly reduce mass and cost and increase reliability (Table 2). Information about the atmosphere and the chemical properties of the Martian regolith are examples.

Existing data indicate that aerocapture is a viable strategy for automated missions to Mars, but improved understanding of the atmosphere of Mars can lead to substantial improvements in spacecraft design resulting in mass savings and improved landing accuracy. The workshop participants concluded that there is **(1) a lack of comprehensive space-time coverage of the atmosphere density profiles (2) a lack of wind data, particularly for the near surface boundary layer, and (3) a lack of knowledge of atmospheric precipitates, aerosols and electricity. The workshop recommends that it is important to improve the data sets in these areas through such missions as the Mars Observer mission which will provide substantial improvements in the atmosphere data set.**

Mars is an active planet. It is possible that the chemical activity of the soil and/or its physical properties and the radiation environment may have unwanted effects on materials and equipment necessary for exploration of the planet. **The workshop concluded that proper materials selection requires an assessment of this possibility and recommends exposure of materials on the Martian surface for long periods of time.**

Category 3. Operational complexity/cost:

Some requirements have an impact on the complexity of operations or their cost and risk (Table 3).

Improved information can reduce design complexity and cost in terms of the operations on the surface of Mars involving human beings. For example, soil chemical reactivity and physical properties such as extremely *fine* grain size may *effect* EVA procedures and protocols. Mobility is a key question and requires more accurate assessment of rock/soil and slope distributions. Radiation hazards may require development of specialized habitats, equipment and warning devices. **The workshop recommends that it is important to: (1) develop improved data about rock/soil and slope distributions and meter scale surface morphology and (2) determine the physical/chemical properties of the Martian soil more precisely and (3) have real time solar flare radiation hazard warning.**

Category 4. Precursor data that enhance science/engineering content:

Certain fundamental data are required to properly plan and carry out a program of exploration (Table 4). Primary is a geologic map with characterizations of the nature of surface materials at a satisfactory level of detail (volcanics, impact-produced surfaces, aeloian features, ice and permafrost phenomena to name a few in the case of Mars). The creation of a map implies some knowledge of the relative ages of the units defined. Absolute calibration of geologic map units, however, significantly enhances the context of further investigations because age data allow rates of processes to be determined and in some cases may distinguish between processes. The workshop concluded that with age, process and process rate information in hand, complex sites can be selected with more confidence that the sites and the information obtained can be placed within a regional or global context. Precursor data which calibrates the geologic map is very important.

An important aspect of human exploration will be the assessment of the resource potential of Mars. Assessments are difficult even in the terrestrial case, but are made much simpler if the geologic context is understood and areas or units likely to contain resources are identifiable. Land forms in which ice played a role are an example.

The workshop recommends that it is important to provide global characterization and age calibration of the Martian surface, and that sample return is required in this respect. Resource

assessment/feasibility studies will be highly dependent on precise and complete geological/geochemical data and present data are inadequate. An improvement in geological/geochemical data is required to ensure optimum site selection and exploration gains.

Category 5. Regulatory requirements/Perception of risks:

Some aspects of planetary exploration are subject to national and international regulation and/or have an impact on public perception of risks associated with human missions (Table 5).

The workshop recommends that it is highly desirable to demonstrate that Martian materials are not hazardous biologically because it would lessen regulatory concern, and concern on the part of the public about potential back-contamination of the Earth. The workshop recommends establishing the nature of the organic chemistry of Martian materials as in aid to designing a quarantine protocol.

In addition, the workshop recommends that documention of the radiation environment in the vicinity of Mars by precursor missions would be an important way to remove the perception that unpredicted radiation risks may be present.

In addition to these requirements, human exploration of Mars will require the development of new technologies and further refinement of existing concepts and techniques. Some of these are directly related to safety and the proper functioning of a mission and must be thoroughly tested and verified before human flight is attempted. **Four technologies were identified by the workshop as requiring thorough testing before committment to piloted flight. They are:**

(1) aerocapture/aerobraking and pinpoint landing capabilities
(2) surface vehicular mobility
(3) terminal hazard avoidance
(4) automated rendezvous and docking.

The first three must be tested and demonstrated at Mars and on the Martian surface, respectively, through unmanned precursor missions. Demonstration of automated rendezvous and docking need not be done in the vicinity of Mars, however an end-to-end demonstration of vehicle systems from the launch from low earth orbit to recovery at earth, including automated rendezvous and docking in Mars orbit would increase confidence in the design of the human exploration missions through verification of the automated (backup) mode of operations.

Components of a Strategy to Prepare for Human Exploration of Mars:

A strategy that eliminates deficencies and strengthens weaknesses in the information necessary for human exploration of Mars would consist of a number of components. The workshop developed five components of such a strategy, namely, (1) research and analysis, (2) Earth orbital studies, (3) Mars orbital studies, (4) Mars surface activities, and (5) sample return.

Research and analysis:

Essential precursor research and analysis are possible in earth and space-based laboratories. Such studies include simulations of the conditions of long duration confinement and stress that crew members will encounter in the course of piloted exploration. **The workshop recommends that the potential of the Space Station should be examined for this purpose.** Similarly closed life support systems may be developed and tested in an Earth orbit enviroment.

There is a significant role for laboratory studies in developing essential precursor data. Remote sensing (radar, spectral reflectance studies, etc) of the Martian surface is a necessary means of providing corellative data for improved assessment of the surface properties of Mars which may come with the Mars Observer and Mars 94 missions. Successful extrapolation of new imagery data will require a more complete and refined remote sensing base. **The workshop recommends studies of the properties of Mars that may be measured by remote means.** Similarly, laboratory simulations of the interaction between the surface and the atmosphere (the boundary layer) would be important precursor studies.

Earth orbital studies:

As previously mentioned, the Space Station should provide an opportunity for important research and analysis and there are other important precursor studies that could be done in earth orbit. **The workshop recommends the demonstration of aerobraking/aerocapture techniques in earth orbit. In addition space adaptation/readaptation studies could be carried out in earth orbit and Space Station related modules could provide a means for testing fractional gravity devices.**

Mars orbital studies:

Precursor studies that could be carried out in Mars orbit include investigation of surface chemistry/morphology, aeronomy studies including the meteorology of Mars, investigations of the near-Mars radiation environment and technology demonstrations.

Mars surface activities:

Important precursor studies must be carried out on the Martian surface. **The workshop recommends the following:**

1. Local operational verification (mobility etc.)
2. Determination of detailed surface and subsurface properties (morphology, rocks, soil properties, accessibility of water etc.)
3. Materials compatibility tests
4. Emplacement of surface environmental monitoring stations
5. Emplacement of navigation aids.

