Space Transfer Concepts and Analysis for Exploration Missions

Contract NAS8-37857

Final Report
Phase 1

Boeing Defense & Space Group
Advanced Civil Space Systems
Huntsville, Alabama

ORIGINAL CONTAINS
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FOREWORD

The study entitled "Space Transfer Concepts and Analysis for Exploration Missions" (STCAEM) was performed by Boeing Aerospace and Electronics, Huntsville for the George C. Marshall Space Flight Center (MSFC). The Boeing program manager was Gordon Woodcock, and the MSFC Contracting Officer Technical Representatives were Phil Sumrall and Alan Adams. Subcontractors to Boeing on this contract included: Camus, Inc., Computer Aided Design Software, Incorporated, General Dynamics (Space Systems Division), GEO Compatible, Paul Hudson, Madison Research Corporation, RemTech Incorporated, and SRS Technologies. The task activities were led by P. Buddington, S. Doll, B. Donahue, R. Finley, E. Henshaw, J. McGhee, P. Ryan, and B. Sherwood, with technical support from M. Appleby, J. Burress, S. Capps, B. Cothran, M. Cupples, D. Eder, E. Fisher, M. Fouche, R. Fowler, S. LeDoux, J. Nordwall, P. Ramsey, N. Rao, T. Ruff, K. Stanley, R. Tanner, D. Thrasher, B. Tillotson, I. Vas, and B. Wallace.

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ABSTRACT

The current study is the first phase of a broad-scoped and systematic study of space transfer concepts for human Lunar and Mars missions. Following the support to NASA in its "90-Day Study," the current project focused on Earth-Mars space transportation systems utilizing vehicles with different propulsive systems. A total of seven such "architectures" were developed, which could be employed to meet mission requirements as determined by representative program goals of "minimum," full science" or "industrialization and settlement." System and subsystem concepts and analyses were evaluated for each of the major space vehicles including their trajectory, power, weight, major materials, radiation, safety, reliability and operations. Based upon the study alternatives, a program life cycle cost was estimated for each of the three space exploration activity levels. A broad range of trade studies was conducted, culminating in technology/system return-on-investment trades, leading to a set of recommended architectures and systems for the program goals investigated.

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NOMENCLATURE

ACRV	Advanced crew recovery vehicle
ACS	Attitude control system
AFE	Aerobrake Flight Experiment
A&I	Attachment and integration
Al	Aluminum
ALARA	As low as reasonably achievable
ALS	Advanced Launch System
ALSPE	Anomalously large solar proton event
am	Atomic mass (unit)
AR	Area ratio
ARGPER	Argument of perigee
ARS	Atmospheric revitalization system
art-g	Artificial gravity
asc	Ascent
ASE	Advanced space engine
AU	Astronomical Unit (=149.6 million km)
BIT	Built-in test
BITE	Built-in test equipment
BLAP	Boundary Layer Analysis Program
BFO	Blood-forming organs
C	Degrees Celsius
CAB	Cryogenic/aerobrake
CAD/CAM	Compter-aided design/computer-aided manufacturing
CAP	Cryogenic all-propulsive
C_d	Drag coefficient
CELSS	Closed Environmental Life Support System
CHC	Crew health care
CG	Center of gravity
C_L	Lift coefficient

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cm	Centimeter = 0.01 meter
c/m	Crew module
CM	Center of mass
c/o	Check out
C of F	Cost of facilities
conj	Conjunction
COSPAR	Committee on Space Research of the International Council of Scientific Unions
CO2	Carbon dioxide
Cryo	Cryogenic
C3	Hyperbolic excess velocity squared (in km^2/s^2)
d	Days
DDT&E	Design, development, testing, and evaluation
DE	Dose equivalent
deg	Degrees
delta V	Velocity change m/s
desc	Descent
DMS	Data management system
EA	Earth arrival
E arr	Earth arrival
Ec	Modulus of elasticity in compression
ECCV	Earth crew capture vehicle
ECWS	Element control work station
ECLSS	Environment control and life support system
EP	Electric propulsion
ESA	European Space Agency
e.s.o.	Engine start opportunity
ET	External Tank
ETO	Earth-to-orbit
EVA	Extra-vehicular activity
F_c	Circulation efficiency factor
FD&D	Fire Detection and Differentiation
F_{ew}	Life support weight factor

F_f	Specific floor count factor
F_{fa}	Specific floor area factor
F_i	Aerobrake integration factor
F_l	Specific length factor
F_n	Normalized spatial unit count factor
F_o	Path options factor
F_p	Useful perimeter factor
F_{pc}	Parts count factor
F_{pr}	Proximity convenience factor
F_{rp}	Plan aspect ratio factor
F_{rs}	Section aspect ratio factor
FSE	Flight support equipment
F_s	Vault factor
F_{ss}	Safe-haven split factor
F_u	Spatial unit number factor
F_v	Volume range factor
FY88	Fiscal Year 1988 (=October 1, 1987 to September 30, 1988. Similarly for other years)
g	Acceleration in Earth gravities (acceleration 9.80665m/s^2)
GCNR	Gas core nuclear rocket
GCR	Galactic cosmic rays
GEO	Geosynchronous Earth orbit
GN2	Gaseous nitrogen
GN&C	Guidance, navigation, and control
GPS	Global positioning system
Gy	Gray (SI unit of absorbed radiation energy = 10^4 erg/gm)
hab	Habitation
HD	High density
HEI	Human Exploration Initiative (obsolete for SEI)
HLLV	Heavy lift launch vehicle
hrs	Hours
hyg w	Hygiene water
HZE	High atomic number and energy particle
H2	Hydrogen

H_2O	Water
ICRP	International Commission on Radiation Protection
IMLEO	Initial mass in low Earth orbit
in.	Inches
inb	Inbound
IP&ED	Implementation plan and element description
IR&D	Independant research and development
Isp	Specific impulse (=thrust/mass flow rate)
ISRU	In-situ resource utilization
JEM	Japan Experiment Module (of SSF)
JSC	Johnson Space Center
K	Temperature in Kelvin units
keV	Thousand electron volt
kg	Kilograms
klb	Kilopounds (thousands of pounds. Conversion to SI units=4448 N/klb)
klbf	Kilopound force
km	Kilometers
km/sec	Kilometers per second
ksi	Kilopounds per square inch
L/D	Lift-to-drag ratio
LD	Low density
LDM	Long duration mission
LEO	Low Earth orbit
LET	Linear energy transfer
LEV	Lunar excursion vehicle
LEVCM	Lunar excursion vehicle crew module
Level II	Space Exploration Initiative project office, Johnson Space Center
LH2	Liquid hydrogen
LiOH	Lithium hydroxide
LLO	Low Lunar orbit
LM	Lunar module
LOR	Lunar orbit rendezvous

LOX	Liquid oxygen
LS	Lunar surface
LTV	Lunar transfer vehicle
LTVCM	Lunar transfer vehicle crew module
L2	Lagrange point 2. A point behind the Moon as seen from the Earth, which has the same orbital period as the moon.
m	Meters
Mars-GRAM	Mars global reference atmosphere model
MARSIN	Trajectory analysis and stagnation point heating for Mars
MASE	Mission analysis and systems engineering (same as Level II q.v.)
MAV	Mars ascent vehicle
M/C_DA	Ballistic coefficient (mass / drag coefficient times area)
MCRV	Modified crew recovery vehicle
m_e	Mass of electron
MEOP	Maximum expected operating pressure
MeV	Million electron volt
MEV	Mars excursion vehicle
MLI	Multi-layer insulation
mm	Millimeter (=0.001 meter)
MMH	Monomethylhydrazine
MMV	Manned Mars vehicle
MOC	Mars orbit capture
MOI	Mars orbit insertion
mod	Module
M&P	Materials and processes
MPS	Main propulsion system
MR	Mixture ratio
m/s	Meters per second
MSFC	Marshall Space Flight Center
Msi	Million pounds per square inch
mt	Metric ton (1000 kg)
MTBF	Mean time between failures
MTV	Mars transfer vehicle
MWe	Megawatts electric
m^3	Cubic Meters

N	Newton, Kilogram-meters per second squared
n/a	Not applicable
NASA	National Aeronautics and Space Administration
NCRP	National Council on Radiation Protection
NEP	Nuclear-electric propulsion
NERVA	Nuclear engine for rocket vehicle application
NSO	Nuclear safe orbit
NTR	Nuclear thermal rocket
N2O4	Nitrogen tetroxide
OSE	Orbital support equipment
OTIS	Optimal trajectories by implicit simulation program
outb	Outbound
O2	Oxygen
PBR	Particle bed reactor
Pc	Chamber pressure
PEEK	Polyether-ether ketone
PEGA	Powered Earth gravity assist
P/L	Payload
POTV	Personnel orbital transfer vehicle
pot w	Potable water
PPU	Power processing unit
prop	Propellant
psi	Pounds per square inch
PV	Photovoltaic
Q	Heat flux (Joules per square centimeter)
Q	Radiation quality factor
RAAN	Right ascension of ascending node
RCS	Reaction control system
Re	Reynolds number
RF	Radio frequency
RMLEO	Resupply mass in low Earth orbit

RPM	Revolutions per minute
RWA	Relative wind angle
R&D	Research and development
	Rendezvous and docking
s	Seconds
SAA	South Atlantic anomaly
SAIC	Science Applications International Corporation
sec	Seconds
SEI	Space Exploration Initiative
SEP	Solar-electric propulsion
SI	International system of units (metric system)
SiC	Silicon carbide
SMA	Semimajor axis
sol	Solar day (24.6 hours for Mars)
SPE	Soalr proton events
SRB	Solid rocket booster
SSF	Space Station Freedom
SSME	Space shuttle main engine
STCAEM	Space Transfer Concepts and Analysis for Exploration Missions
stg	Stage
surf	Surface
Sv	Sieviert (SI unit of dose equivalent = Gy x Q)
S1	Distance along aerobrake surface forward of the stagnation point
S2	Distance along aerobrake surface aft of the stagnation point
S3	Distance along aerobrake surface starboard of the stagnation point
t.	Metric tons (1000kg)
TBD	To be determined
Tc	Chamber temperature
TCS	Thermal control system
TEI	Trans-Earth injection
TEIS	Trans-Earth injection stage
t.f.	Tank weight factor
THC	Temperature and humidity control
TMI	Trans-Mars injection

TMIS	Trans-Mars injection stage
TPS	Thermal protection system
TT&C	Tracking, telemetry, and control
T/W	Thrust to weight ratio
UN-W/25Re	Uranium nitride - tungsten/25% rhenium reactor fuel
VAB	Vehicle assembly building
VCS	Vapor coolled shield
Vinf	Velocity at infinity
WBe_2C/B_4C	Tungsten beryllium cabide/boron cabide composite
WMS	Waste management system
W/O	Without
WP-01	Work package 1 (of SSF)
w/sq cm	Watts per square centimeter (should be Wcm^{-2})
Z	Atomic number
zero g	An unaccelerated frame of reference, free-fall

[order: numbers followed by greek letters]

100K	$\leq$100,000 particles per cubic meter larger than 0.5 micron in diameter
7*n*7	Where n=(0,2-6): Boeing Company jet transport model numbers
+e	Positive charge equal to charge on electron
-e	Charge on electron
Δ	Delta
σ	Standard deviation
μg	Microgravity

1. EXECUTIVE SUMMARY

1.1 Introduction

Contemporary investigations of human exploration missions began with the work of the National Commission on Space commonly known as the "Paine Commission Report" (Reference 1). Further studies within NASA were reported in the report titled "*Leadership and America's Future in Space*", commonly known as the "Sally Ride Report" (Reference 2). NASA formed the Office of Exploration, which conducted a series of "case studies" reported in fiscal year 1988 and 1989 (References 3 & 4). These case studies examined a range of issues and scenarios, including human missions to the Moon and Mars and to Mars' small moons. Certain conclusions were reached; one important to the present study was that human missions to Mars' moons were not recommended. Consequently, the present study did not investigate such missions.

As a result of these activities, the Marshall Space Flight Center (MSFC) prepared and issued the solicitation for the present study in early fiscal year 1989. The general objective was to perform a broad-scoped and systematic study of space transfer concepts for human lunar and Mars missions.

The current study commenced in August 1989. By this time, President Bush had announced his Space Exploration Initiative (SEI) at the occasion of the 20th anniversary of the Apollo Moon landing July 20, 1969. Accordingly, the first few months of the study effort were largely directed to supporting the NASA "*90-Day Study*" in response to the President's initiative (Reference 5).

As examination of the President's initiative evolved, considerable emphasis was devoted to "architectures". Architectures can, of course, be defined in many ways. In this study, we chose to focus on a hardware-oriented definition, in which a transportation "architecture" is defined in terms of a selected principal Earth-Mars transfer propulsion technology. The architectures are named for the principal Mars transfer propulsion technology, or mission profile, whichever is most significant. The architectures include all the space transfer

systems necessary to accomplish the missions, the mission profiles, operating modes, and support systems.

We chose four basic architectures: cryogenic/aerobraking, nuclear thermal rocket, nuclear electric, and solar electric propulsion transfer technologies. In addition, we identified three additional architectures derived from operational variations on the cryogenic/aerobraking architecture: L2-based/lunar oxygen, Mars direct, and a family of cycler profiles.

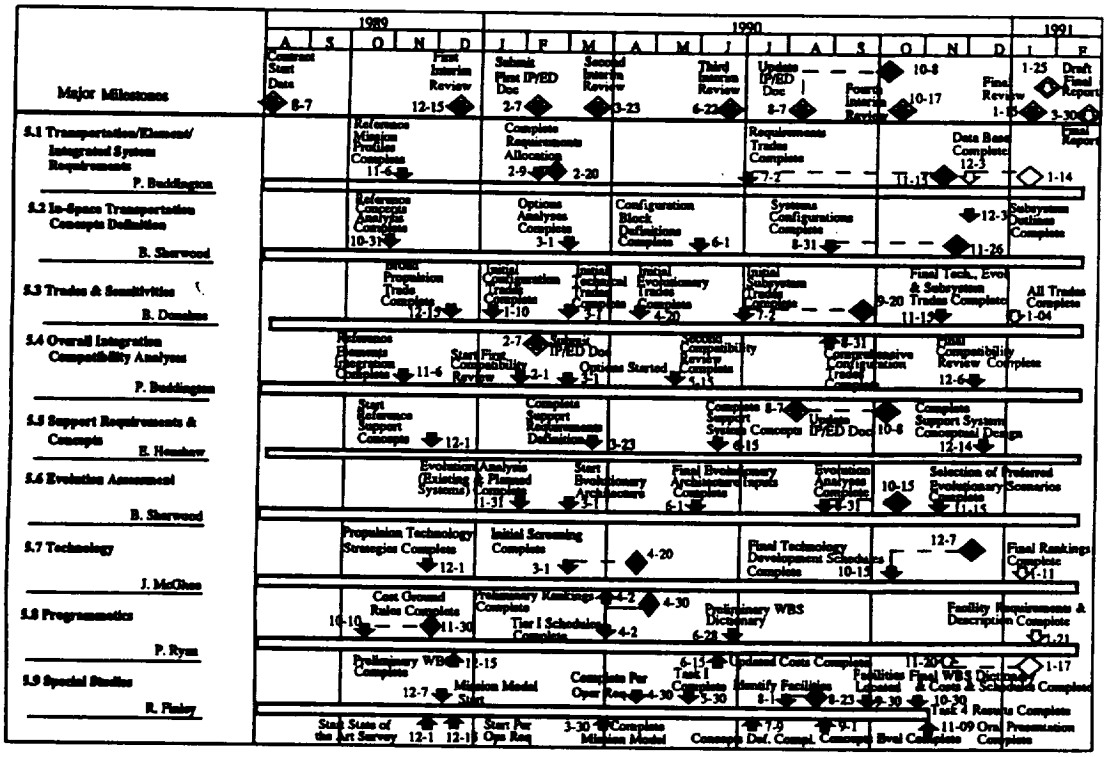
1.2 Study Description and Objectives

1.2.1 Description

The current report covering a period of 18 months, includes results of the eight basic study tasks described below and technical directives performed during this period. The study schedule with a list of tasks is shown in Figure 1-1. Further details of the study findings are provided in the Implement Plan and Element Description Documents (References 6-11) and the "Cost Data Volume" (Reference 12). This study was originally intended to give equal emphasis to lunar and Mars missions. When President Bush announced his Space Exploration Initiative (SEI), the study was directed to concentrate on Mars in-space transportation.

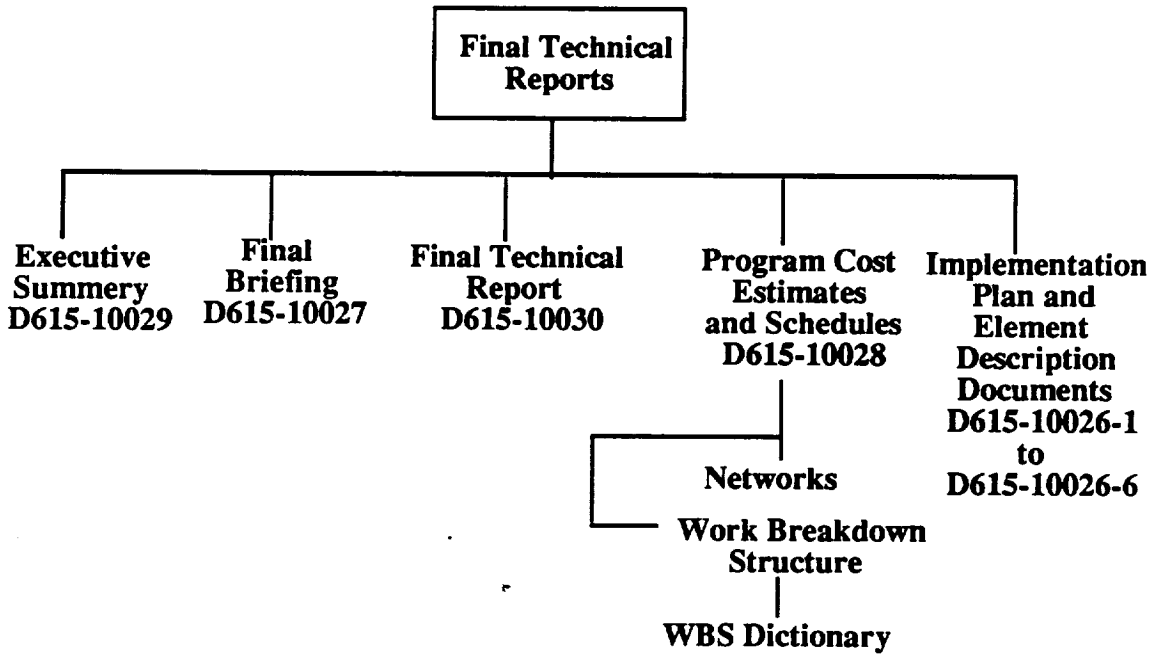
1.2.2 Guide to Study Documents

The results of this study have been documented in several contract-end reports, Figure 1-2. The three primary reports are: (1) the "Executive Summary", which summarizes the findings of the study; (2) the "Technical Report", which outlines the study activities including a discussion of the results; and (3) the "Cost Estimates and Schedules" document, which gives details of program and system costs with the accompanying schedules and life cycle costs to meet the proposed activity levels of the program. A work breakdown structure was also developed with an associated WBS dictionary.



SCHEDULE FOR SPACE TRANSFER CONCEPTS AND ANALYSIS FOR EXPLORATION MISSIONS

FIGURE 1-1



PHASE I STUDY FINAL REPORT DOCUMENTS

FIGURE 1-2

An additional set of documents called "Implementation Plan and Element Description" describe in detail the conceptual designs and analyses of space transfer system elements developed during the study. This provides additional backup for the results presented in the final technical report.

1.2.3 Objectives

The contract statement of work set forth nine major objectives for the study. These are presented here in italics. Events during the period of performance necessitated additions to some of the objectives, described in plain text.

a. *Assess and critique the NASA-provided mission model(s)/mission descriptions/scenarios, and from this develop in-space transportation system requirements for each scenario investigated.* Mission models were developed during the 90-Day Study. To meet other objectives of the study, we formulated additional scenarios.

b. *Define and assess in-space transportation concepts to the subsystems level, including the applications of advanced technology, for departure/return/vicinity, landing/ascent, aerobrake elements and associated support systems for postulated mission scenarios.*

c. *Define and assess habitability conceptual elements including common module derivatives.*

d. *Examine, characterize, and compare different alternatives. Assess element commonality between mission scenarios. Recommend and further define preferred concepts and develop them into integrated configurations.* This objective was broadened to include support for NASA Level II trade studies.

e. *Assess all operations associated with candidate concepts and missions to determine impacts, constraints, support requirements, and interfaces as they relate to the launch vehicles and launch or basing facilities.*

f. *Develop programmatic data, cost and schedules for the defined concepts, support equipment, and facilities.*

g. Conduct comprehensive program, performance, and system level trade studies. Perform sensitivity analysis of key design and program drivers and select the best overall concept(s). This objective was broadened to include architecture assessments as described below.

h. Perform overall integration compatibility analyses for transportation elements with launch vehicles and infrastructure.

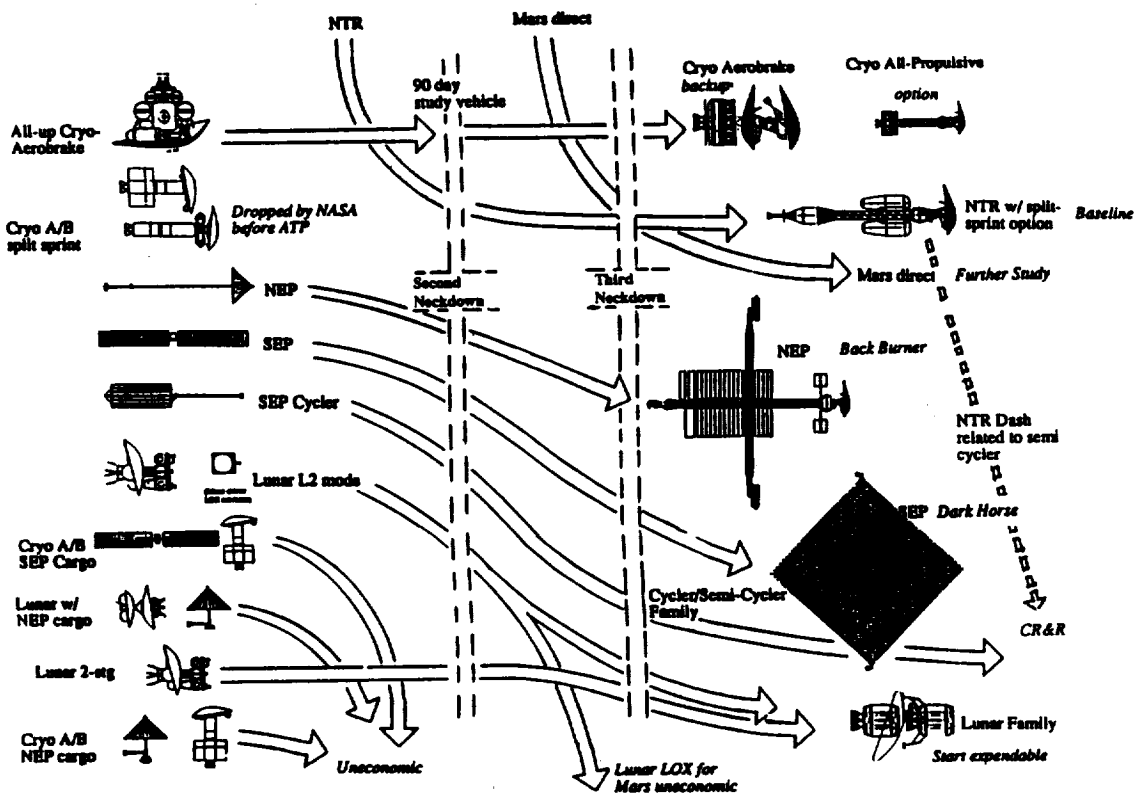
i. Identify and prioritize enabling and enhancing technology requirements for Advanced Development and Technology programs.

1.2.4 Overall Study Flow

The study flow, as required by MSFC's statement of work, began with a set of strawman concepts, introduced others as they were identified, conducted four "neckdowns", and concluded with a resulting set of concepts and associated recommendations. As the study progressed, much discussion among the SEI community centered on "architectures". In this study, architectures were more or less synonymous with concepts, since the statement of work required that each concept be fully developed including operations, support and technology.

An initial neckdown of a large number of potential candidates produced the ten concepts as shown in Figure 1-3. Combinations of major technologies, such as electric propulsion and aerocapture, were quickly determined to be uneconomic in view of high development costs. Further, we found that electric propulsion systems could perform both crew and cargo Mars missions if crews are transported to and from the electric system at about lunar distance by a lunar transfer vehicle. New systems architectures introduced during the study included nuclear thermal rocket (NTR) and Mars direct. NTR was introduced as an option by NASA during the 90-Day Study. We introduced the Mars direct profile (everything is landed on Mars; the return propulsion system is loaded with oxygen and perhaps fuel on Mars) in March 1989. Martin-Marietta subsequently developed and publicized one variant of this concept. Lunar oxygen for Mars missions was found uneconomic because of long payback time for the launch mass required to emplace lunar oxygen production on the Moon. Lunar oxygen has a reasonable return on investment for lunar transportation at two or more lunar trips per year. The cycler architecture was broadened to include semi-cyclers. Late in the study we introduced an NTR-dash mode (described later in this report)

closely related to the semi-cyclers. Current status of the remaining concepts is a result of the fourth "neckdown" as indicated by italics on the Figure.



CONCEPTS, ASSESSMENTS, AND EVOLUTION
FIGURE 1-3

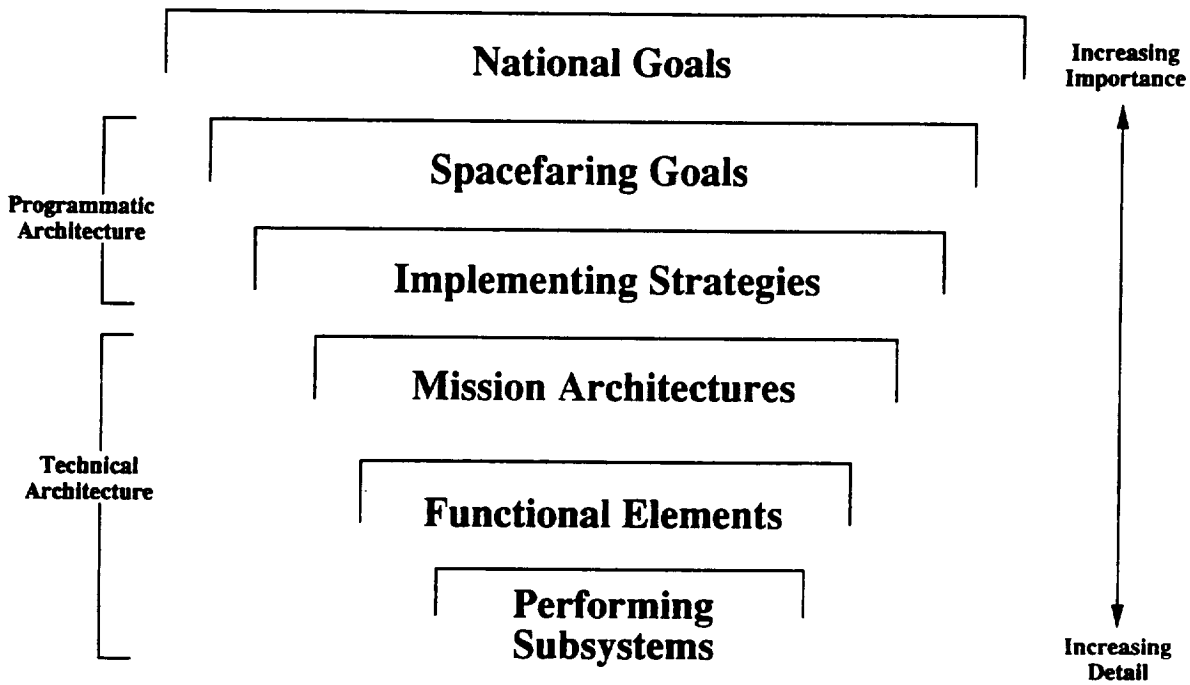
1.3 Architecture Analyses

1.3.1 Technical Approach to Architecture Analysis

Descriptions of space architectures are related to hierarchical information structures underlying modern approaches to data bases and structured software systems. A space architecture involves at least six hierarchical levels, beginning with national goals the space program aims to satisfy, and descending through lower levels to the systems and subsystems best suited to the national goals.

This study evaluated architectures in the hierarchical framework depicted in Figure 1-4. While the focus was on mission architectures from the point of view of space transfer systems, their elements and their subsystems, it was necessary to relate these to

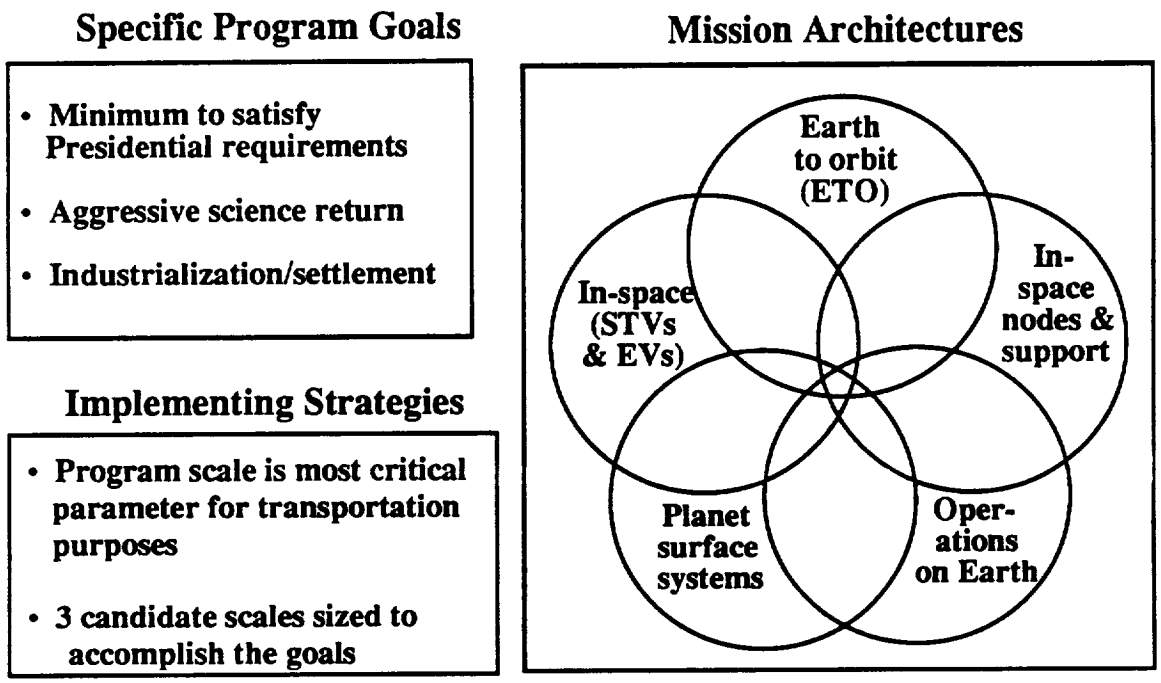
programmatic architecture factors. Exploration program goals have been spelled out qualitatively by President Bush: "Back to the Moon to stay, and then on to Mars". The quantitative level of activity at each destination is yet to be determined, as are the strategies (what are we going to accomplish and how). We considered a range of specific mission goals and associated activity levels, assumed certain strategies, and analyzed architectures by defining space transfer concepts, estimating subsystem performance, synthesizing complete in-space transportation architectures, and developing life cycle program scenarios. We did not analyze the relationships between objectives for a long-range space program and contribution to national goals. The range of mission goals considered by the study covers a broad span of space objectives.



A SPACE PROGRAM HIERARCHY
FIGURE 1-4

Program activity level (scale) is the most important factor in selecting space transportation architectures. Therefore, we defined three activity levels covering the probable range of interest for SEI, from a minimum program that barely meets the President's stated SEI objectives to an ambitions program leading to industrialization of the Moon and the beginnings of settlement of Mars.

Mission architectural issues are highly interrelated as suggested by the overlapping circles in Figure 1-5. We used available information on planet surface systems to set overall requirements for transportation systems, and performed preliminary node analyses, mainly to determine what is required for assembly of the in-space transportation systems and their payloads, and compared assembly on Space Station Freedom vs. off Space Station Freedom. We considered the effects of ETO architecture by examining a range of ETO launch capabilities and costs. We considered Earth-based operations only to the extent of estimating launch site facilities costs for heavy lift and payload processing.



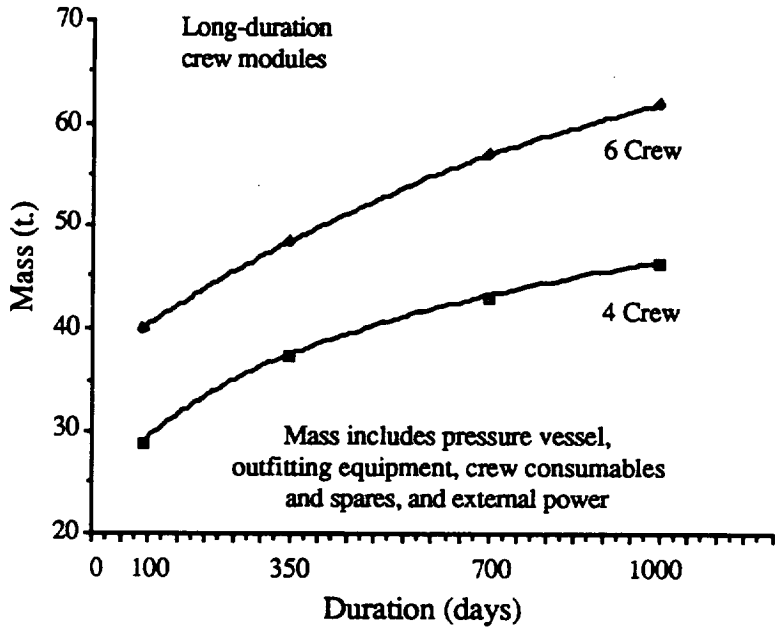
STUDY ARCHITECTURAL SCOPE
FIGURE 1-5

Mission rules were adopted as an element of program strategy. Further elements of strategy were use of reasonably conservative mass and performance estimates, hardware commonality, adequate time for technology advancement where needed, and options selected for minimum life-cycle cost commensurate with reasonable return on investment, and for reasonable technology risk. Return on investment analysis employed a range of complete program scenarios as described later.

1.3.2 Generic Requirements and Transportation Architecture Drivers

Space transportation is similar to other transportation systems in terms of architecture driver requirements, as summarized on Figure 1-6.

- Like any transportation system, space transportation is driven by:
  - How much?
  - How fast?
  - How far?
  - How often?
- For the Moon, cargo delivery up to 45 t., or crew of 6 with a few t.
- For Mars, cargo delivery up to 50 t., or crew of 6 and 20-30 t.
- Desire 2 - 3 landings per mission for early Mars trips.
- Massive transit hab for the long trip drives Mars in-space transportation.
- Mars mission delta V (how fast?) increases rapidly as trip time reduced.