Sample return:

Sample return is vital component in a strategy to satisfy piloted exploration precursor requirements. Sample return allows determination of soil toxicity and reactivity. Sample return provides data for resolution of forward and back contamination issues. Sample return allows testing for the availability of critical resources (nitrogen, phosphorous, water etc) in Martian materials. Sample return is a critical step in developing the deeper understanding of martian geological processes necessary to properly plan and carry out the human exploration of the planet. **The workshop recommends sample return as a vital step in the preparation for human exploration of Mars.**

APPENDIX I

WORKING GROUP REPORTS: MARS EXPLORATION STRATEGIES PRECURSORS/PREREQUISITES TO HUMAN EXPLORATION MISSIONS

Introduction

One of NASA's long term goals is to expand human presence in space beyond low earth orbit. Human exploration of Mars is a means of achieving this goal while simultaneously satisfying important scientific and engineering objectives.

A workshop to consider the precursor requirements for the human exploration of Mars was held at Caltech in Pasadena, California from March 7 through the 9th, 1988. The purpose of the workshop was to consider the requirements for missions and studies that would be necessary to properly plan and carry out the exploration of Mars by human beings. These requirements, once defined and ranked, will be provided to the Office of Space Science and Applications by the Ofice of Exploration as a guide to the implementation planning of precursor missions and studies. Major missions and programs advanced by OSSA will then be incorporated into the scenarios developed by the Office of Exploration as part of its recommended initiatives. The workshop considered requirements for only the first manned missions and that the missions would be of the conjunction class, implying a one year stay time on the surface of Mars.

The workshop was attended by 26 participants from various universities, industrial concerns and government agencies. (The names and affiliations of the attendees are listed in appendix II). The participants formed five working groups. Each group considered certain aspects of piloted missions or particular data classes as follows:

Working Group I. Mission strategies:

Two end-member strategies were defined at the outset of the workshop. The two strategies considered were (a) a "robust" strategy, defined as a piloted mission that would land at a predetermined site which had not been previously visited by a robotic mission and (b) a "targeted" strategy involving human landings at carefully prepared sites well characterized by earlier robotic missions.

Working Group II. Science objectives and requirements:

Science objectives and functions were defined for each of the above strategies. The workshop defined requirements that must be met for both strategies before human exploration can reach its full potential. In addition, the workshop identified strategies for demonstrating the resource potential of Mars for long term human habitation.

Working Group III. Atmosphere, atmospheric physics and the radiation environment:

The nature of the Martian atmosphere has a significant impact on spacecraft design and mission planning. A subgroup characterized existing data and the impact these data and their quality (or lack of it) have on key mission functions such as pinpoint landing, descent and ascent profiles and accuracies and spacecraft mass and volume characterisitics. Key precursor requirements relevent to specific knowledge of the atmosphere were defined.

The radiation environment, both in interplanetary space and on the surface is of prime concern in the case of piloted missions. The subgroup reviewed current data and identified requirements important for safe human flight.

Working Group IV. Surface properties and operations

A subgroup considered the surface properties of Mars (rock distribution, soil mechanics, slope profiles and distributions, and others) and their impact on functions necessary for proper, complete and safe exploration of the surface by humans. Existing data were considered and information required for effective and safe human activities on the surface was listed.

Working Group V. Life sciences:

The life sciences including physiological, psychological and human factors considerations, play a key role in piloted missions. A working group considered major areas of concern and defined key precursor requirements that must be met before long-term spaceflight such as that envisioned in the human exploration of Mars is possible.

Working Group Reports

Working Group I and II: Mission Objectives and Strategies:

Working group I developed a set of mission objectives that piloted missions should achieve. General objectives that are independent of strategy or mission types are:

1. To send people to the surface of Mars and return them safely to Earth.

2. To conduct scientific investigatons uniquely enabled by the presence of people.

3. To investigate the potential for sustained activity on the surface of Mars.

Investigations that are made possible by the presence of people may take place in transit, in orbit and on the surface. The definition and specification of experiments, observations and investigations to be carried out is beyond the scope of this report. The question of whether or not Mars has the capacity to support a long term base is open, and a fit subject for investigation through piloted missions. People living and working on the surface of Mars will be able to answer this fundamentally important question.

Working Group II defined two exploration strategies as means for achieving these general objectives. These strategies have different long-term implications and spring from different philosophies of exploration but both have similar requirements for precursor data. The two strategies are termed "robust" and "targeted" (Table 1).

In the robust strategy, piloted vehicles are landed at sites never before visited. The sites are selected based upon global and regional understanding of the processes and surfaces properties of the planet obtained through a series of precursor missions. The mission is self contained, it does not necessarily require rendezvous with previously landed vehicles, nor does it depend upon materials that may be inferred to be present by earlier surveys. The mission is exploratory with both regional reconnaissance and local detailed site survey objectives. The scientific objectives are local reconnaissance and detailed studies with documented sample collection with limited mobility. The robust strategy does not necessarily have a long term evolutionary implication, although it is not excluded.

The targeted strategy relies upon extensive site preparation by preceding missions. The landing site, having been characterized in detail, is prepared for follow-on piloted landings by installation of devices and materials which may include consumables, navigaton aids, tools and equipment, dwelling places and ascent and surface mobility vehicles etc. Targeted missions can explore more complex sites because fundamental site geological properties are known and provide a context for more elaborate investigations. The targeted strategy points towards verification of the potential of Mars (and of the site or sites selected) for long term support of people.

A fundamental difference between the two strategies is a much greater reliance on robotic exploration in the case of the targeted strategy. The targeted strategy allows exploration of more complex sites and the exploration can be focused so as to accelerate studies of the feasibility of human habitation. This implies the existence of an evolutionary strategy leading to the determination of whether of not it will be feasible to inhabit Mars, and probably would result in human exploration of Mars at a later date than is the case for the robust strategy.

The two strategies are compared as shown in Table 1. Detailed scenarios for each strategy have yet to be worked out.

An important aspect of the human exploration of Mars is the clear distinction that can be made relative to the Apollo program and the landings on the Moon. Mars has an atmosphere and its surface is much more active, both in terms of geological processes and chemistry than the Moon's surface. In addition, the extraordinary duration of the missions currently conceived results in hazards and requirements that did not come into play in the much shorter missions of the Apollo program.

A set of objectives and requirements for the two mission strategies devised are shown in Table 2. In terms of objectives, both strategies are functionally equivalent, although the robust strategy relies more heavily on the abilities of humans than does the targeted. The specific objectives shown in Table 2 are pointed towards determining site habitability.

Working Group III. The Atmosphere, Ionosphere, and Gravity

Major data classes considered by working group III were: 1.) atmospheric density and temperature profiles, 2.) winds, 3.) gravity, 4.) atmospheric composition/aerosols, and 5.) the plasma environment. Results are summarized in Table 3.

<u>Atmosphere density and temperature</u>

<u>Current Status</u>:

A Mars atmosphere specification document is being developed for the Mars Rover Sample Return Mission. This document will contain recent data relevant to engineering and design parameters for spacecraft in the Martian atmosphere. In addition COSPAR models define envelopes around the excursions of temperature and density measured by the Viking misssion.

Measurements of the atmosphere were made during Viking lander entry. Accelerometers on board the Viking landers measured atmospheric density from 120 km to 26 km. The precision of this measurement depends on accelerometer precision (0.02%) telemetry resolution (0.127 m/sec^2) and drag parameters, consequently a precision estimate is difficult to make. Atmospheric pressure and temperature on the aeroshell was measured from 90 to 6 km and from 27 to 6 km respectively. After jettison and deployment of the parachute, temperature and pressure were measured from 6 to 1.5 km and from 3.8 to 1.5 km respectively. These measurements have excellent consistency and provide a superior description of the northern summer atmosphere of Mars at two localities. These entry measurements provide atmospheric structural details that are averaged out in models describing average conditions. However details can be important. Viking temperature profiles show dramatic oscillatory structure from about 30 km to 120 km that may cause concern in the design of automated systems for aerobraking.

The atmosphere of Mars is highly variable. Surface pressure varies by as much as 30% because of volatile sublimation at the poles. Resonance as a result of solar heating causes a 15% pressure difference during the day. Surface heating and re-radiation to deep space causes a large diurnal atmospheric temperature range (190 K to 240 K during the summer) but the atmosphere has little variability during the winter when the temperature is approximately 150^{o}K. Global dust storms inject particulates into the atmosphere. The dust particles are very efficent energy absorbers and the day-night thermal structure of the atmosphere can change drastically during a dust storm. Dust storm heating can expand the atmospheric densities in the upper levels to values that exceed those of terrestrial re-entry regions.

Small scale wave structure occurs in the atmosphere at altitudes of 30-50 km where the drag is sufficient to achieve capture. The characteristics of this wave structure are not well known. Variations in density that result form the waves cause variations in the G forces (drag and lift) acting on descending spacecraft. These variable forces affect landing accuracy.

Assessment:

Existing data are adequate to show that aerocapture is a viable strategy for Mars entry. Successful descent of the Viking landers has provided a working climatological/meteorological model for lander descent and ascent operations. A primary limitation for more precise design decisions is the lack of comprehensive space-time data for density profiles. Mars Observer will provide useful new data in this respect. Mars Observer and the Mars 94 mission of the Soviet Union may provide information on the large scale variations of temperature below about 70 km with 5 km resolution. The methods to be employed will not resolve the smaller scale gravity wave structure in the atmosphere. Mars Observer will be sun synchronous but will be able to determine vertical profiles through the atmosphere 1.5 hours (about 23 degrees longitude) on each side of the synchronous position. Data therefore will be biased towards a particular time of day (or night).

Recommendations:

Mars Observer will considerably improve the data set concerning the atmosphere. However, Mars Observer will not measure small scale wave structure at altitudes (30-50 km) where drag allows capture. The selection of entry trajectory parameters (lift over drag ratio of the entry vehicle, steering etc) depends on density profiles and variations in real time in the atmosphere, primarily from 0 to 80 kms. Gravity wave structure in the atmosphere at 30 to 50 km altitude will have an effect on entering spacecraft which could translate into larger landing ellipse errors than would otherwise be the case. More comprehensive data such as could be obtained during a Mars Rover mission or a mission such as the proposed Mars Aeronomy Observer is highly desirable. In terms of maintaining orbital altitude and attitude, more data would be highly desirable. An argument could be made for a Mars Aeronomy Observer to provide such data, plus gravity wave data at lower altitudes, but the workshop did not reach a consensus on this point.

Wind direction and velocities

Current Status:

Wind observations exist for only the two Viking sites where wind gusts at 1.6 meters height sometimes exceeded 40 m/sec. Wind data for 1000 sols from the Viking 2 site were analyzed. A strong diurnal component occurs each day. The east-west zonal winds were estimated to exceed 120 m/sec above 20 km elevation. Global dust storms propagate at a meridional speed near 10 m/sec.

Assessment:

Wind data are limited. Although the density of the atmosphere is low, wind shears (and particulates) will be of concern during ascent. Wind data will not be acquired by Mars Observer.

Recommendation:

A surface meteorological network is recommended to define the properties and characterisitics of the atmosphere boundary layer.

Atmosphere composition/aerosols

The composition of the atmosphere was determined from measurements obtained by the mass spectrometers on the Viking Landers with a precision of one part in one thousand in general. Water vapor abundance was continuously mapped by the Viking Orbiters for 1.5 Mars years. Large amounts of water vapor have been observed over the northern polar region in summer, with little observed during winter.

More precise compositional data will allow more precise spacecraft design. To improve the predictions for aeroheating the following are needed: 1) thermal protective material catalytic reaction rates with Mars atmosphere species to +/- 50% precision, 2) non-equilibrium thermal model of reaction rates of species at high temperature in the atmosphere at to +/- 25% and 3) thermodynamic and transport properties of atmosphere species at high temperatures to +/- 15%.

Dust in the atmosphere probably consists of clay size particles that are likely to be silicates. Particle size estimates range from 0.4 to 10 microns. Studies of the optical depth versus wavelength at various phase angles during dust storms as measured by Viking orbiters allow estimates of the size and vertical distribution of dust particles to be made. Local dust storms are expected to be lower in height (15-20 km) but more dense by a factor of 2. The dust loading in dust devils is approximately 3 x 10^{-8} particles/cc or about 10^{-3} gm/cm^3. Atmospheric dust may render remote temperature sounding methods, such as Mars Observer techniques, unusable, and new approaches may need to be devised.

The frequency, location, composition and sizes of high altitude clouds are important to aerobraking vehicles that may be traveling at several kms per second as low as 20 to 30 kms above the surface. Clouds on Mars are formed of ice crystals. Particulate density in clouds are on the order of 10^{-8} gm/cm^3. Shadows indicate altitudes of up to 50 km with sizes of 100-200 km. High altitude clouds may be CO_2 ice.

Little is known about atmospheric electricity or its effects.