TRANSPORTATION ARCHITECTURE DRIVERS
FIGURE 1-6

The 45-t. cargo delivery requirement for the Moon arises from a *campsite* concept. This is a fully integrated turnkey habitat system for early lunar operations; it enables lunar crews to stay up to 45-60 days on the Moon for construction and man-tending astrophysics observatories, and for construction of larger habitation facilities as required by the program scenarios.

The requirement for cargo delivered with crew was allowed to be flexible, tailored to transportation system capabilities. It is more important to match the lunar transport system to the HLLV than to achieve some particular cargo-with-crew capability. The 45-t. capability fits an LTV tandem-stage expendable mode, where the LTV is sized for a direct-

to-surface mode and later for LOR mode.

For Mars, 20-30 t. of cargo delivered with crew enables a campsite operation. (The campsite for Mars is less massive because early staytimes are shorter and less energy storage mass is needed.) A system that meets this requirement can deliver about 50 t. in a cargo mode. Since Mars opportunities are infrequent, there is benefit in multiple landings on early missions in order to reach earlier selection of an exploration base site.

Mars missions take months to years. The shortest missions presently discussed are about 200 days -- nuclear split sprints and gas-core nuclear rocket missions. Durations of 450 to 1000 days are more likely for early missions. We used historical spacecraft volume versus duration trends and Space Station Freedom ECLSS and subsystems data to develop crew module mass trends. Over the range 90 to 1000 days, the ratio of crew module mass is only about 1.5. While short trip times may be desirable, one cannot expect a dramatic reduction in crew module mass with shorter trip time.

It has often been presumed that a Mars mission can be carried out with as few as four people. We conducted a skill mix analysis, shown in Figure 1-7, concluding that the minimum Mars mission crew will be at least 6, but could possibly go to 8. This conclusion was driven by two factors: (1) skills per crew member were limited to 2 primary and 1 secondary, and (2) all critical skills were presumed to require backup.

Assumptions

- **Technology freeze in 2008 -- level 5 for phase B, 10 years prior to flight**
- **High probability of mission success (defined as *landing crew on Mars and returning them to Earth*)**
- **30 day minimum surface stay, with rover capability**
- **Skills per crew member: 2 primary and 1 secondary**
- **No back-up for primary science skills**
- **Interpersonal psychology critical for Mars-class durations**

T	A/D	S	Functions	Man	Machine
✓		✓	Medical	✓	
✓		✓	Dental	✓	
✓		✓	Electrical Maintenance	✓	✓
✓		✓	Mechanical Maintenance	✓	✓
✓			Science Enroute	✓	
		✓	Science on Surface	✓	
✓	✓	✓	Guidance		✓
✓	✓	✓	Navigation		✓
✓	✓	✓	Control		✓
✓	✓	✓	Communication	✓	✓
✓	✓	✓	Command	✓	
✓		✓	Housekeeping	✓	✓
T = Transfer A/D = Ascent/Descent S = Surface					

# Crew \ Skill	Primary 1	Primary 2	Secondary
1	Planetology	Exobiology	Dentist
2	Scientist	Physician	Resource Sc.
3	Physician	Dentist	Scientist
4	Elect. Maint.	Pilot	Computers
5	Mech. Maint.	Commander	Elect. Maint.
6	Elect. Maint.	Computers	Pilot

MINIMUM CREW SIZE ASSESSMENT FOR MARS MISSIONS

FIGURE 1-7

1.3.3 Candidate In-Space Transportation Architectures

Architectures were defined in terms of space transfer propulsion/technology: cryogenic/aerobraking; nuclear thermal rocket; nuclear electric propulsion, and solar electric propulsion. Additional ones were defined as operational modes where an operational mode is as important as a propulsion theme. For example, a cryogenic/aerobraking Mars transfer system operated from an L2 node, using lunar oxygen, was considered a unique architecture even though the technology is common with a cryogenic/aerobraking system operated from low Earth orbit (LEO). The additional architectures were: (1) a lunar L2 libration point-based system using lunar oxygen; (2) a "Mars-direct" mode that lands everything on Mars and uses in-situ propellants; and (3) four variations on the cycler concept described (Reference 1). Many more sub-variants are possible. We identified roughly 70 transportation hardware architectures considering all reasonable sub-variants. The seven architectures are summarized in Figure 1-8.

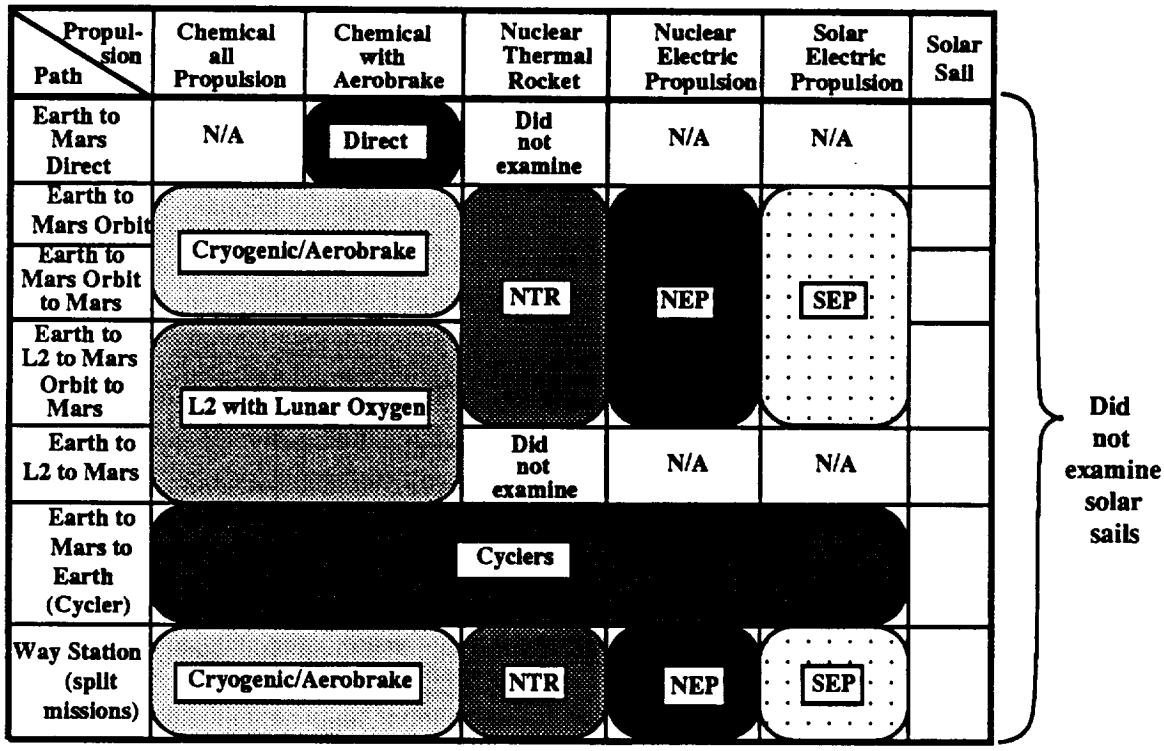
Architecture	Features	Rationale
Cryogenic/aerobraking; all-propulsive option	Cryogenic chemical propulsion and aerobraking at Mars and Earth. LEO-based operations.	NASA 90-day study baseline; lower development cost
NEP	Nuclear-electric propulsion for Mars transfer; optionally for lunar cargo.	High performance of nuclear electric propulsion
SEP	Solar electric propulsion for Mars transfer; optionally for lunar cargo.	High efficiency of solar electric propulsion; find cost crossover for array costs.
NTR (nuclear rocket)	Nuclear rocket propulsion for Lunar and Mars transfer.	High Isp of nuclear rocket enables avoidance of high-energy aerocapture at Mars.
L2 Based cryogenic/ aerobraking	L2-based operations; use of lunar oxygen.	L2 base gets out of LEO debris environment. Lunar oxygen reduces resupply by ~ factor 2.
Direct cryogenic/ aerobraking	Combined MTV/MEV refuels at Mars and LEO. "Fast" conjunction profiles.	Eliminates Mars orbit operations.
Cycler orbits	Cycler orbit stations a la 1986 Space Commission report	Eliminates boosting massive Mars transfer vehicle.

SPACE TRANSPORTATION ARCHITECTURES

FIGURE 1-8

These seven architectures were reported by this study in March 1989, as covering the options available to Mars program architects. In the subsequent nine months, no reasonable candidates outside this set were identified. A number of variations on the cycler architecture were found, including one called "flyby-dash," suited to nuclear thermal rocket propulsion, also called "NTR-dash" when implemented with a nuclear rocket.

Concurrent with this study, during the summer and fall of 1990, the American Institute of Aeronautics and Astronautics (AIAA) conducted an Innovation Outreach, as part of the broader Outreach activity initiated by the Space Council. As part of the AIAA Outreach, the AIAA Architecture Working Group constructed a map of Mars transportation architectures, a matrix of propulsion concepts and pathways (Reference 13). This was intended to bound all possibilities except for radically new propulsion technologies that might be proposed. It successfully captured all of the AIAA Outreach inputs except those related to solar wind magnetic sails. (MagSails represent a new concept worthy of further investigation.) We cross-correlated our seven architectures and their operating modes with the AIAA map, as shown in Figure 1-9. The architectures encompassed all parts of the AIAA map except for certain ones intentionally omitted.



MARS PROPULSION ARCHITECTURE ALTERNATIVES ON AIAA MAP
FIGURE 1-9

Certain regions on the AIAA map were not examined for the following reasons:

a. All-propulsive cryogenic propulsion for landing on Mars is not practical given the roughly 4 km/sec deceleration that can be achieved by a landing aerobrake.

b. We did not examine landing a nuclear rocket on Mars, given contamination concerns and the present lack of an NTR technology that can operate with an oxidizing propellant. (The NTR landing concept proposes the use of Mars' atmosphere CO_2 as reaction mass.)

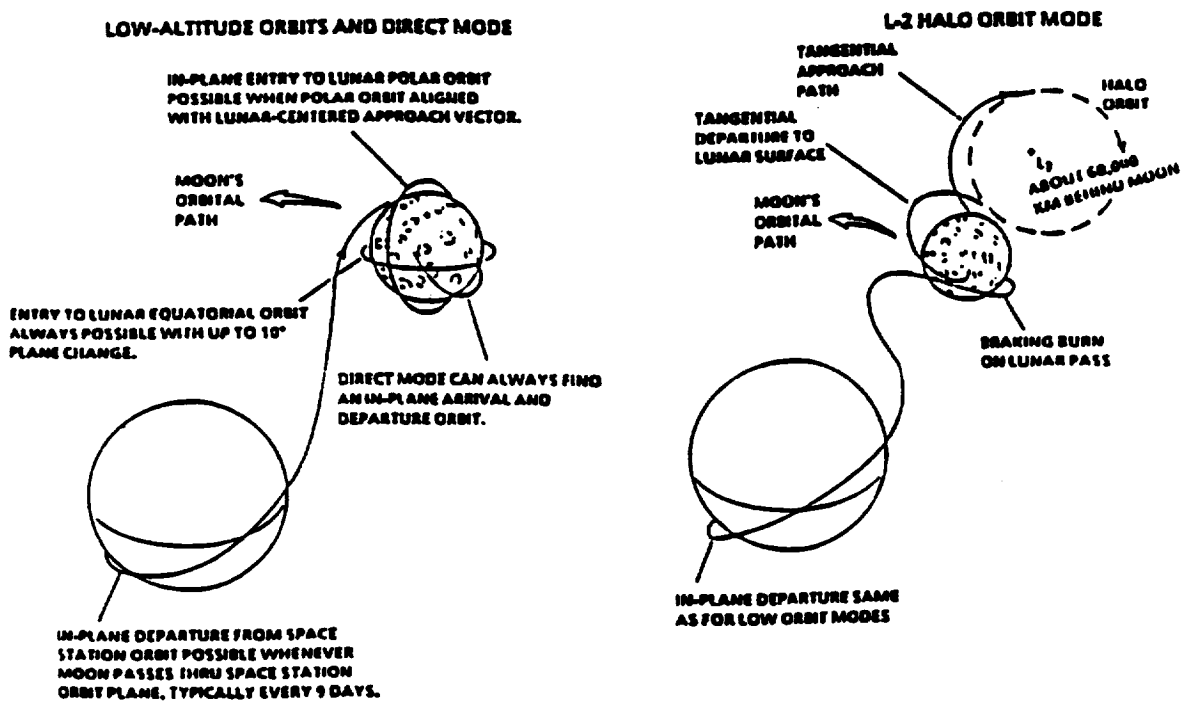
c. Low-thrust electric propulsion systems are not capable of planetary landing or ascent.

d. Solar sails can produce thrust (at Earth's distance from the Sun) of 4.5 to 9 N/sq. km.; the low areal thrust capability implies tens of square kilometers for solar sails; a technology that can handle these huge areas does not seem to be in hand.

1.4 Mission Profiles and Modes

1.4.1 Lunar Profiles

The present study analyzed only high-thrust profiles. Three basic profiles are pertinent to lunar transportation: *Direct, Lunar Orbit Rendezvous* (LOR), and *Halo Orbit Rendezvous* (HOR). These are depicted in Figure 1-10.



LUNAR MISSION MODES
FIGURE 1-10

The direct mode lands the entire mission vehicle on the Moon. This eliminates lunar orbit operations and concerns with alignments for return to lunar orbit from the surface. Lunar orbit rendezvous (LOR) was used on Apollo. It is ordinarily the most efficient mode. The Earth-Moon transfer system is left in lunar orbit; an excursion vehicle goes to the lunar surface and back. In some modern renditions of LOR, an aerobrake and tanks loaded with

return propellant are left in lunar orbit. The principle is the same: don't land non-essentials on the Moon. The halo orbit option, discovered and described by Robert Farquhar of GSFC, is the same as LOR except that the node point is a lunar libration point, typically L2, rather than low lunar orbit (Reference 14). The halo orbit requires greater delta V than LOR, about the same total mass as the direct mode, and is competitive only when lunar oxygen is available.

1.4.2 Lunar Mission Design Constraints

Lunar missions were assumed to depart from a particular low Earth orbit, because they always involved the mating of at least two Earth-launched payloads. If the operational low Earth orbit is at 28.5 deg inclination as we assumed, an in-plane departure is possible about every nine days.

Apollo lunar landings were constrained to have early (lunar) morning lighting conditions at the landing site. The Sun was behind the descending Lunar Module and the sun line was below the descent path. This constraint is seen as applicable to SEI lunar missions until improved landing sites are prepared and electronic landing aids are in place. Landing site lighting constraints are tied to the lunar solar month, while other constraints are tied to the lunar sidereal month. Missions strongly constrained by the sidereal month, such as lunar polar orbit LOR with return to Space Station Freedom orbit, will only have one favorable launch period per year. At other times, lunar orbit wait times of weeks are required to satisfy constraints.

Lunar equator surface sites and equatorial orbits permit ascent or abort every revolution of an orbiting spacecraft (LOR mode). Direct and halo orbit rendezvous profiles permit ascent or abort any time from any site. For the LOR mode, a lunar parking orbit can be chosen for any desired surface stay, but low energy return from the surface to the parking orbit is only possible at particular times.

Return constraints depend on whether LOR is used and if so, on lunar parking orbit inclination. Figure 1-11 is a "truth table" for lunar mission constraints. As may be seen, an LOR polar mission with return to Space Station Freedom (SSF) orbit is the most constrained. LOR equatorial, direct, or halo orbit rendezvous with return direct to Earth have no return constraints.