Assessment:

Dust loading in the atmosphere is not well understood in terms of density, particle size and vertical profiles. All measurements are for entire columns and the density at any given point in a column can vary greatly from the average. Particle sizes are inferred, there have been no direct measurements. The mechanisms which bring about conditions resulting in dust storms are not well understood. Forecasting requires near real time observations. Forecasts for even a few days in advance require hemispheric (and global with respect to major dust storms) observations taken at frequent intervals.

Recommendation:

Aerosols and dust in the atmosphere may influence spacecraft design and operation. Data are limited at present and additional measurements are highly desirable. Some in situ measurements are needed of the airborne dust and cloud particles together with remotely sensed spectra at various phase angles in order to place the in situ measurements into a global perspective. Reliable forecasting of atmospheric conditions requires real time data not now available. A meteorological network would provide such data.

Atmospheric electrical phenomena should be studied.

<u>Ionosphere/radio wave propagation</u>

<u>Current Status</u>:

The properties of the ionsphere are grossly known. However, there are questions concerning radio wave propagation. The particles and fields environment is not well known. No measurements have been made in low Mars orbits (115-1000 km).

<u>Assessment</u>:

The properties of the ionosphere do not constitute a major barrier to human exploration. The radiation environment in low Mars orbits (115-1000 km) is not well defined however.

<u>Recommendation</u>:

Properties of the ionosphere that are of interest from a requirements point of view may be studied adequately by the Mars 94 mission (but the Mars 94 mission will be at a higher orbit than that anticipated for a piloted mission).

<u>Gravity</u>

<u>Current Status</u>:

Gravity data were acquired by the Mariner and Viking missions. Gravity anomalies in the Martian crust may cause gradual changes in repetitive orbits but they will not effect aerocapture or aeroassisted landing accuracy.

<u>Assessment</u>:

The gravity field data are adequate for support of piloted missions. The data will be improved significantly by the Mars Observer mission.

Recommendation:

Current plans for refinement of the gravity field data during the course of the Mars Observer mission should be carried out.

Summary of group III Recommendations:

Prerequisites and precusor studies recommended by working group III are in category 2, high leverage requirements. (If it is assumed that current data are sufficient to allow an aerocapture mode during a piloted mission, and that forecasting the presence of a global dust storm at arrival at Mars will not be a requirement of the mission rules class, then there are no category 1 Go/No-Go requirements relative to the atmosphere).

Category 2: High leverage requirements

1. Existing data are adequate to show that aerocapture is a viable strategy at Mars. Successful descent of the two Viking landers has provided a working climatology/meteorological model for lander descent and ascent operations. However, further refinement of density/temperature profiles would allow refinement of design parameters. The Mars Observer mission will provide critical new data that will be important in planning for piloted missions.

2. Existing data are adequate to define the climatological extremes of temperature and pressure, but are not adequate for wind, relative humidity and electrical charging at the surface. Surface meteorological networks could provide this information. Forecasting requires adequate near and real time data not now available. A surface meteorological network is recommended to provide such data.

3. There is a lack of data concerning atmospheric condensates and electrical phenomena and these should be studied.

4. To improve the predictions for aeroheating the following data are needed: 1.) thermal protective material catalytic reaction rates with Mars atmosphere species to +/- 59% precision, 2.) non-equilibrium thermal model of reaction rates of species in the atmosphere at high temperature to +/- 25% and 3.) thermodynamic and transport properties of species on the Martian atmosphere at high temperature to +/- 15% precision.

5. Refinement of gravity data through the Mars Observer mission should be carried out.

Working Group IV. Surface physical properties:

Working group IV considered the surface properties of Mars including, topography-cartography, morphological-geological data, soil mechanics, chemical properties, the radiation environment (natural, cosmic ray, induced and solar) and near surface winds etc. The above classes of data are shown in Table 4 as a function of operational capabilities and/or functions required on the Martian surface. Table 4 includes brief comments concerning the adequacy of each of the data classes.

Topography-Cartography:

Current status:

Regional planimetric mapping is complete for the entire planet at a scale of 1:2,000,000. Topographic mapping is approximately 40% complete with 500 meter contours. A planet-wide geodetic control net has worse case errors of 5 km horizontally. Vertical errors are about 1 km in the equatorial regions and as bad as 3 km elsewhere. A digital image map is being compiled with Viking orbiter images and will have a resolution of 230 meters/pixel (allowing recogntion of an object of 5x230 meters). The digital map is tied to existing geodetic control. A digtal terrain model will be completed at about the same time and will have a spot size of about one km square and a vertical accuracy of about one km. Mars Observer altimeter data will significantly improve the topographic-cartigraphic data base.

Assessment:

Topographic and cartographic data are adequate for general planning purposes but are not adequate for site and/or traverse planning.

Recommendation:

Accurate topographic maps tied to a precise geodetic control net are essential for planning for the development of facilities to support long term habitation on Mars. Improved planimetry such as that provided by Mars Observer and possible subsequent missions will be required.

Geological data-Geologic maps:

Geologic maps form the basis for interpreting the evolution of the planet and for developing strategies to determine whether or not there are accessible and abundant resources. Maps are interpretations of Viking Orbiter and Mariner images which show the morphology and land forms of Mars on a several hundred meter length scale. The highest resolution data have 10 meter pixels (so a 50 meter object would be recognizable). Such data are extremely limited. Global mapping has been essentially completed. Systematic regional mapping using Mariner 9 data has been completed, and a few areas have been mapped using Viking data. The 1:2 million Viking photomosaics provide a data base for improved geologic mapping.

The geologic map divides the surface into geomorphic provinces which have been shaped by a specific process or processes. Important processes on Mars are aeolian, volcanic, fluvial, impact, tectonic and, at the poles, glacial. How these processes operate on Mars and how important they are is conjectural. The geomorphic provinces are placed in a time or stratigraphic framework. Time is measured by the crater frequency distributions on the Martian surface, but the time measurement is a relative one. Determining the age or ages of major time markers on the surface, such as one or more large impact events, would provide an absolute time calibration for the geologic map.

Data of all categories suggest that the surface of the equatorial regions is covered with dust-sized, iron-rich, palagonite-like weathering products. These deposits are cms or less to several tens of cm in thickness in those areas in which estimates are available from Viking infrared thermal mapper data and earth-based radar. This material in interspersed with blocks and possibly bedrock.

Assessment:

The geologic map data provide a basis for exploration and are adequate for identifying potential areas of interest. The geologic map, however, in the absence of ground truth, is not yet adequate for resource assessment, although it can be used to make sensible judgements as to the most likely places to look for resources. In addition, there has been no measurement of the timing of the events which have shaped the surface. The geologic map is not an adequate means for assessing terrain and safety factors.