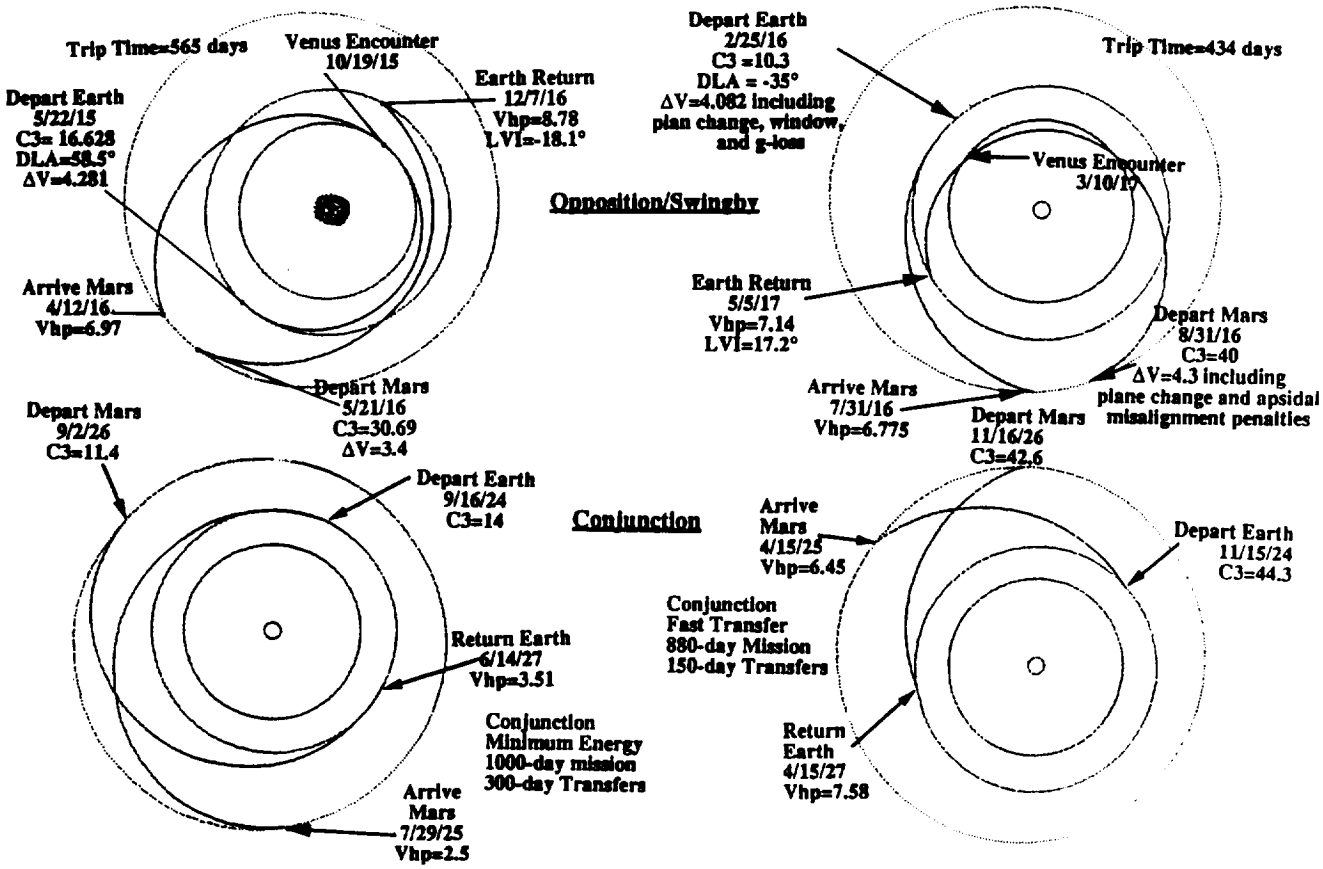
Mission Mode	TLI Constraint	Landing Site Lighting (unprep. site)	Ascent	Return to Earth	Return to Space Station Orbit
Single Launch Direct	No	Yes	No	No	Yes, 9 days
Single Launch to LOR Equatorial	No	Yes	Yes, every orbit	No	Yes, 9 days
Single Launch to LOR Polar	Yes, 14 days	Yes	14 days	Yes, 14 days	Yes, 9 days and 14 days
Single Launch to Halo Orbit	No	Yes	No	No	Yes, 9 days
Earth Orbit Launch Direct	Yes, 9 days	Yes	No	No	Yes, 9 days
Earth Orbit Launch to LOR Equat.	Yes, 9 days	Yes	Yes, every orbit	No	Yes, 9 days
Earth Orbit Launch to LOR Polar	Yes, 9 days and 14 days	Yes	14 days	Yes, 14 days	Yes, 9 days and 14 days
Earth Orbit Launch to Halo Orbit	Yes, 9 days	Yes	No	No	Yes, 9 days

LUNAR FLIGHT MECHANICS CONSTRAINTS

FIGURE 1-11

1.4.3 Mars Mission Profiles and Modes

The two basic Mars profiles are called conjunction and opposition, so named for the relative positions of Earth and Mars during the mission visit period at Mars. The conjunction profile requires almost three years for a Mars mission and inherently has long stay times, a year or more, at Mars. The opposition profile has no specific minimum mission duration, and a maximum less than two years. Opposition stays at Mars are short, typically 30 days. Opposition profile durations less than about a year do not appear practical for propulsion systems likely to be available to SEI before 2025. Representative profiles are depicted in Figure 1-12.



TYPICAL EARTH/MARS ROUND TRIP TRAJECTORIES

FIGURE 1-12

Mars mission design is concerned with long duration space crew exposure to micro-gravity and cosmic ray radiation environments; conjunction fast transfers provide a means to minimize both problems. At typical opposition/swingby energies, conjunction fast transfers are less than six months each way. The minimum-energy profile invokes no trip time constraints and has long transfer times, i.e. about a year each way, for the "difficult" years when Mars is near aphelion, e.g 2009 and 2025. For "easy" year opportunities, minimum-energy transfer times can be as short as 200 days. Since the conjunction profile is able to center transfers on the minimum-energy time, delta V penalties for fast transfers are significantly less than for opposition profiles. Conjunction fast transfer profiles are in the same trajectory family as minimum-energy ones.

The opposition profile gives priority to reduced mission duration rather than to minimum energy. In the case of a minimum-energy transfer to Mars, the space vehicle arrives too late for a minimum-energy return. While the conjunction profile simply waits for the next low-energy return, the opposition profile pays the energy price to return to Earth after a short (30-60 day) stay at Mars.

The opposition/Venus swingby (OVSB) profile was selected for the 90-Day Study because it achieves relatively fast, 1.5 year trips at energies only modestly greater than conjunction missions. Three types of OVSB profiles occur: outbound swingby, return swingby, and dual swingby. Most of the OVSB profiles described in recent studies are local minimum energy, with durations that approach 600 days and dual swingbys that can exceed 700 days. These durations can usually be shortened at modest energy penalties. Sometimes, as in the 2015/16 opportunity year, both outbound and return swingby options exist, with different durations. For 2015/16, the minimum energy outbound duration is 565 days; the return option is 435 days, and the latter can be reduced to 400 days or less at a moderate energy cost. OVSB's at less than 450 days, however, are uncommon.

These profiles are not contiguous as are conjunction fast transfers, in that low-energy date combinations for one of these profiles are separated from the others by high-energy regions. Earth departure dates for outbound swingbys can occur as much as a year later than for inbound swingbys; dual swingbys are usually somewhere in between.

Advantages of these profiles are summarized in Figure 1-13; an advantage of one translates to a disadvantage of the other. A simple count of advantages leads to a preference for the longer mission. Two of the advantage/disadvantage factors need some explanation:

a. The round trip time for the opposition profile is less than the Earth/Mars synodic period of 26 months; therefore continuous presence at Mars is possible by operating the transportation system in a crew rotation/resupply mode. Conjunction profiles do not offer continuous crew presence at Mars unless crews stay on Mars about 3 years, with total times away from Earth about 5 years. Opposition profiles support a crew exchange operation with average 26 months' stay on Mars and total duration about 3.5 years, assuming successive profiles of the same type. Opposition/swingby profiles are somewhat irregular. An outbound Venus swingby profile leaves Earth nearly a year before an inbound swingby. A crew leaving on an outbound swingby and returning on an inbound swingby will spend 5 years away from Earth, about 3 on Mars. A crew leaving on an inbound and returning on an outbound will spend 3 years away from Earth, about 2 on Mars.

Opposition Advantages

- Shorter overall trip time, by at least a year.
- Transfer vehicle usually returns in time to be reused on next opportunity.
- Enables crew rotation/resupply mode with synodic period stay time.

Conjunction Advantages

- Lower energy; significantly less launch mass unless very high Isp available.
- Venus swingby complexity not needed.
- Long stay times at Mars.
- Shorter transfer times.
- Longer launch windows.
- Elliptic parking orbits can be optimized.
- More hardware reuse.

CONJUNCTION AND OPPOSITION PROFILE FEATURES FOR MARS MISSIONS
FIGURE 1-13

b. Elliptic parking orbits can be optimized for conjunction missions because the long stay time at Mars permits tailoring of the parking orbit inclination and period so that nodal regression and apsidal advance jointly align the orbit for minimum-energy departure.

1.4.4 Mars Mission Design Constraints

The main design constraint on Mars missions is the timing of departure windows, a result of planetary alignment periodicity. While an entire opportunity, considering all profiles from earliest to latest can last about a year as noted above, the window for a particular profile lasts from about two weeks to about two months.

A departure opportunity for a particular low Earth orbit, such as any chosen assembly orbit, lasts only a few days. This is because a low-energy departure occurs only when the low Earth orbit contains the departure S-vector. Due to relatively rapid low Earth orbit

nodal regression (about 7 degrees per day) the low-energy condition lasts only a few days. This problem may be alleviated by a three-burn departure.

Additional constraints arise from capture orbit design. Elliptic capture orbits usually minimize Mars capture and departure delta Vs. The savings can be significant, i.e. about 1 km/sec for capture and for departure. Elliptic orbits, however, are constrained regarding the location of periapsis, i.e. orbit inclination and line of apsides must be aligned for capture and departure to minimize delta V. Periapsis location is further constrained to enable daylight landing. Low L/D (0.3 - 0.5) landers are limited to landing near periapsis for elliptic parking orbits. High L/D landers (>1.0) have significant cross-range and down-range capabilities. Capture orbit and landing site constraints reflect into interplanetary trajectory design in ways that are incompletely explored.

Earth-Mars trajectory designs frequently assume that periapsis-to-periapsis transfers at Mars are always possible, whereas they generally are not. As noted, in the case of conjunction profiles with long staytimes at Mars, selection of orbit period and inclination permits control of arrival and departure geometry to enable in-plane, periapsis-to-periapsis transfers. In the case of short staytimes at Mars, secular orbit perturbations don't have enough action time to align both plane and periapsis except for short-period orbits. One can always find an orbit that allows in-plane arrival and departure. The problem is periapsis alignment. Short-period (near-circular) orbits can provide rapid enough orbit perturbation to attain alignment. However, for these orbits the significance and delta V advantage of periapsis alignment diminishes.

1.5 Matching Architectures to Profiles

1.5.1 Lunar Architectures

The only lunar architecture evaluated was cryogenic, with all-propulsive and aerobraking options. This architecture can perform the three lunar profiles described above. Evaluations compared profiles and the degree of reuse of lunar vehicles.

1.5.2 Mars Architectures

Mars architecture matching to mission profiles including basing at Mars is shown in Figure 1-14. Operational variant architectures are subsumed under their basic propulsion technology. Four important issues are central to mission profile selection: crew radiation exposure, crew time spent in zero g, the component of mission risk that increases with mission duration, and the added cost of shortening trip time. At one extreme is the notion, frequently expressed, that a Mars round-trip mission should be completed in a year or less. This is possible with certain advanced propulsion technologies, but at considerably higher cost than for longer trips. At the other extreme, trip time is seen as much less important than minimum mass and cost; conjunction profiles are advocated. Crew time in zero g can be minimized by artificial-g spacecraft design. Increase in risk with duration is difficult to quantify.

Mission Profile	Propulsion				Basing	
	Cryo/ All-Prop	Cryo/ Aerobrake	NTR	NEP/ SEP	Orbit	Surface
Conjunction Minimum Energy	√	No advantage over propulsive capture	√	√	√	Later
Conjunction Fast Transfer	Excessive IMLEO	√	√	√	No. Reason for fast transfer is less GCR dose	√
Opposition/ Swingby	Same	√	√	Note 1	√	As a resupply mode
Opposition/ Fast	Same	Excessive IMLEO	√	Not able to make fast trips	√	Same
Opposition/ Split Sprint	Same	Same	√	Cargo only	√	Same

PROPULSION APPLICABILITY TO MARS MISSION PROFILES
FIGURE 1-14

Crew radiation exposure comes from solar proton events (flares) and galactic cosmic rays, and from manmade sources if nuclear propulsion or power are used. Unshielded energy deposition from GCRs varies from 50 to 100 milliGray (5 to 10 rad) per year. Cosmic

rays are highly ionizing; a *quality factor* is applied to energy deposition dose to obtain biologically effective dose. Including quality factor, the low end of the unshielded range does not constrain Mars mission architectures, but the high end exceeds the present NCRP astronaut radiation guideline of 500 milliSieverts/yr (50 rem per year; this guideline is for space shuttle and space station missions; no guidelines have been given for Mars missions). It is possible that guidelines will be reduced in the future, in view of recent research indicating higher cancer risk at a given dose than previously thought.

Five profile options are presented. Conjunction fast transfer implies transfers much less than one year. Opposition/swingby trajectories vary from about 440 to about 550 days. Opposition/fast profiles imply 450 days or less, usually without a swingby. The split sprint is a variation on the fast opposition profile in which the MEV and propellant for the return from Mars are sent in advance on a low-energy profile.

If galactic cosmic ray exposure must be controlled, either shielding must be provided on the transfer vehicle crew habitat or exposure times must be reduced. Shielding the transfer vehicle habitat dramatically increases its mass, requiring high performance propulsion such as nuclear, or favoring a cycler concept where massive habitats are emplaced on a suitable repeating trajectory and left there. To reduce exposure time, the applicable profiles are: (a) conjunction missions with fast transfers, i.e. 180 days or less, (b) fast opposition profiles, e.g. less than 1-year round trip, and (c) Mars surface rendezvous (Mars direct) on a fast opposition profile. The cycler/semi-cycler architectures offer shielding on the Earth-Mars leg, typically 5 months, and provides a 5-6 month conjunction transfer on the return trip. Their transfer times are in the conjunction fast transfer range. During the long stay at Mars, the crew must be on the surface most of the time unless the Mars orbit habitat is shielded.

Available propulsion options become very limited for fast missions. At one year, the only sensible options are NTR split sprints, where return propellant is prepositioned at Mars on a low-energy profile, or the use of a nuclear gas-core rocket. Below one year, the gas-core rocket quickly becomes the only option. We did not evaluate gas-core rockets in this study because too little is known about the technology to make cost and programmatic estimates.

1.6 SEI Program Activity Levels for Architecture Analysis

Selection of transportation architectures is influenced by activity level much more than by purpose or strategy, because transportation basically responds to "how much" and "how often." Three activity levels, "minimum," "full science menu," and "industrialization and settlement," were defined to assess the sensitivity of architecture selection to activity level.

1.6.1 Activity Level Definition

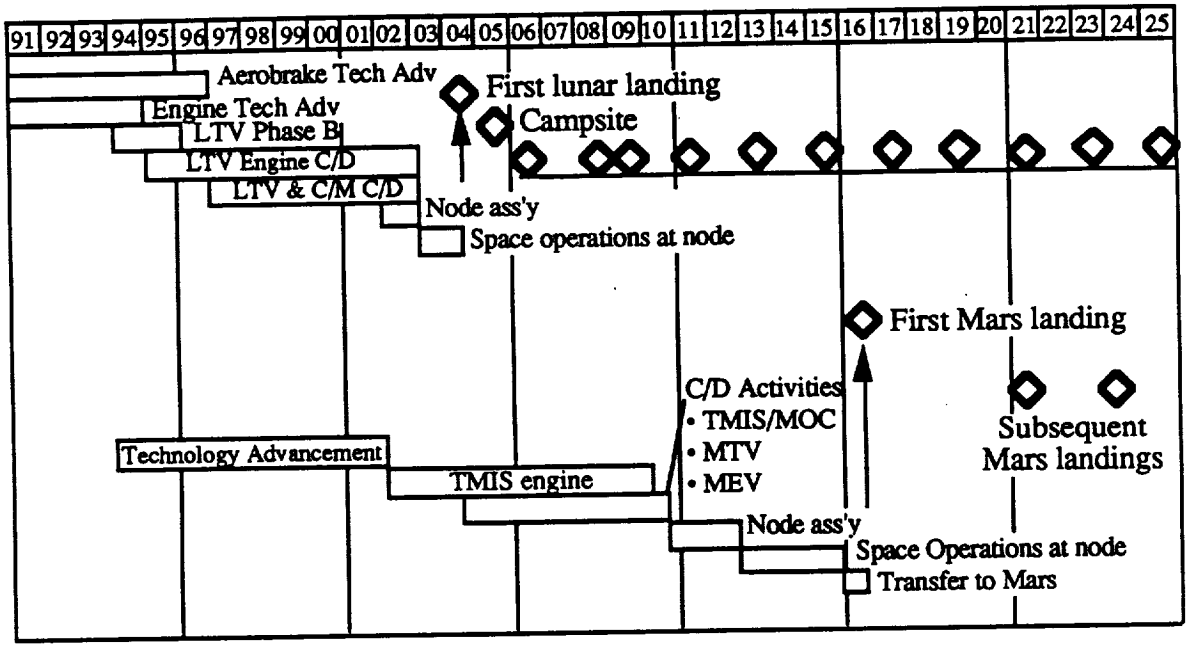
The three levels of activity are summarized in Figure 1-15. The minimum was just enough to meet the President's objectives; in fact "return to the Moon to stay" was interpreted as permanent facilities but not permanent human presence. The minimum program had only three missions to Mars. The median (full science) program aimed at satisfying most of the published science objectives for lunar and Mars exploration. The maximum program aimed for industrialization of the Moon, for return of practical benefits to Earth, and for the beginnings of colonization of Mars. The range of activity levels, as measured by people and materiel delivered to planetary surfaces, is about a factor of 10. The range of Earth-to-orbit launch rates was less, since we adopted results of preliminary trade studies, selecting more advanced in-space transportation technologies as baselines for greater activity levels.

Minimum	Median (full science)	Industrialization/settlement
Just enough to meet President's objectives	Meet science objectives of Lunar/Mars exploration	Return of practical benefits to Earth
• Permanent lunar facilities, not permanent human presence	• Human permanence	• Extensive facilities and infrastructure on the Moon by 2025
• Astrophysics observatories • Man-tending capabilities • Explore interesting sites	• Opportunity of lunar geoscience • In-situ resource technology	• Lunar population 30 by 2025
• Three missions to Mars	• Order of magnitude more crew time on Mars	• Mars population 24 by 2025
• Similar to Apollo • Two sites per mission • Samples within a few km. of landing sites	• Approaches permanent base (stay time > 1 year)	• Capable of increasing Mars population by 24 per opportunity by 2025

ACTIVITY LEVELS FOR ARCHITECTURE EVALUATION

FIGURE 1-15

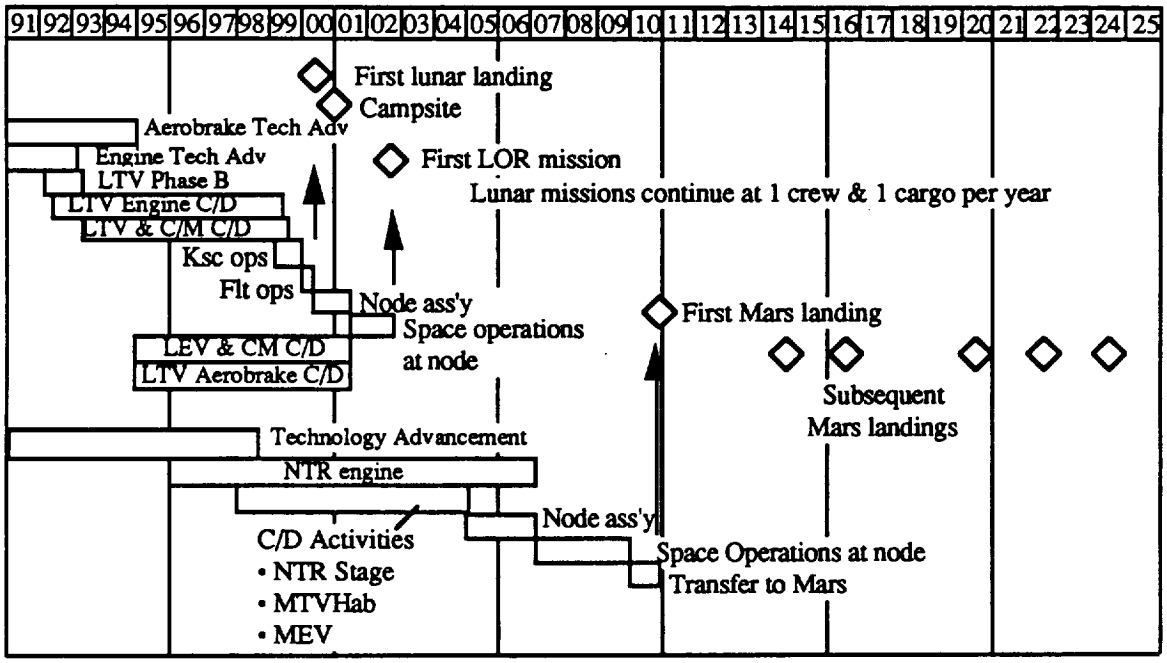
The minimum program scenario is shown in Figure 1-16. It averages about 0.5 lunar trip per year and has only three Mars missions. Lunar science facilities are man-tended. Each Mars mission carries two landers (MEVs) for added exploration capability and a measure of rescue capability. Surface stays are about 30 days. Lunar and Mars in-space transportation systems are expendable.



MINIMUM PROGRAM SCENARIO
FIGURE 1-16

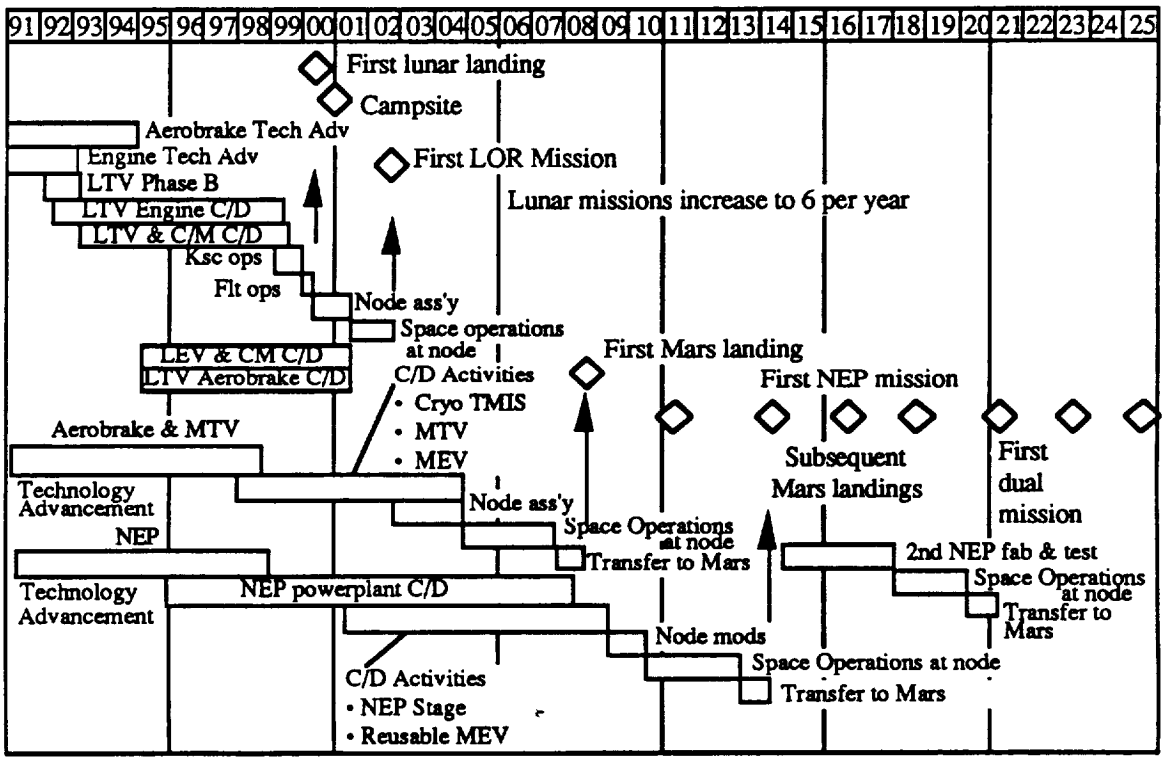
The full science program is shown in Figure 1-17. It averages about 2 lunar missions per year, to establish permanent human presence on the Moon with adequate supplies and equipment for extensive science and exploration. Lunar oxygen for lunar transportation is introduced about mid-way through the lunar program. Six Mars missions are accomplished, with later missions staying on Mars for more than a year. The Mars missions use multiple landers, as many as four late in the program.

The industrialization and settlement program shown in Figure 1-18 is very aggressive for both the Moon and Mars. Thousands of tons of industrial equipment are delivered to the Moon, driving lunar cargo trips up to five per year. Lunar oxygen is placed in production as early as possible. One crew trip per year leads to a population of 30 as crew stay times on the Moon increase to several years.



FULL SCIENCE PROGRAM SCENARIO

FIGURE 1-17



INDUSTRIALIZATION AND SETTLEMENT PROGRAM SCENARIO

FIGURE 1-18

Initial Mars missions use a cryogenic/all-propulsive system. The aggressive nature of the scenario merited an initial Mars mission as early as possible, and the reference nuclear electric propulsion system is not ready in time. The NEP missions are operated in a crew rotation/ resupply mode, opposition profile, with each crew staying one synodic period (about 2.2 years). The reference scenario evolves to reusable MEVs based on Mars, fueled from Mars resources. Heavy cargo capability is provided, up to 250 t. per opportunity by 2020. The Mars population grows to 24, and by the end of the scenario can continue growing by 24 or more per opportunity.

These activity levels were selected with underlying program objectives in mind:

a. The minimum lunar program establishes astrophysical observatories on the Moon and provides a man-tending capability to maintain them. To the extent that man-tending lunar visits are not needed for the observatory system, the transportation capability can be used to explore interesting lunar sites for lunar geoscience objectives.

b. The minimum Mars program is very similar to Apollo, i.e. six sites visited for short periods (two sites per mission and three missions); samples obtained within a few kilometers of each landing site. If the manned visits are preceded by suitable robotic missions, the scientific payoff for these visits can be high relative to the investment.

c. The "full science" lunar program adds human permanence at the Moon for extensive scientific and technological exploration. Where the minimum program offers very little opportunity for lunar geoscience, this program offers much. It also permits development of in-situ resource technology for production of surface systems. The reference program also emplaced a lunar oxygen production system to serve the transportation system.

d. The "full science" Mars program has many times the crew person-days on Mars as compared to the minimum program. This program falls short of a permanently-occupied base on Mars, but achieves surface stays greater than a year.

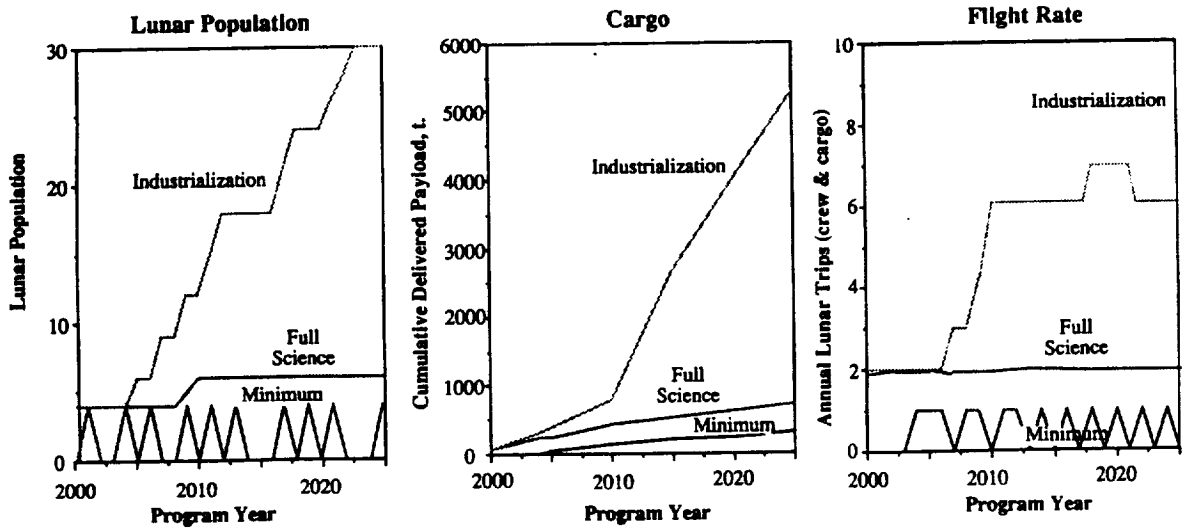
e. The lunar industrialization program adopts production of helium-3 as a strawman industrial objective. Delivery to Earth of lunar helium-3 at an equivalent power generation rate of 1 GWe by the year 2025 was selected as a target. This is roughly consistent with DoE thinking as to timing of the earliest commercial fusion reactors, and it leads to an orderly lunar buildup with growth rates in a plausible range.

Whether a lunar helium-3 scenario, or for that matter any other space-energy-for-Earth scenario, is economically feasible, remains an open question. A lunar helium-3 scenario defined in a 1989 NASA report was used as a convenient strawman to set high-activity transportation requirements (Reference 15).

f. The Mars settlement program moves towards Mars settlement, with population growth up to 24 per opportunity by 2025. It is clear that further scale-ups of transportation systems could reach population growth rates high enough to be a genuine settlement. Mars is the most Earth-like planet (but it isn't very Earth-like). Speculation about Mars as a habitable world suggests that a human civilization could eventually thrive there. Some scientists contend that the atmospheric pressure on Mars could be increased, e.g. by greenhouse gases, to the point that space-suits would not be required.

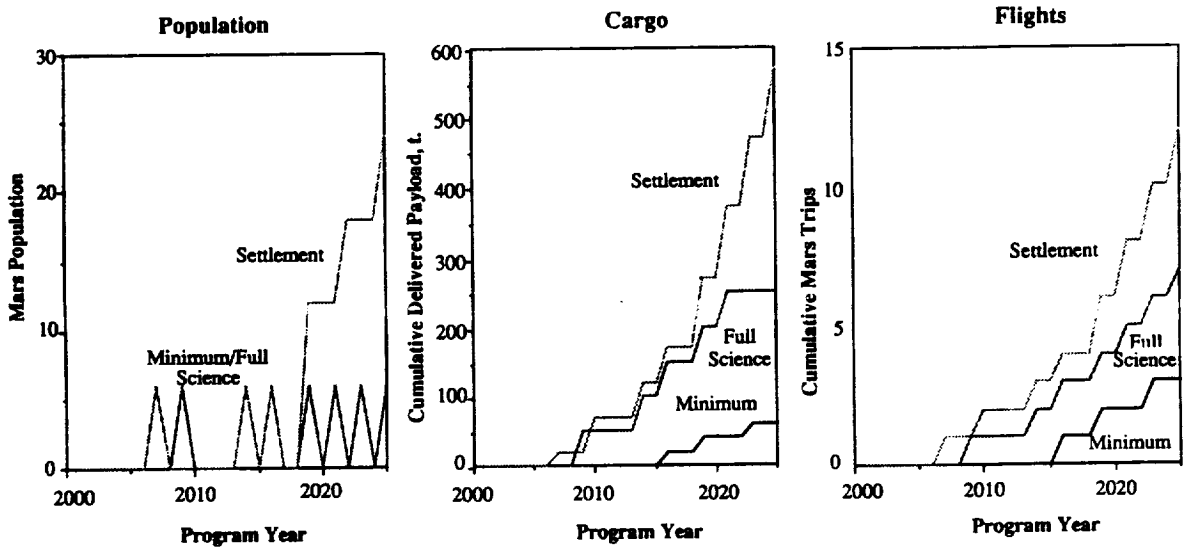
1.6.2 Program Scenario Comparisons

The lunar and Mars program scenarios in terms of population, cumulative cargo delivered, and flight rate are compared in Figures 1-19 and 1-20. The lunar population for the minimum scenario is four people for 30 to 40 days about every other year. The Mars population for the minimum scenario is 6 people on each of 3 conjunction missions, with two 30 to 40 day surface stays per opportunity. The full science menu scenario grows to year-long surface stays on conjunction missions. The lunar industrialization program goes to long stay times with indigenous food growth to build population. The Mars proto-settlement program obtains continuous presence by operating the NEP on an opposition-like profile in crew rotation/resupply mode. Later in this scenario, a second NEP is operated to provide two trips to Mars each opportunity. These scenarios served as inputs to the manifesting and life cycle cost analyses to facilitate overall programmatics analysis.



LUNAR PROGRAM COMPARISON

FIGURE 1-19



MARS PROGRAM COMPARISON

FIGURE 1-20

1.7 Architecture Evaluation Results

Architecture evaluation is driven by a few factors: development cost, operations cost, keeping options open, mission risk, and evolution potential. Architecture evaluations directly translate into technology advancement priorities.

Missions to Mars set the performance criteria for selection of transportation options and evolutions for the Space Exploration Initiative. Lunar missions are of short duration, i.e.

days, with modest payloads. They benefit much less from advanced propulsion performance than Mars missions. Mars missions last months to years; because of their duration, they require massive, costly crew habitats for the transfers from Earth to Mars and back. Further, lunar missions require high thrust propulsion, at least for crew missions, while both crew and cargo missions to Mars can use low-thrust electric propulsion. A strictly lunar scenario would admit only nuclear thermal rockets as an advanced propulsion option, since the low thrust of electric propulsion systems makes them ill-suited for lunar crew transport and they appear not worth developing merely for lunar cargo transport.

1.7.1 Ground Rules and Assumptions for Comparisons

A set of nominal bench-mark performance and mass requirements were used to illustrate performance comparisons, as summarized in Figure 1-21. These are typical, and do not represent any particular Mars opportunity. For the conjunction case, performance margin over absolutely minimum energy is provided to obtain reasonable transfer times of 5 to 10 months. For the opposition case, the values are close to those for the 2016 Venus swingby opportunity, which is of median difficulty.

Conjunction	C3	Vhp	DV
Earth departure	15	3.87	4000
Mars arrival	10.4	3.23	1200
Mars departure	14	3.75	1500
Earth arrival	16	4	1340*

* Return to 24-hr elliptic orbit; vehicle is subsequently refueled by LTV for return to LEO.

Opposition	C3	Vhp	DV
Earth departure	25	5.0	4300
Mars arrival	35	5.9	3000
Mars departure	35	5.9	3400 **
Earth arrival	49	7	2525 *

* Return to 24-hr elliptic orbit; vehicle is subsequently refueled by LTV for return to LEO.
** Includes apsidal misalignment penalty

Standard Mars parking orbit 250 km x 24.6 hr.
Mars landing - 1200 m/sec Mars ascent - 5500 m/sec.
Direct return, Mars surface to conjunction Earth transfer - 6775 m/sec.
Notes: For aerobraking, Vhp limit 7 km/sec; if Vhp is higher, e.g. sprint trajectories, excess Vhp is taken out propulsively. Post-aerocapture ΔV is 200 m/sec. No Earth arrival ΔV for expendable cases.

Payloads

Transit hab:	Opposition fast trip - 35 t.	Opposition/ swingby - 40 t.	Conjunction 50 t.	ECCV - 8 t.
Expendable MEV	Aerocapture & landing - 84 t.	Landing only - 76 t.	Includes 25 t. surface payload	
Reusable MEV-	40 t. resupply			

GROUND RULES AND ASSUMPTIONS FOR COMPARISON
FIGURE 1-21

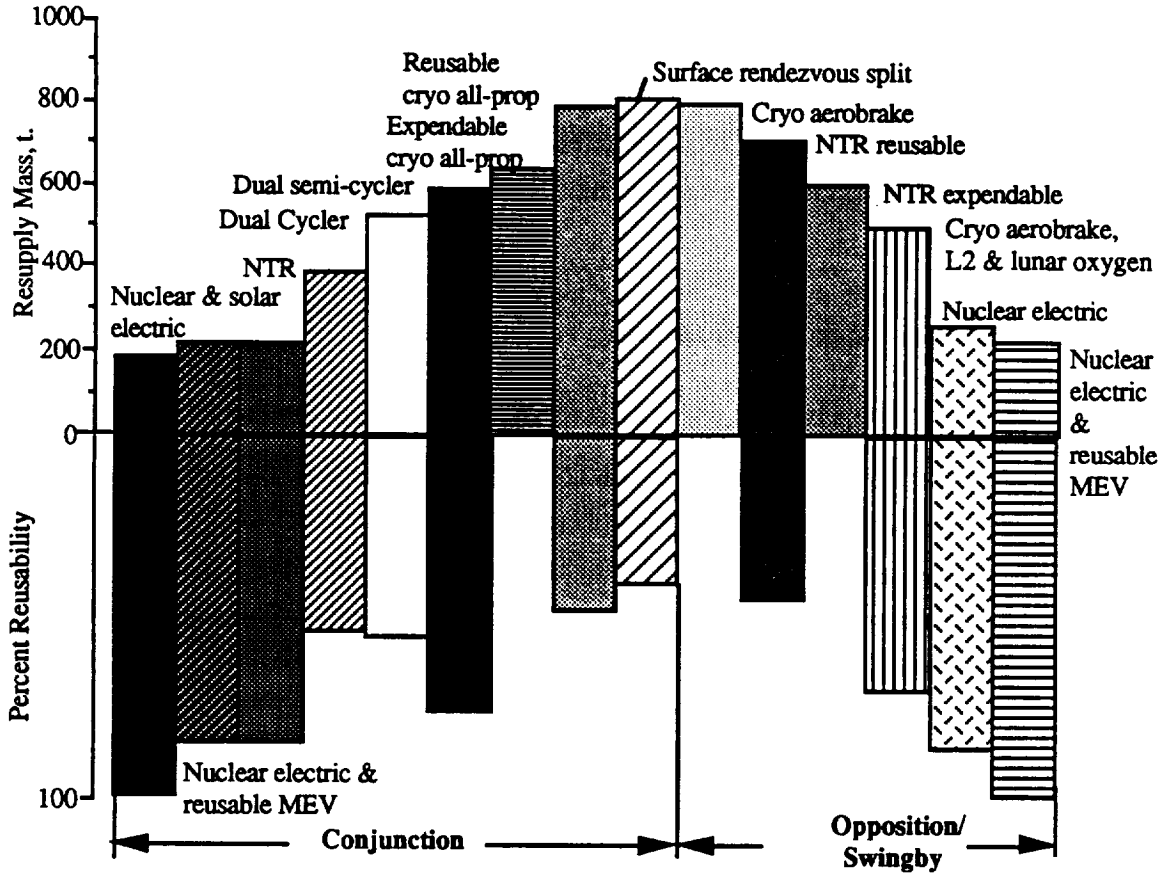
We used 475 Isp for cryogenic propulsion and 925 Isp for nuclear thermal rocket. Trades reported elsewhere in this report also used 1050 Isp, representing an advanced-technology nuclear engine.