Recommendation:

It is highly desirable to provide some sort of calibration of the geologic map and the assumptions concerning Martian processes upon which the map is based. Only after such testing, accomplished solely by sample return, will it be possibe to make judgements based on facts concerning the long term habitability of Mars. Consequently, sample return is a vital part of the precursor strategy leading up to the exploration of Mars by people.

Systematic mapping of the surface using the Viking 1:2 million photomosaic base should be completed. The Mars Geological Mapping program using 1:500,000 base maps is the best means of characterizing potential landing site geological properties, and should be continued

Rock, soil and slope distributions:

Current status:

Estimates of the distribution of rocks, "soil" (particulate material) and slopes have been made by Moore (USGS) and are included in the Mars Rover Sample Return Environmental Model document (Code IZ, Johnson Space Center). These estimates are based in Viking lander data, Viking infrared thermal mapper (IRTM) data, earth-based radar measurements and earth-based spectral reflectance data. Rock abundances on the surface are estimated by thermal infrared-spectral differencing techniques. Spatial resolution is not adequate with present data to discriminate between sites smaller than 100-200 km. Only the cumulative distribution is determined by these methods, not the rock size distribution. It is not possible, for example to distinguish between the effects of a few very large rocks or a large number of small ones. Extrapolations of data from the Viking lander sites are risky because the Viking sites are unusual in the relationship between their albedo and the observed rock populations. Consequently, on a global scale, cumulative rock distributions have errors of about a factor of two.

Slope distributions are estimated from Viking data and from earth-based radar. Analyses of the quasi-specular component of the polarized radar echoes yield "root-mean-square" (rms) slopes. The rms slopes are analogous to algebraic standard deviations of slope probability distributions. The rms slopes apply to slope lengths 1-1000 times the radar wavelength (theory) or about 250 times the radar wavelength (lunar experience). The rms slopes apply to that fraction of area not covered by surface to near surface roughness elements (usually rocks) that are 1/3 to 3 times the wavelength. For the ubiquitous, uniform lunar regolith with a relative dielectric constant near 3.0 at 13 cm wavelength, penetration of the radar waves is not large (~ 2-3 m) and correlation between surface roughness and radar roughness is good. This is probably the case for much of Mars where relative dielectric constants are 3.0 or larger. For some areas on Mars where relative dielectric constants are 2.0, or even less, penetration is much larger (~ 7 m and larger), and the correlation of surface roughness and radar roughness may not be as good as it was for the Moon..

Assessment:

The distributions of rocks and soil on the surface are not well enough known to support planning of exploration activities by people. Although walking and or rover-assisted traverses could be planned, there is no reasonable certainty that the region or area of interest would in fact be trafficable at a useful rate. Sensible judgements could be made if there were a basis for extrapolating the interpretations of Viking lander site data to different areas and regions at a greater resolution than is now possible. At present, however, there is no good basis for asssigning errors to such extrapolations other than to say that they are large.

Recommendation:

It is recommended that a means be found to provide new corroborative data for assessing present models relative to trafficability. High resolution images of the surface in areas for which remote sensing data currently exists or could be obtained is one method. Such images could be obtained by the balloon experiment of the Soviet Mars 94 mission.

Chemical properties of surface materials:

Current Status:

Viking data indicated that the Martian soil contains a chemically active oxidant/reactant component (or components). The upper limits for the abundances of these oxidants can be estimated by considering the most active samples. The three Viking biology experiments, each employing a different chemical methodology, imply abundances of the oxidant/reactant that range from a fraction of a ppb to a few tens of ppm. Even if several oxidants were present it is doubtful that their combined concentration could exceed one or two hundred ppm by weight. These trace quantities are unlikely to have any significant effect on materials. Although they may be corrosive, their abundances are too low to represent a serious hazard. This conclusion of course applies only to the materials in the upper few cms of the Martian surface at the Viking sites, although there is no good reason to suppose that such material is not representative in terms of oxidant/reactant concentrations.

Assessment:

There is at present no reason for supposing that oxidants/reactants in the Martian soil are a serious hazard. However, the data apply to the upper few cms of the regolith and the abundances of oxidants/reactants could be substantially higher at depth. In addition, the long term effects of the chemical environment represented by Martian materials on common materials used for equipment and machinery is not well known.

Recommendations:

Materials should be exposed at the surface of Mars in order to assess durability and reliability over a period of years.

Soil mechanics:

Current Status:

Mechanical properties of the Martian soil have been summarized by Moore (USGS) and are included in the Mars Rover Sample Return Environmental Model (1988). Five surface material classes are considered based on analyses of Viking lander data. The classes are weak, nominal, strong, sand and rock. Nominal material corresponds to the crusty to cloddy material at the Viking 2 site, whereas strong material corresponds to the blocky material at the Viking 1 site. Weak to strong materials probably are clay size (0.14 to 2 microns), but these materials may be cemented to form clumps. Sands probably are 100-300 microns in diameter based on saltation size in the Martian winds. Cohesions, angles of internal friction, bulk densities, thermal inertias and dielectric constants for all the classes are reasonably well in hand, although the dielectric constant is not well enough known to allow precise radar corrections. The angle of internal friction of drift material is less well known and some testing through simulations of the derived value would be useful.

Assessment:

Viking data plus radar data provide a solid basis for Martian soil mechanics estimates and these estimates are adequate for planning purposes. Drift material was tested as well as possible during the Viking experiment period. However, the test protocol could have UNDERestimated the angle of internal friction of drift material and, consequently, some testing using suitable simulants (i.e. very fine-grained kaolin powders and/or clodlets) at Mars atmospheric pressures is in order. Such testing could be carried out in suitably equipped terrestrial laboratories. If structures are to be built at a particular site, the properties of that site would have to be determined in detail including measurements at depth.

Terrains of unusual or uniqely Martian morphology may have unexpected properties in terms of soil mechanics.

Recommendations:

Angles of internal friction for drift material should be verified. Possible variations in soil mechanics in unusal terrains should be investigated. The grain size of wind blown material may extend to extremely small sizes. This material, presumably clays, could be detrimental to the operation of machinery because of its abrasive quality and ability to filter through very small openings. Designs should consider this problem.

Radiation environment:

Current Status:

Radiation on the Martian surface has three components (1) U, Th and K decay (2) cosmic ray radiation (including solar flares) and radiation induced by cosmic ray bombardment and (3) solar radiation with the UV flux of particular interest. The composition of Martian materials can be estimated well enough to make reasonable estimates of the contents of uranium, thorium and potassium at likely landing sites. Concentrations are unlikely to exceed those found in terrestrial granitic rocks. The cosmic ray flux at the surface and the resulting induced radiation can be calculated from existing data and calculations should be made so as to provide estimates of the long term flux. In addition, the effects of giant solar flares need to be considered. The UV flux similarly can be calculated. The long term effects of radiation and of the UV flux on materials exposed at the surface may be a matter of concern.