Solar and nuclear electric systems optimize between 5,000 and 10,000 seconds. Their performance was calculated by low-thrust path generators; the delta Vs shown in the figure do not apply to electric systems. For purposes of the comparisons, we assumed the electric systems spiraled out of and into Earth's gravity well to LEO, but are manned only after and before the spirals. Analysis shows that the electric systems are somewhat more efficient if parked at a high-orbit node such as Earth/Moon L2 and resupplied by a smaller electric-propulsion ferry vehicle.

1.7.2 Architectures Mass and Reusability Summary

Resupply performance, i.e. resupply mass delivered to LEO, for the principal Mars profiles, for opposition and conjunction missions is shown in Figure 1-22. Resupply performance is the key measure of efficiency for repeated missions; for single or expendable architectures, the total initial mass in Earth orbit (IMLEO) applies. Also shown is the fraction of propulsion and mission hardware available for reuse at the end of the mission; this is an important mission cost factor since hardware replacement cost can easily exceed ten times the cost to place mass in LEO. Generally, the modes with lower resupply requirements and higher reusability employ more advanced propulsion and imply more developmental effort. Therefore, one expects low resupply modes to be attractive mainly for larger-scale programs, where the greater technology and development effort is effectively amortized. Less advanced systems are cost effective for lesser programs. Note that for Mars surface rendezvous (Mars direct), the performance as a crew delivery system for a man-tended but long-duration stay base is quite attractive. Since Mars direct is only operable on a conjunction profile, it is not well suited to crew rotation for a permanent base. The NTR-dash profile appears well-suited to this requirement.

In addition to examining basic architectures, various combinations and evolutions were also considered. As examples, the L2/lunar oxygen architecture benefits from advanced propulsion (NTR, NEP or SEP) for the Earth orbit-L2 leg. Values shown on the chart are for the all-cryogenic L2/lunar oxygen option.



ARCHITECTURE COMPARISON FOR RESUPPLY MASS AND REUSABILITY
FIGURE 1-22

A mass advantage does not necessarily translate to a cost advantage. For example, lunar oxygen supply to the L2/lunar oxygen architecture appears to have attractive performance. A payback analysis was performed using simple scaling equations. In the case of all-cryogenic propulsion, the IMLEO savings is about 300 t. per mission, compared to cryogenic/aero-braking from LEO. The lunar oxygen production rate needed to fly every Mars opportunity is about 360 t./yr. The quantity of production and power equipment needed for this production rate is not well-defined; a moderately optimistic figure is 1 t. equipment per t./yr production. To save 300 t. per mission, 360 t. of production equipment is delivered to the lunar surface. Since the ratio IMLEO to lunar cargo is about 6, the IMLEO cost is over 2000 t., and breakeven occurs only after 7 to 8 Mars opportunities.

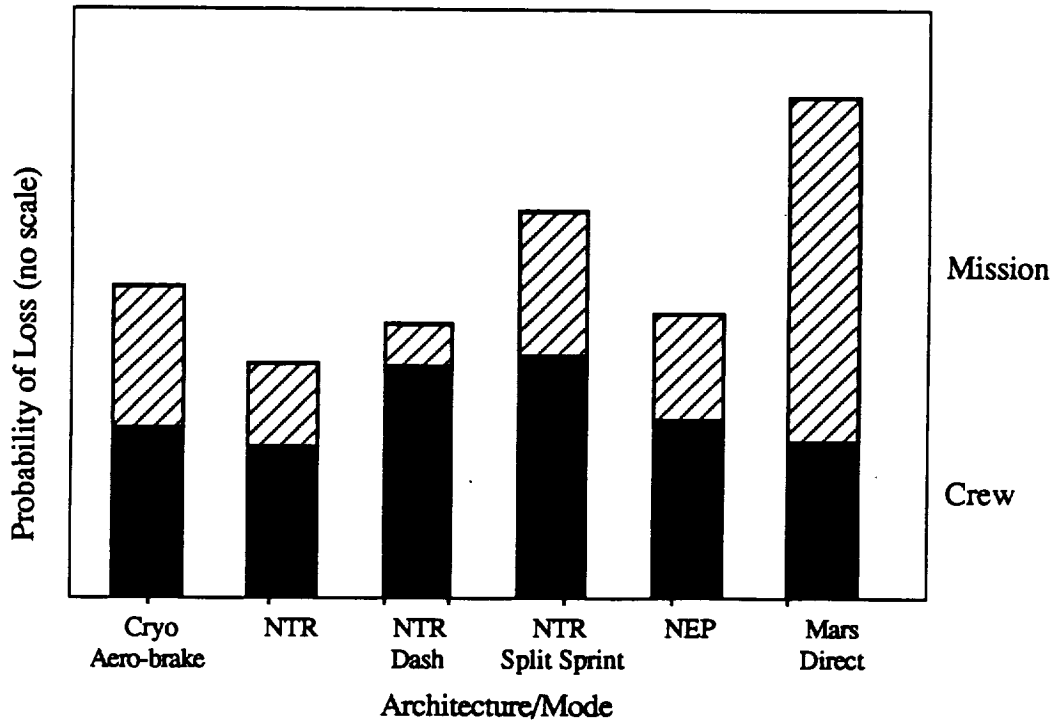
Use of advanced propulsion reduces the resupply requirement and the payback time for lunar oxygen. However, if advanced propulsion is available, why not use it directly for the Mars mission? Use of a mass driver to deliver the lunar oxygen to L2 (in cannisters) also improves the payback, but introduces another new technology. We did not evaluate this option in the present study.

1.7.3 Mission Risk Comparison

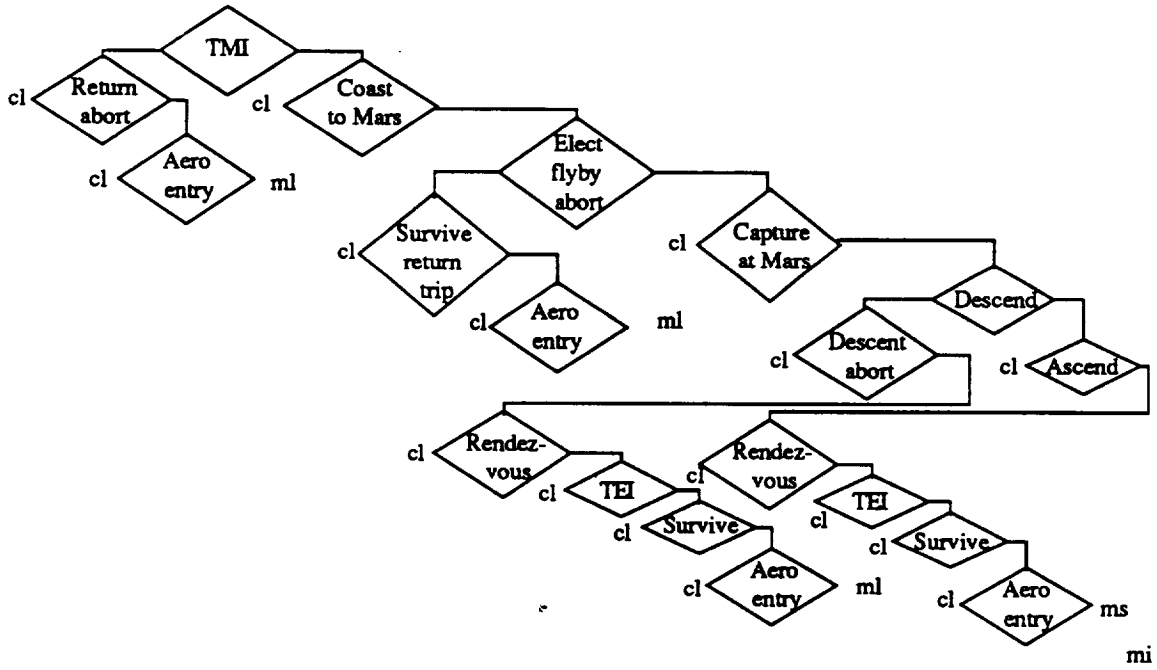
Mission risks were compared in a semi-quantitative way. The methodology is rigorous and quantitative, but reliability and safety estimates for SEI hardware and maneuvers are quite rough. We made representative estimates with an attempt to be consistent, i.e. the same type of maneuver was given the same reliability estimate for all cases. Plausible differences were used, e.g. aerocapture was judged higher risk than propulsive capture. Abort modes were included where available.

Comparative risks for crew loss (cl) and mission loss (ml) or several architectures and modes are shown in Figure 1-23. NTR shows the least risk because of its propulsive capture advantage, and because a free return abort was assumed, as it was for the cryo/aerobrake. The NTR/dash mode does not permit free return abort or descent abort at Mars, so some mission loss risk turns into crew loss risk. As Mars transportation matures and a safe refuge on the surface of Mars is available, the NTR/dash mode risk will be comparable to the other NTR mode. The NTR split sprint mode also exhibits higher risk because of lack of abort modes, e.g. no free return. NEP is shown comparable to, but slightly riskier than NTR. The NEP case is sensitive to the lifetime dependability of the propulsion system; this figure is much more uncertain than NTR reliability. Mars direct has a higher mission loss risk because of its complex automated operations, but the crew loss risk is comparable to the others. The perception of crew loss risk for Mars direct is probably higher than the real risk.

A risk tree for the cryo/aerobrake is illustrated in Figure 1-24. A probability of success value is assigned to each event, and the cumulative probabilities for mission loss, crew loss, and mission success are calculated. As a check, the probabilities must sum to 1.000.



MISSION RISK COMPARISON
FIGURE 1-23



CRYO/AEROBRAKE MISSION RISK TREE
FIGURE 1-24

1.7.4 Man Rating Requirements

The recommended approach to man-rating is shown in Figure 1-25 and lists the systems/sub-systems for which a man-rating requirement was identified. A sequence of major tests and demonstrations to achieve nuclear rocket man-rating is shown in Figure 1-26, illustrating the man-rating approach. Note that two flight demonstration options exist. A decision of which to use depends on whether cargo delivery to Mars is needed before the first manned mission, as would be the case if a conjunction fast transfer and long surface stay is required on the first mission to reduce galactic cosmic ray exposure to the crew.

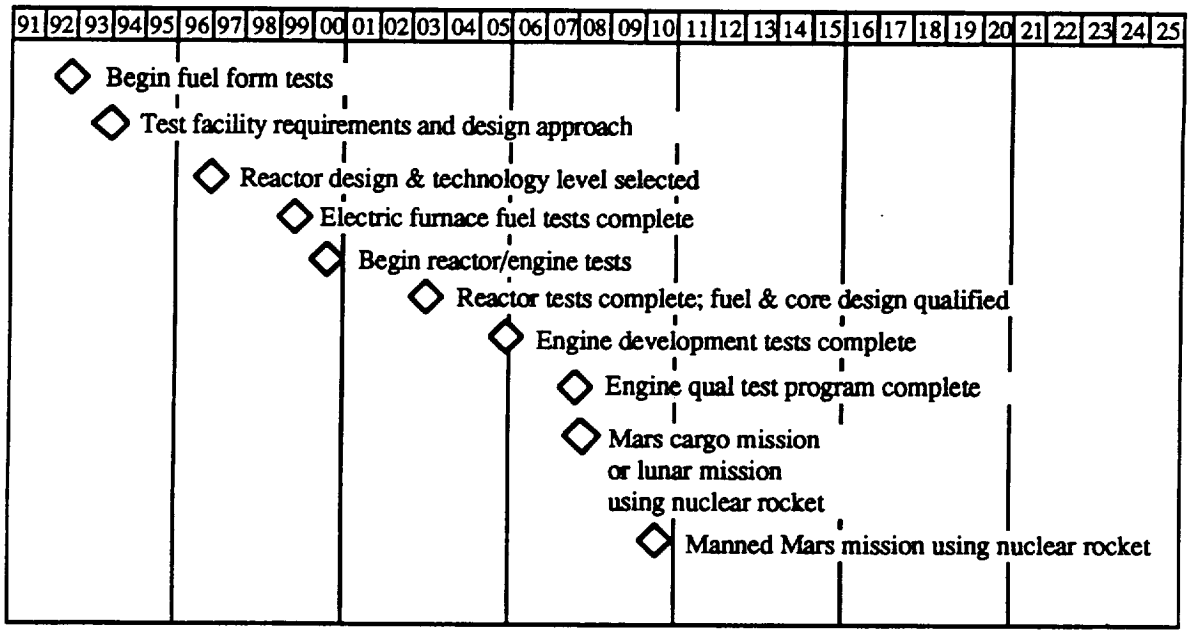
Approach

- Ground-based testing wherever possible.
- Use flight program activities to bootstrap, e.g. lunar aerobrake program builds confidence in Mars aerobrakes.
  - Flight demonstration of critical functions, e.g. Mars cargo landing, before critical manned use.
  - Life demo for long-duration systems before critical manned use, e.g. ECLSS on SSF or lunar surface before manned Mars mission.

Subjects

- Aerobrakes
- Cryogenic rocket engines
- Nuclear rocket engines
- Cryogenic propellant systems
- Attitude control propulsion systems
- Nuclear & solar electric propulsion systems
- ECLSS/TCS
- Crew modules/hab systems
- Vehicle power
- Avionics & Communications systems
- Surface transportation systems

MAN-RATING REQUIREMENTS
FIGURE 1-25

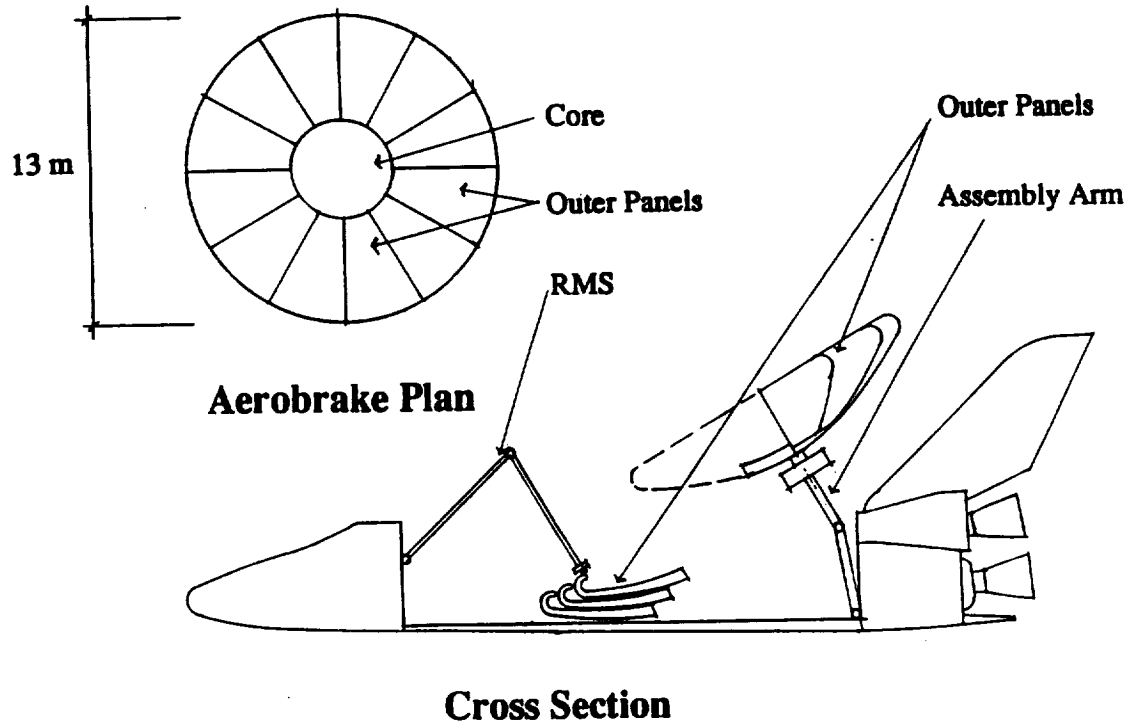


NUCLEAR ROCKET MAN-RATING PROGRAM ELEMENTS
FIGURE 1-26

1.7.5 On-Orbit Assembly

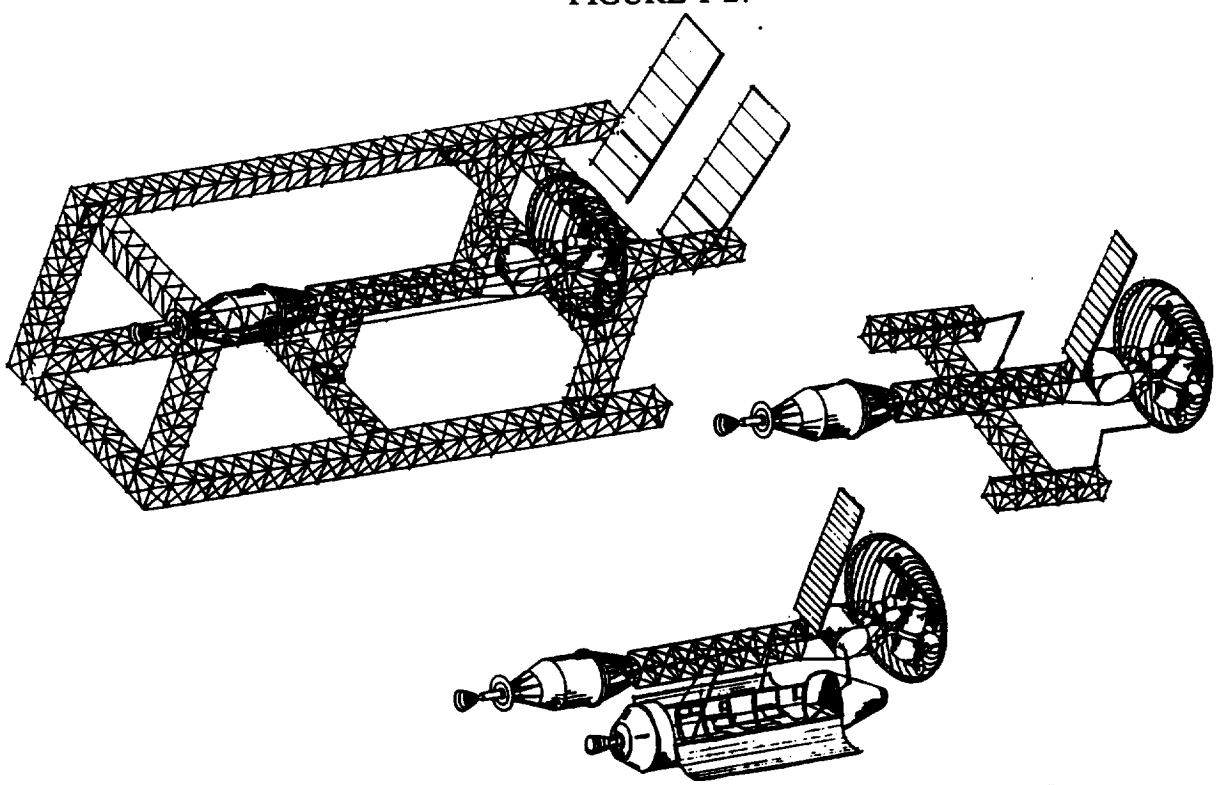
The recommended lunar architecture requires no on-orbit assembly until later in the program. The initial tandem-direct lunar transfer vehicle (LTV) can be launched fully assembled and fueled. Two are needed, requiring in-space rendezvous and berthing. Later, a lunar aerobrake may be introduced, but lunar aerobrake assembly can probably be accomplished from a shuttle Orbiter payload bay as illustrated in Figure 1-27. The underslung heavy-cargo version of the LTV will require some deployment or assembly as it exceeds the 10-meter diameter we assumed as an HLLV diameter limit.

Operations analyses during the latter part of the study concentrated on alternative assembly concepts for the principal architectural options (cryogenic/aerobraking, nuclear thermal rocket (NTR), nuclear electric (NEP) and solar electric (SEP)). Shown in Figure 1-28 are three assembly concepts for the NTR. The one on the left is the conventional "build a shed around it" concept. The center illustration makes the vehicle its own assembly platform with certain attachments such as debris shielding, removed before trans-Mars injection. The right-hand picture is a "smart HLLV" concept where the first launch includes assembly provisions as part of the HLLV payload carrier. Evaluations of these alternatives have not



AEROBRAKE ASSEMBLY TEST IN LEO

FIGURE 1-27



ON-ORBIT ASSEMBLY CONCEPTS USING MARS NTR

FIGURE 1-28

yet been accomplished. The thrust is to simplify the assembly facility by simplifying the assembly process through vehicle design provisions.

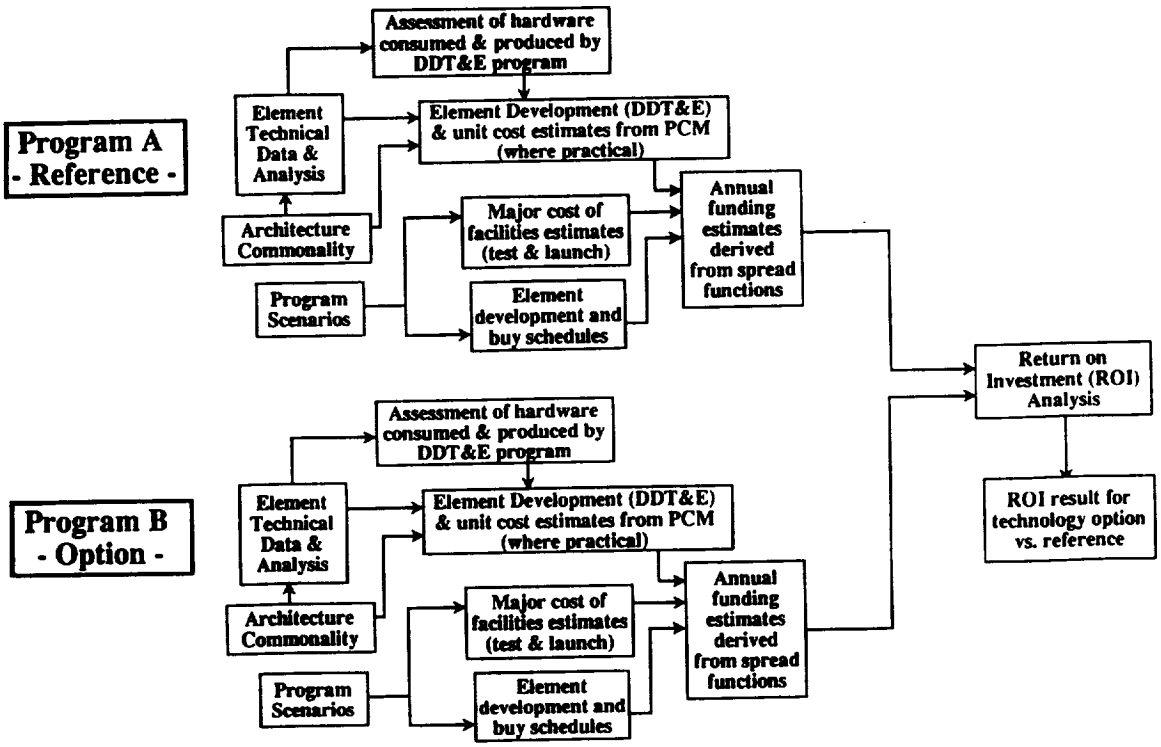
The NEP presents the most difficult assembly challenge because of its extensive fluid systems. The SEP is very large, but the assembly process is repetitive and well-suited to robotics. The NTR is easiest, and the cryogenic options somewhat more challenging because of the greater diversity of assembly tasks.

1.7.6 Costs and Schedules

Initial mass in Earth orbit (IMLEO) is often used for comparisons between propulsion options. However, other factors have great cost impact. Reusability and reduction of development cost by reducing the number, complexity, size, or risk of developments are very important.

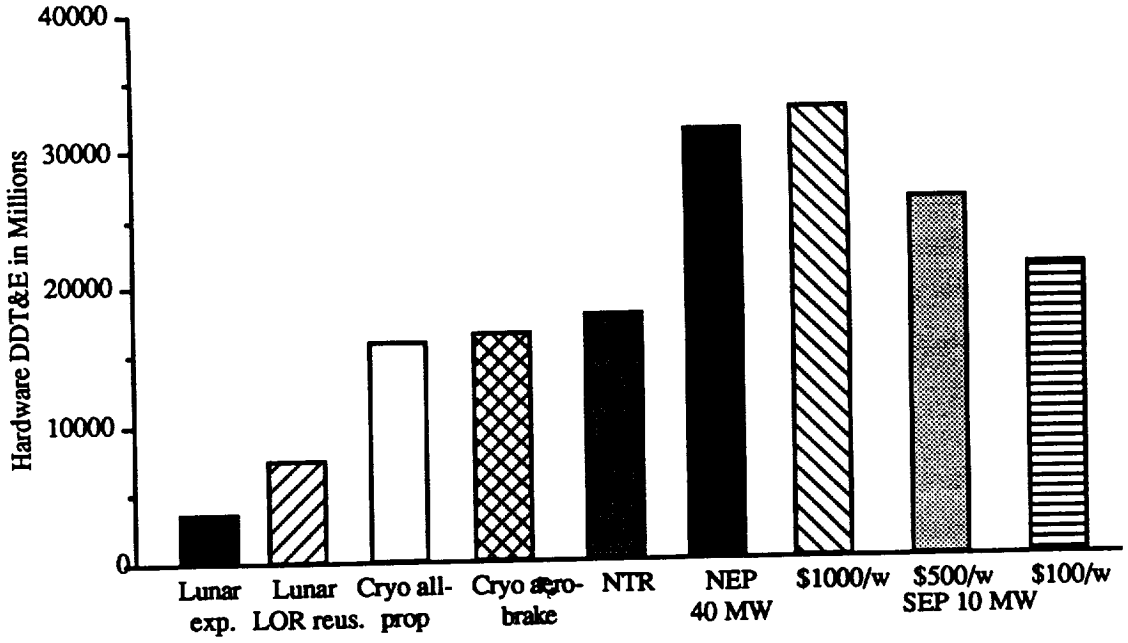
All cost analyses in this study were performed in constant 1990 dollars. i.e. no attempt was made to forecast inflation. The cost analysis logic used in this study is illustrated in Figure 1-29. Development and unit costs for architecture elements were estimated using the Boeing Parametric Cost Model (Reference 16). Complete DDT&E costs were developed by estimating the equivalent number of production units consumed in the development program, e.g. for test articles. The first flight article was assumed produced by the DDT&E program; subsequent flight articles were assigned to production programs. DDT&E estimates for the main architectural options are shown in Figure 1-30. We adopted a minimum production rate of 0.3 units per year. Our scenarios avoided shutdown and restart of production lines. Technology advancement and advanced development programs were estimated for each option, using available estimates for these activities together with our judgments as to what technology advancements and advanced developments are necessary.

In order to get life cycle costs, technology advancement, advanced development, and full-scale development (DDT&E) and production scenarios were established for each program activity level. Launch and flight operations were manifested according to the top-level schedules shown earlier. These were loaded into a spread-sheet life cycle cost model that generates annual funding profiles Examples appear later in this section.



COSTING METHODOLOGY FLOW DIAGRAM

FIGURE 1-29



IN-SPACE TRANSPORTATION DDT&E COMPARISON

FIGURE 1-30

To evaluate alternative architectures, the alternatives were manifested (e.g. NTR versus cryogenic aerobraking) for the same program scenario, including changes in technology advancement, advanced development, DDT&E, production, and HLLV launches. Life-cycle costs were generated for the alternatives. This provided comparative life cycle cost profiles for which return on investment (ROI) could be calculated.

1.7.7 Results of Return on Investment Analyses

The representative ROI scenario is that one option has greater front-end investment cost, leading to savings later in the program through more advanced technology, more hardware reuse, fewer HLLV launches, or combinations thereof. We used a criterion that a ROI of 5% or better is acceptable. That is the approximate cost of money to the government in constant dollars. Return on investment is a severe criterion for front-end investment. To obtain a favorable ROI, 10% or better, an investment must generate large savings. The most advantageous situations occur when a technology advancement reduces near-term DDT&E cost. Technology advancement costs are usually small compared to the DDT&E savings and the savings occur soon after the investment.

The return on investment analyses is summarized in Figure 1-31. Details of the methodology is explained in Reference 12. Results designated "no ROI" had one case always more expensive than the other. An ROI can be calculated only when funding streams cross.

The lunar storable case has very negative ROI because while less technology money (none) is spent, more vehicle stages must be developed so that the negative cost impact of not doing the essential cryo management and engine technology is large and early. The case for reusable lunar transportation is negative for a minimum lunar program and weak for a median program; it is strong for an industrialization-class program. The other results are discussed below and are included here for completeness.

Case	Stor LOR vs cryo direct exp	Reus. LOR vs cryo direct exp	Reus. MEV vs exp MEV	SEP vs NTR		SEP vs NEP	NTR			NTR dash vs NEP	Lunar oxygen	
				$100/w	$500/w		vs cryo all-prop		vs cryo aero brake			
Program	Min	Full science	Ind/ settl	Full science		Any	Min	Full science		Ind/ settl	Full science	Ind/ settl
Result	-85	4.9	No ROI	9.6	-11	No ROI	1.7	15.9	13	44	4	10
Conclusion	Cryo	Reuse case weak	Reus MEV higher LCC	SEP	NTR	SEP better if less cost	CAP	NTR		NTR	No LLOX	LLOX

RETURN ON INVESTMENT SUMMARY
FIGURE 1-31

1.7.8 Specific Results for In-Space Transportation Options

Each of the principal options could become a preferred selection under plausible program circumstances.

a. For a minimum lunar activity level, a simple tandem-staged direct expendable mode is attractive. While expendable systems require continuing hardware production, the production lines must be kept open in any case. For minimum activity levels, the cost of having open lines produce hardware is quite small. At two lunar missions per year, the return on investment for developing an efficient reusable lunar orbit rendezvous (LOR) system is only about 5%. Programs with activity levels of four or more lunar missions per year benefit significantly from efficient reusable lunar transportation.

b. For a minimum Mars program, consisting of perhaps a half-dozen landings of a few days' stay time each (that is what Apollo accomplished on the Moon), cryogenic all-

propulsive minimum-energy missions with multiple landers, e.g two or three per mission for two or three missions, are very attractive. This offers the opportunity to briefly explore six sites at minimum cost and minimum technology risk (Apollo explored six sites, spending a few days at each site). Carrying multiple landers per trip provides a desirable degree of rescue capability.

One concern with this implementation is that astronauts are committed to almost three years in space each Mars mission. There are significant issues regarding zero-g and cosmic ray exposure. While these concerns can presumably be dealt with, i.e. through zero-g countermeasures, artificial g, or suitable shielding, solutions may be costly in mass and complexity .

c. The performance potential of a nuclear thermal rocket (NTR) leads to less initial mass than cryogenic/aerobraking for most mission profiles. A nuclear rocket can eliminate the need for high-energy aerocapture at Mars; this is an important advantage. On the other hand, the development program for a nuclear rocket requires significant investment in effluent containment test facilities. Return on investment tradeoff of nuclear rocket versus cryogenic/ aerobraking at the median Mars activity level favored the nuclear rocket. If Mars exploration progresses to a permanently-occupied base, aerocapture and NTR are complementary technologies in the NTR-dash mode; this traded favorably versus nuclear electric propulsion in ROI analysis.

The nuclear thermal rocket improves mission flexibility and reduces constraints on mission profiles. A nuclear rocket is the most promising propulsion system for fast Mars trips (a year or less). Fast trips, however, are indicated as expensive in terms of total mass and hardware expended. While fast trips are technically interesting, they are probably not affordable in a space program with constrained funding.

d. We found that electric propulsion systems are suitable for Mars crew transportation if (1) operated from high-altitude nodes such as L2, or (2) boarded by the mission crew at about lunar distance, where the crew fly to the electric propulsion vehicle on a lunar transfer vehicle (LTV). Trip times are competitive with all but fast-trip split-sprint nuclear thermal rocket systems, i.e. about 450 days for nuclear electric propulsion (NEP) and about 550 days for solar electric (SEP). On conjunction fast transfer profiles, NEP delivers 150 to 200 day transfers each way and SEP about 250.

The inherently high reusability and low resupply mass of electric systems offers life-cycle cost advantages at high activity levels. Development cost for NEP and array production cost for SEP are major issues. Resolution of the array production cost issue will require a manufacturing technology program. Cost and return-on-investment results showed that estimated NEP development costs are not effectively amortized even at the settlement activity level when compared with a nuclear rocket operated in the dash mode. SEP, at current array costs (~ $1000 per watt), is estimated as more expensive to develop than NEP. SEP becomes very attractive at $100/watt, showing about 10% return on investment versus NTR at the median activity level. If a low-cost SEP is possible, it is also attractive for lunar cargo.

e. Special architectures offer unique advantages in particular circumstances. For example, lunar libration point staging is attractive for low-thrust systems because spiral operations out from and into Earth's gravity well can be conducted by an electric orbit transfer vehicle in parallel with interplanetary transfers by the interplanetary SEP or NEP.

Lunar libration point operations offer reduced Earth launch mass for cryogenic/aerobraking profiles through use of lunar oxygen (the return on investment in lunar oxygen facilities is not favorable), and to electric propulsion systems because the interplanetary vehicle need not execute low-thrust spirals out of and returning into Earth's gravity well. Neither of these potential advantages applies to nuclear thermal rockets; libration point operations for nuclear rockets were not considered.

Mars direct simplifies flight operations at Mars at the expense and risk of propellant production on Mars; it is more efficient than Mars orbit rendezvous in a crew transport mode after a base is established, but not as efficient as NTR-dash. It appears too risky (lack of abort modes) for an initial mission. Mars direct offers potential advantages where galactic cosmic ray concerns drive us to conjunction fast transfer profiles with long surface stays. It is not suitable for crew rotation and resupply of a permanent base because it is confined to the conjunction profile, and leads to gaps in crew presence at Mars.