Assessment:

The radiation environment is, in principle, a known quantity. Long term effects given calculations of the fluxes in question on people and materials is less certain.

Recommendation:

Radiation should be monitored at the surface with both passive and active dosimeters to determine whether or not there is a potential for detrimental effects.

Summary of group IV recommendations:

Precursor requirements and studies concerning the surface properties of Mars as recommended by working group IV fall into categories 3 and 4.

Category 3: Prerequisites with operational implications

1. Topographic data and geodetic control data are inadequate for detailed planning. Improved geodetic control would be highly desirable.

2. Mobility is a key question. Current data are not good enough to determine whether or not humans could move about satisfactorily at any sites other than the Viking sites. A means of extrapolating interpretations from the Viking sites to areas of interest with trustworthy error statements is essential. This means high resolution imagery (submeter) of a areas and correlation with remote senseing measurements. These data must then be correlated with geologic data concerning land forms and land-forming processes to provide a data base for making mobility judgements.

3. Soil chemical reactivity and physical properties such as extremely fine grain size may effect crew health and safety by impacting, for example, ingress and egress procedures, materials selection, and system designs and operational limits.

4. Radiation is not likely to be a hazard, however giant solar flare particle events like the August 1972 event effect crew safety both in space and on the surface of Mars and a predictive/warning capability is required.

Category 4: Prerequisites which enhance science/engineering objectives

1. Global characterization of surface units establishes the context for exploration. Superior judgements can be made to guide human exploration efforts if the global characterization (the geologic map) of Mars is tested and refined by continued mapping efforts and by sample rerturn. For example:

   - calibration of the absolute ages of major units and stratigraphic markers allows discrimination between formations and in some cases processes and therefore optimizes intensive investigations during exploration by people. Better precursor data allows observers to investigate more complex sites (made unambiguous by improved precursor data) with confidence that major issues may be resolved.

   - improved interpretation allows superior assessments of the location and nature of resource-bearing units.

   - precursor data accelerates feasibility studies of the potential for long term human habitation by, for example, establishing the existence of water resources and providing proof of concept of extraction processes, or determining the availability of essential elements such as nitrogen and phosporous

These factors are dependent in large part upon sample return, and sample return, consequently, is an essential part of the preparation for further exploration of the planet.

Working Group V. Life Sciences:

Life science problems are of paramount concern during any piloted mission. A working group evaluated current data in the light of the extradordinary duration of the currently conceived piloted missions to Mars. The working group considered both physiological (e.g. microgravity effects, and radiation effects) and psychological problems. Aspects considered are summarized in Table 5.

Physiology/transit-orbital phases

Current status

Physiological adjustment to spaceflight is a complex process that is imperfectly understood. Problems range from temporary motion sickness to loss of mineral matter. Of great concern are physiological changes induced by a weightless environment. In the skeletal system, these include demineralization of the bones, kidney stone formation and effects on structural integrity. Loss of mineral mass from skeletal elements is a serious problem that effects structural integrity. Bed rest studies suggest that four hours of exercize per day are necessary to prevent a negative calcium balance. This is a severe penalty to impose. In addition, it is not clear that bones that have lost mineral matter will reform as they were prior to the loss. Repeated exposure to microgravity could accelerate mineral loss problems.

Kidney stone formation is favored in a weightless environment and appears to be related to bone demineralization.

Muscle problems involve atrophy and neuromuscular changes. Nerves appear to forget how to direct muscles. Skills may have to be relearned when exposed to gravity and time to relearn may not be available when descending to the surface of Mars or when on the surface. Cardiovascular deconditioning occurs and orthostatic intolerance develops

The immune system appears to behave differently in a weightless environment. How this difference affects immunity is unknown. Each individual's immune system is different from others and it is at present difficult to predict susceptibility to disease, although it clearly would be useful to have a predictive capability when selecting crew members.

There are in addition, problems associated with the neurovestibular system. Motion sickness includes visual disorientation, headache and sensory conflict within the autonomic nervous system. Adjustment to these difficulties takes a number of days depending upon individual characterisitics and may not be of long term concern. However, if an artificial gravity environment is employed, motion sickness in space could be a problem during transit phases of a long duration mission because there could be occasional or repeated exposure to a weightless environment.

Assessment:

The physiological effects of prolonged weightlessness are severe and not yet well enough understood to devise counter measures effective over the duration of a piloted mission to Mars. Counter measures must be developed before a mission is feasible. Counter measures may include artificial gravity, exercize therapy, drug therapy or some as yet unappreciated method. The immune system is effected by weightlessness, consequently individual crew members may react differently to the threat of disease. Predicting the susceptibility of crew members to disease is difficult and error prone at present but of clear importance in selecting crew members.

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Recommendations:

Methods to offset the effects of prolonged weightlessness are required before human exploration of Mars is feasible. Testing and simulation procedures could be developed for use on the Space Station. A Space Station facility, separable from the Space Station, and dedicated to the task of studying the effects of prolonged weightlessness would accelerate the process of solving these problems. In addition, better predictive criteria for forecasting crew member health must be developed.

Physiology/radiation:

Current Status:

People will be exposed to galactic (and solar) cosmic rays during piloted missions to Mars. The flux as a function of energy (and distance from the sun) is reasonably well known. The dose through thin shielding is about 10 rads at solar minimum and 4 rads at solar maximum for the galactic component. The major uncertainty is the secondary radiation produced by the interaction of the primary flux with spacecraft materials and the crew. Various models of radiation transport through spacecraft materials already exist but these models ignore secondaries and their effects or employ approximations without treating them explicitly. Over short periods the dose accumulated is acceptable but a journey to Mars would involve dose equivalents an order of magnitude higher than previously experienced. A conservative estimate of depth-dose equivalent during a years transit to Mars is 50 rem. This is the maximum allowable per year for crew members in space in routine missions. The NCRP recommends 50 rem per year as a guideline for exploratory missions and a potential for exceeding it would not preclude a piloted mission, but the dose estimate involved very conservative estimates of the secondary radiation.

Solar flares are a potential threat. The best protection at present is a space with special shielding that could be uitilized for 12 hours approximately through the peak of the flare particle flux. Means of forecasting flares and of detecting their beginnings from the earth as well as from the spacecraft are required.

Radiation on the Martian surface is a potential hazard. Radiation may derive from decay of indigenous nuclides and from nuclides produced by cosmic rays. The latter effects can be approximated by calculation, but the former must be measured.

Assessment:

Radiation is a serious hazard. Flux and radiation characteristics are known well enough to allow spacecraft designers to minimize the risk. The secondary flux is not well known, particularly for unspecified spacecraft designs and materials. Radiation on the surface can be approximated but the errors are likely to be large.

Recommendations:

1. Spacecraft design must consider the potential of the materials used in generating a secondary cosmic ray flux.
2. Increased information about transport, fragmentation and relative biological effectiveness of different charged particles would improve accuracy of the risk estimates.
3. Solar flare detection and warning is required.
4. Special shielding for giant solar flare events is required.
5. Active and passive dosimetry is required on the surface of mars in order to assess the radiation hazard insofar as humans are concerned.

Physiology/toxicity

Current Status:

There is a possibility that some Martian materials may be toxic. At present there is no definitive evidence of toxicity but the oxidant/reactants previously mentioned may include compounds that are toxic (but not corrosive to materials necessarily) even in trace amounts. Current data suggests abundance levels of a few hundred ppm or less, but species are unidentified. In addition, physical properties such as grain size may prove to be a hazard. Submicron particles for example may prove to be a concern.

Assessment:

Potential toxicity cannot be assessed in the absence of identification of compounds that occur in Martian materials and an assessment of their biological effects.

Recommendation:

Materials should be returned from the surface of mars. The mechanisms for sample return must ensure that some part of the returned material retains all of the characteristics and compounds present in the material's native state.

Psychological parameters

Current Status:

A piloted mission to Mars lasting several years will impose severe psychological stress. Effects on crew well being and performance will be significant. Experience with submarines, isolated stations in Antarctica, saturation diving chambers and other similar environments suggests that reactions will be variable and differing in severity between individuals. The effects of prolonged close confinement, exposure to hazard, prolonged weightlessness or low gravity, untested environmental designs and life support factors are all relatively unknown.

Assessment:

Psychological parameters are of critical importance in long duration missions. Psychological evaluations of potential crew members will be important but methodologies and criteria have yet to be worked out. An experience base exists but has not been sufficently studied.

Recommendation:

Studies of psychological stress in environments similar to that which will be experienced by people on piloted missions to Mars should be carried out. In addition, simulations, perhaps in a Space Station environment, appear to be an essential means for developing an appreciation of the psychological stresses that may occur, and for developing methods for dealing with them. Better predictive and evaluation criteria can then be developed.

Summary of group V recommendations:

The requirements arising from physiological and psychological factors are stringent and fall into category 1. They must be satisfied before a piloted mission can be undertaken. The requirements and recommendations are as follows:

Catogory 1: Go/No-Go requirements

1. Measures to counter the effects of longterm exposure to weightlessness or to low gravity environments must be developed. Development necessarily includes experimentation and simulation in order to understand the effects and consequences of exposure.

2. Crew evaluation procedures must include more fully developed methods for assessing susceptibility of crew members to disease under a variety of conditions.

3. Radiation is a serious hazard. Space craft design and material selection must take into account the production of secondary particles by cosmic rays. Appropriate

shielding must be in place to protect the crew from giant solar flares. Solar flare detection, both earth-based and on board, is a necessity. Effects of long term exposure to the anticipated flux should be assessed.

4. Martian materials may be toxic. This possibility is impossible to assess without sample return from the surface. Because potentially toxic compounds may be unstable, some fraction of the samples returned must be maintained under Mars ambient conditions.

5. The psychological stresses associated with long duration spaceflight are not well understood. Methods for anticipating stress and psychological trauma and for dealing with them must be devised. Better means of predicting susceptibility to psychological stress must be developed.

6. Some, but not all, of the life science issues can be investigated and resolved using the Space Station and the Shuttle. Simulations and experimental procedures can be devised to provide critical data pertinent to both physiological and psychological questions. The potential of the Space Station in this respect is very important and is deserving of support.

REQUIREMENTS FOR THE HUMAN EXPLORATION OF MARS

Introduction

Human exploration of Mars will be extremely complex. To define the data necessary to support human exploration, it is useful to consider data classes or types that will be required during a piloted mission. The quality of the data can then be assessed and a determination made as to whether or not it measures up to the tasks to be accomplished and, if not, what needs to be done to improve the data to acceptable levels.

Two types of data classifications were generated for consideration. The first (table 1.) is a functional classification and correlates data classes with mission phases whereas the second (table 2.) lists data requirements by data groups and corresponds to the structure now in use by the Office of Exploration.

The workshop utilized these classification schemes and produced some variations from them. The variations consist of additions and deletions to the tables plus matrices that generally correlate function or operation with data class or type in more detail than previously considered.

INSTITUTIONAL AND PROGRAM MANAGEMENT REQUIREMENTS

The requirements in this section are divided into three major types as follows:

- AGENCY CAPABILITIES: includes requirements for capabilities the Agency must possess in order to effectively implement the development and operation of the programs defined by the NASA Office of Exploration

- ASSIGNMENT OF DEVELOPMENT RESPONSIBILITIES FOR MULTI-CENTER PROGRAMS: includes requirements on the organizational structure of NASA which are needed to enable the Agency to efficiently implement the programs defined by the NASA Office of Exploration

- PROGRAM MANAGEMENT METHODOLOGIES: includes requirements to be implemented as part of each program defined by the NASA Office of Exploration.

SECTION 1.0: AGENCY CAPABILITIES

(1.1) The Agency shall develop the (development and operational) requirements on each program undertaken by NASA in full consideration of the requirements of all (including Office of Exploration and non-Office of Exploration; operational and under development) other Agency programs. In other words, the system engineering function for all NASA programs shall be performed in a manner which optimizes (satisfies the goals of the programs for the lowest total (initial + lifecycle + user) cost within the specified constraints of each program) the total set of Agency programs rather than in a manner which optimizes each program individually.

(1.2) The Agency shall define a process which enables the requirements identified by one NASA Code or program office to be levied on another NASA Code or program in order to support the implementation of the "total program requirements" defined as a result of Requirement (1.1) above. This process must include a clear definition of the funding responsibilities associated with the implementation of interprogram requirements (eg, who is responsible for funding the implementation of a given requirement - the Code/program identifying the requirement or the Code/program responsible for implementing it).

(1.3) Given the requirements on an element, the Agency shall possess the capabilities (eg, system engineering which defines the requirements on each subsystem of the element, subsystem development, element integration (ie, integration of the developed subsystems into an element), interface documentation techniques for all project interfaces, requirements traceability techniques, performance measurement systems, and so forth) needed to completely develop the element. In other words, NASA shall possess all of the programmatic techniques, methodologies, and tools required to enable the Agency to act as a prime contractor for the development of any element defined by the NASA Office of Exploration.

SECTION 2.0: ASSIGNMENT OF DEVELOPMENT RESPONSIBILITIES FOR MULTI-CENTER PROGRAMS

(2.1) The organizational structure of the Agency shall be well enough defined and flexible enough to accommodate the development and operation of any type of system involved in the accomplishment of NASA programs in an efficient and unambiguous manner. For example:
- the "theme" of each NASA center (research, project development, subsystem development, operations, and so forth) should be clearly defined so that the assignment of program-related (development and operations) activities for any given program is clear and so that the assignment of program-related activities can be made in a manner which reinforces the development of each center in relation to its theme,
- the "themes" of all NASA centers, when taken together, must cover all of the areas required for the development and operation of any program undertaken by NASA,

- assignment of program-related responsibilities should be able to meet the needs of the program without requiring duplication of the efforts between centers (eg, building of facilities, such as test beds, at one center when these already exist at another, hiring of personnel with a particular skill by one center when personnel with the required skill are underutilized at another center, and so forth) or relocation of large numbers of personnel between NASA centers.

(2.2) Each layer of management involved in the accomplishment of a given NASA program shall either work directly for (not be matrixed to) the management layer directly above it (eg, project managers work directly for the program manager) or there shall be a written contract between a manager's supervisor and the next layer of program management for the activities required in support of the program (eg, the program manager shall have a written contract with the center director for whom the project manager works for the production of the program products for which the center is responsible; the center director may then delegate the production of these products to the project manager who works for him).

(2.3) All personnel working in the program or projects offices associated with a given program shall be badged to the NASA Code responsible for the development of the program regardless of at which center these personnel are located. The Code responsible for the program shall have a written contract with the director of each center for the support of the program/projects office personnel located at his center.

(2.4) No "extra" layers of management shall be imposed between the manager of the office responsible for the definition and/or integration of a system or element and the manager directly responsible for the implementation of the requirements generated by the office responsible for definition and/or integration. (For example, in order to perform its job, the project office for a particular element must generate requirements (including interface requirements) on each subsystem of the element. The project office shall then levy these requirements directly on the manager for the development of each subsystem. If several subsystems are located at a single center, there shall not be a "center manager" responsible for all of the subsystems at the center, for whom all the subsystem development managers work.)

SECTION 3.0: PROGRAM MANAGEMENT METHODOLOGIES

(3.1) The Program Office for any newly authorized program shall, within 60 days of its creation, produce a written Program Management Plan. This plan shall include, but not be limited to, the following items:

- designation of all government agents (eg, NASA field centers) involved in the implementation of the program
- a description of all of the major functions involved in the accomplishment of the program along with a flowchart showing the relationship between these functions
- a clear designation of accountability specifying the agent responsible for the accomplishment of each function; when two or more agents are involved in the accomplishment of a given function, the specific responsibilities of each agent shall be clearly designated; under no circumstances should any definition activity or decision be shown to be arrived at as a result of 'negotiation' between two or more agents - though negotiations may be conducted in support of a decision, each decision must be shown as the responsibility of a specific agent.
- a "map" showing how the assignment of activities to a given NASA center is in accordance with the "theme" specified for the center by the Agency (see Section 2.0)
- specification of the phases and milestones of the program
- a definition, for each phase and milestone, of the products to be produced; these definitions must be specific enough to be verifiable upon completion of the phase/milestone (for example, to specify that a particular phase will result in "definition of the configuration of a system" is ambiguous and insufficient; instead, the parameters which must be specified in order to define the configuration shall be designated in the Plan)

(3.2) The mission (what a system is required to do) and system design (description of the hardware of the system which, as a result of the system engineering function, has been determined to be the most efficient system capable of meeting the mission requirements within the constraints of the program) requirements for any given program shall be recorded in separate documents. The mission requirements shall be documented in the Program Requirements Document (PRD). The system design requirements shall be documented in the Program Definition Requirements Document (PDRD).

(3.3) There shall be clear traceability between all levels of the requirements of a given program. All program requirements shall be traceable back to the mission requirements for the program (these are the "top-level" requirements of the program).

(3.4) Commonality shall not be considered to be a goal of any program in and of itself. Instead, commonality shall be viewed as one method available for use in the accomplishment of the goals and objectives of any given program. For each program, these goals and objectives shall be clearly specified and the program shall be structured in the manner which best allows it to accomplish these goals and objectives. (The "optimum" level of commonality for the program should be achieved as a result of this structuring. Commonality should not be treated as an independent activity within the program).

DEFINITIONS

- **SYSTEM**: the total set of elements developed under a single program; in its broadest sense, the "system" includes both flight and ground elements.
- **ELEMENT**: refers to an essentially modular part of the total system in which the subsystems are relatively self-contained. Usually, though not always, this "modular, self-contained subsystems" property of an element is caused by the fact that the element functions as a separable, independent unit during some phase of the mission. The command module, the lunar module, the Saturn V booster, and the Solid Rocket Booster are all examples of elements. Although it does not separate during the mission, the main engines module of the space shuttle is considered to be an element since it does fit this modular, self-contained subsystems definition. Additionally, this element is treated as an independent unit during processing. Under the above definition of an element, the mannedcore portion of the space station would be considered to be a single element which is divided into several subelements for development and assembly purposes. (Each of the free-flying platforms of the space station program is also an element.)
- **SUBSYSTEM**: The total set of hardware within a single element responsible for the provision of a particular major function (eg, power, data, communications, lifesupport, and so forth); The complete set of subsystems for each element (or, the major functions associated with the subsystems of the element) are specified in the Project Management Plan of the Project Office responsible for the development of the element.

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16. Abstract

The Office of Exploration (OEXP) at NASA Headquarters has been tasked with defining and recommending alternatives for an early 1990's national decision on a focused program of human exploration of the solar system. The Mission Analysis and System Engineering (MASE) group, which is managed by the Exploration Studies Office at the Lyndon B. Johnson Space Center, is responsible for coordinating the technical studies necessary for accomplishing such a task. This technical report, produced by the MASE, describes the process that has been used to conduct exploration studies and discusses the missions that have been developed in a "case study" approach. The four case studies that were developed in FY1988 include:

1. Human Expedition to Phobos
2. Human Expeditions to Mars
3. Lunar Observatory
4. Lunar Outpost to Early Mars Evolution

The final outcome of this effort is a set of programmatic and technical conclusions and recommendations for the following year's work.

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