Reusable MEVs using Mars oxygen, and methane or hydrogen if available, are interesting as an evolutionary development, mainly because their greater reusability may have significant life cycle cost benefit. In our settlement scenario analysis, the reusable MEV came on line too late to have a net payoff. This concept needs further evaluation.

Cyclers may be advantageous if interplanetary transfer habitats need extensive radiation shielding or if large crews and consequent massive transfer habitats are needed to satisfy mission objectives. Early in a Mars program, cyclers do not have enough advantage over simple conjunction all-propulsive (cryogenic or nuclear) or aerobraking to merit their need for infra-structure pre-positioning, operational complexity and give-up of abort modes.

1.8 Integration Compatibility and Summary of Level 2 Trades

Integration compatibility analyses were performed to ensure that conceptual designs of elements of the space transfer systems would work together and could be integrated with other system elements such as Earth-to-orbit launch, orbital assembly, and surface systems.

1.8.1 Launch Manifesting and Launch Site Operations

Launch manifesting analyses concentrated on a launch vehicle capability of 100 to 140 t. mass and payload shroud size of 10 m. diameter by 30 m. cylinder length. Larger shroud sizes were briefly investigated. All system elements could be manifested in the nominal shroud size. The principal problem areas were (1) large Mars transfer habitats used up most of the shroud volume but only about half the available lift capability; (2) Mars capture and landing aerobrakes, as designed, required two launches each because of packaging volume; these launches only used about 10% of the available lift capacity. The habitat problem is endemic because habitats are large low-mass pressure volumes. The situation can be improved somewhat and is not a major issue. The aerobrake problem is a function of the design. A different structural design approach has been identified that is expected to ameliorate the problem at least to where two aerobrakes can be launched in one ETO flight rather than two flights per aerobrake.

Launch site operations analyses were coupled with on-orbit assembly analyses to derive overall sequences that observe (1) the constraint of allowing interface verification for all hardware prior to launch, and (2) feasible assembly sequences. The principal effect on assembly sequence is that the Mars transfer crew habtitat is launched late in the hardware sequence (but before the major propellant launches) because of its multiple interfaces with other elements.

1.8.2 Space Assembly

Reviews and analyses of orbital assembly operations did not identify major problems.

a. Most of the proposed operations are simpler than those planned for assembly of Space Station Freedom and presumably would enjoy a greater experience base as well as more of an assembly facility than a shuttle Orbiter;

b. Opportunities to design vehicles for ease of assembly have only begun to be explored;

c. Benefits of very large launch vehicles appear not worth the cost. The point of diminishing returns seems reached at 100 - 120 t. payload and a 10-m diameter by 30-m shroud. Very large vehicles do not eliminate on-orbit assembly and certainly do not eliminate on-orbit launch processing, which is a more difficult issue.

Space assembly analysis concentrated on sequences and timelines. Pictorial representations were used for these analyses as necessary to understand the nature of specific operations. Most of these pictorials were sketches, not finished document art. We baselined mainly telerobotic assembly for four reasons:

a. The required telerobotic operations are not much more complex than those conducted by the shuttle RMS and those planned for the Space Station Freedom arm.

b. Crew time on orbit, and the costs for accommodations and transportation for crew, are very expensive.

c. A large share of these operations, e.g. those involving hardware loaded with propellants, must take place off Space Station Freedom for safety reasons. For these, telerobotic operations are much preferred. The requisite telerobots can perform the rest of the assembly tasks; little or no additional investment is required.

d. Many of the operations involve sizes and masses of equipment beyond the capability of EVA humans.

We presumed the use of EVA crew for contingencies and that test or mission crews would be on board the vehicle for critical all-up systems and prelaunch tests. In most cases, assembly timelines were constrained by the launch rate which assumed no more than one HLV per 90 days. In a few cases with the highly complex electric propulsion systems, assembly timelines used up the available time between HLV launches. Given the uncertainty in assembly timeline estimates, these cases represent potential assembly time problems.

1.8.3 Surface Systems

Relatively few surface systems interface requirements were available. Some of our lunar vehicles outgrew (in height) reasonable dimensions for compatibility with an overhead payload unloader. Therefore, we developed concepts for underslung-payload lunar cargo vehicles with payload clearance as large as 7 meters diameter. The Mars crew MEV has an integral surface payload that is not offloaded. This MEV has a performance capability to deliver 50 t. to Mars' surface, but an unloading concept was not developed.

1.8.4 Contiguous Crew Volumes

All final vehicle design concepts had contiguous crew volumes, e.g. between MTV crew habitat, MEV ascent cab, MEV surface hab, and ECCV.

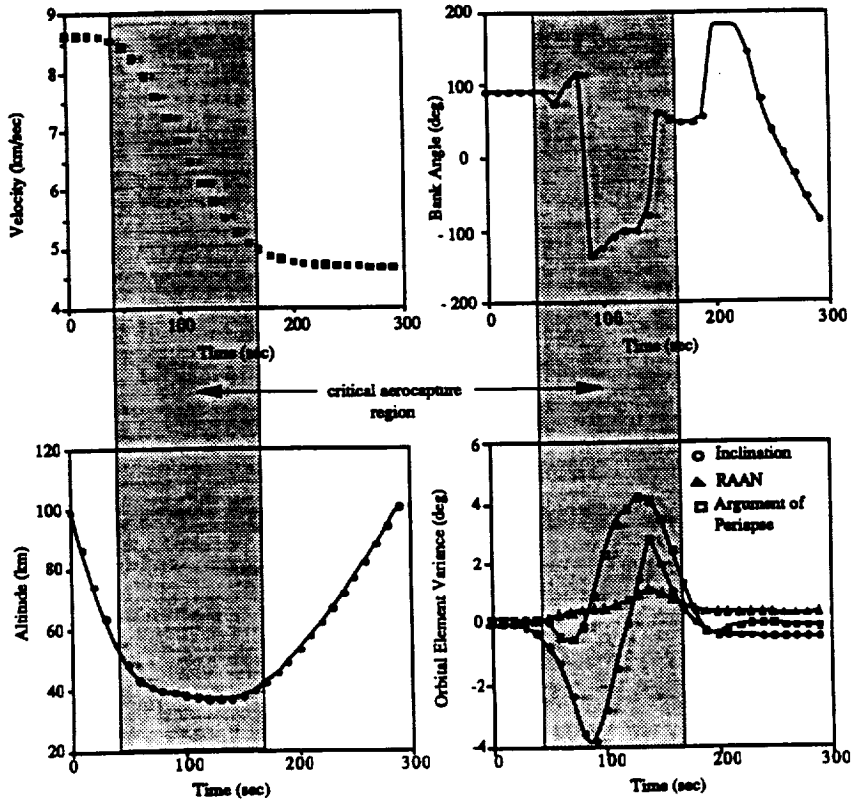
1.8.5 Aerocapture and Aerobraking Operations

Analyses were conducted for integration compatibility as well as for performance.

1.8.5.1 Trajectory Generation

Capture L/D was set at 0.5 for most of the studies. This appeared adequate. Guided trajectory simulations were conducted to ascertain performance requirements on the MEV to rendezvous with the MTV after aerocapture. Typical results are shown in Figure 1-32. Development of a cryogenic/aerobraking configuration that captured both vehicles behind a single brake proved unsuccessful. The implication of separate captures is that the lines of apsides of the capture orbits must be controlled so that post-capture rendezvous is possible. This proved to be the most difficult challenge for the capture GN&C system. Post-capture

delta V of about 150 m/sec was indicated by our simulations. We believe further work can reduce this requirement to less than 100 m/sec.



TYPICAL GUIDED TRAJECTORY AEROCAPTURE STATES
FIGURE 1-32

1.8.5.2 Pinpoint Landing

Pinpoint landing is a requirement for any LEV or MEV returning to a specific base or going to a particular exploration site. Pinpoint landing on the Moon was demonstrated on the Apollo program and is not seen as an issue. During the present study, we simulated pinpoint landings using a trajectory design tool, with the result that Mars atmosphere density does not have much effect on site accessibility. This does not demonstrate that the GN&C problem is solved, but it appears much less difficult than aerocapture. We found that landing L/D 1 is needed for pinpoint landing unless the landing site is close to the (presumed elliptic) parking orbit periapsis latitude. High- thrust space transfer systems benefit from elliptic parking orbits; these orbits are usually optimized for each opportunity and are different from one to the next. Low-thrust systems can access any parking orbit and presumably do not need high L/D landers for pinpoint landing.

1.8.5.3 Forebody Aeroheating

Forebody heating analyses were conducted for aerocapture, landing and Earth return for a range of L/D from 0.25 to 1.0. Typical results are shown in Figure 1-33. Principal conclusions were:

a. Mars aerocapture heating is severe above arrival C3 about 30 km^2/sec^2 (entry velocity about 7.4 km/sec). Most opposition profile trajectories yield C3 about 50, thus presenting a challenge to aerobrake TPS design.

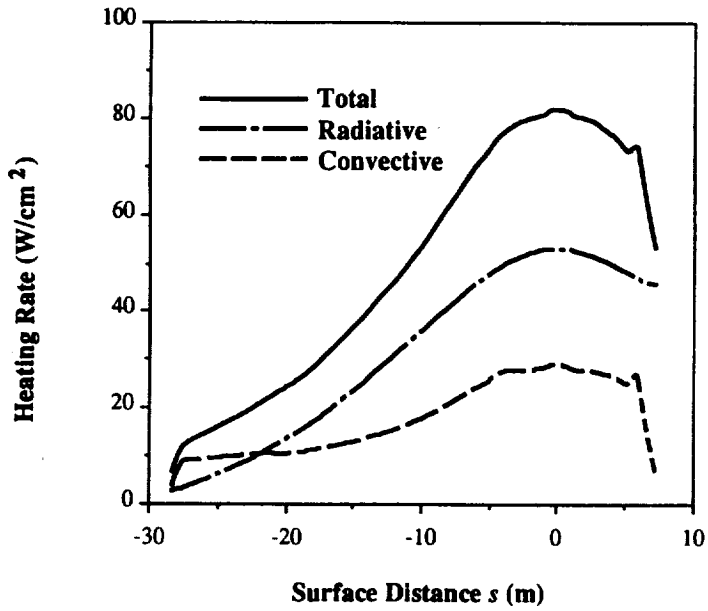
b. Much of the heating, especially at higher velocities, comes from radiation. Thus, the higher L/D shape, which has a much smaller nose radius, creates a lesser overall TPS materials challenge.

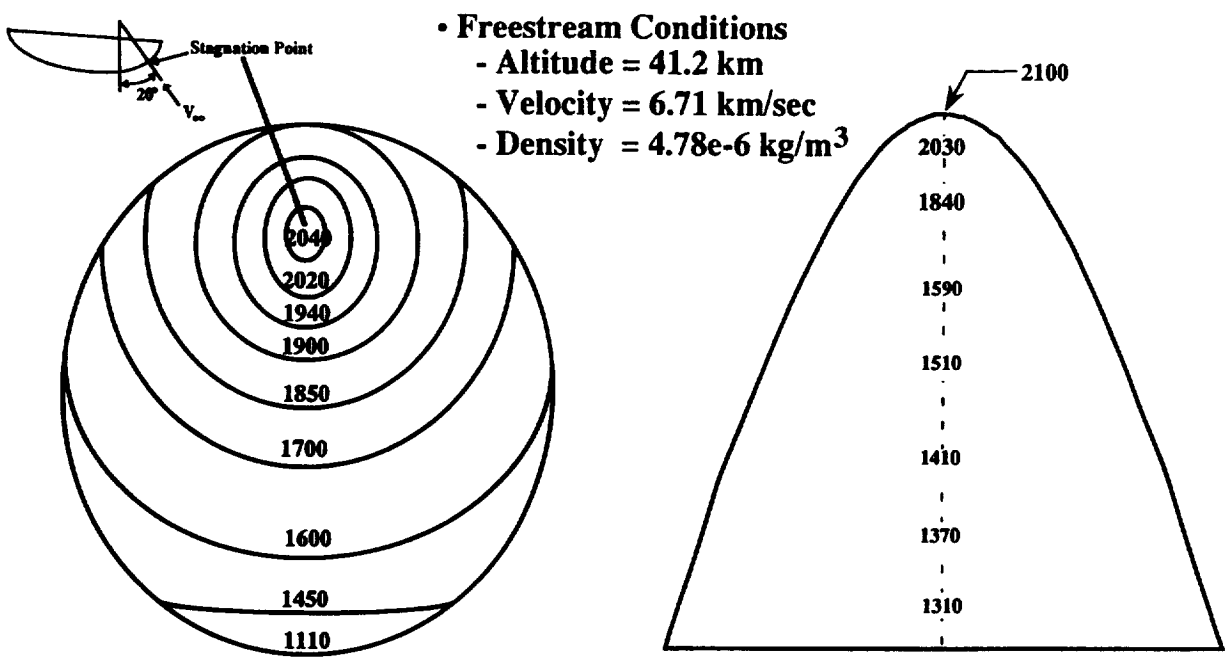
c. Aeroheating for Mars landing from orbit is not severe and can probably be handled by a "warm structure" system.

d. Aeroheating on a vehicle-scale aerobrake for lunar return is within, or nearly within, the capability of shuttle tiles.

e. Aeroheating on a vehicle-scale aerobrake for Mars return is similar to that for Mars arrival at C3 50. Since this would be a second use for a Mars aerobrake, it presents significant materials challenges.

f. Aeroheating for crew-only capsules at Earth return can be handled by ablators. Except for fast-trajectory Mars returns, the ablator state of the art is generally adequate.





MARS AEROCAPTURE AEROHEATING: (TOP) MAX FOREBODY HEATING RATE, (BOTTOM) TEMPERATURE IN K FOR L/D 0.5 AND 1.0 VEHICLE

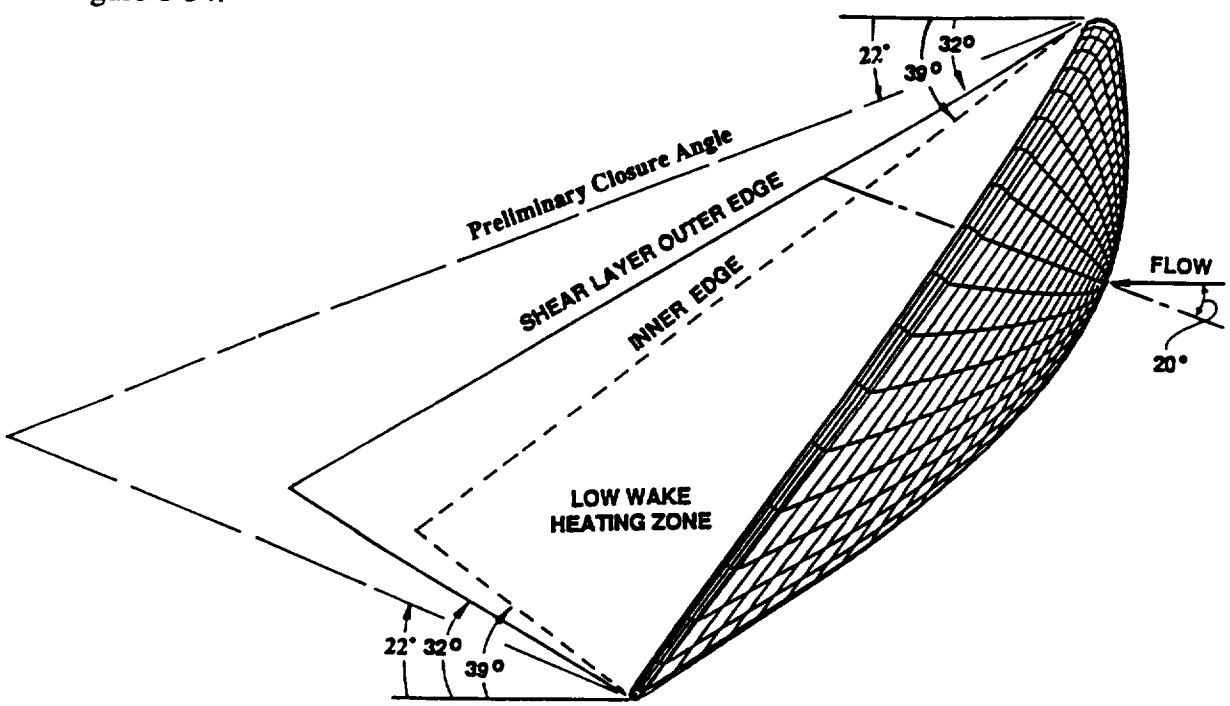
FIGURE 1-33

1.8.5.4 Wake Aeroheating

Wake heating analysis was conducted for one Mars aerocapture configuration and case. Conclusions were:

a. Wake heating in the "protected" zone is on the order of 3 W/cm^2. Modest thermal protection such as NOMEX felt should be adequate.

b. The "protected" zone is smaller than previously estimated as illustrated in Figure 1-34.



WAKE CLOSURE ZONE FOR L/D = 0.5 AEROBRAKE
FIGURE 1-34

1.8.6 Earth Return

The return to Earth should always be by crew capsule (ECCV). Even if the vehicle returns to Earth orbit by aerocapture, the Space Station Freedom orbit is normally accessible only after a phasing orbit wait of weeks (line of nodes problem). Lunar returns can be conveniently phased at the Moon but the time is still required. Selected lunar trajectory timing can avoid wait periods but this places another constraint on lunar mission timing and is therefore undesirable.

1.8.7 Summary of Level 2 Trades

These trades were conducted to support NASA Lunar/Mars Exploration Project Office activities. A thorough report on these trade studies and their results is contained in the two-book Implementation Plan and Element Description Document Volume I: Major Trade Studies.

1.8.7.1. Lunar Excursion Vehicle/Mars Excursion Vehicle Commonality

Since the LEV and MEV are about the same size, with about the same propellant load and perform similar funciotns, although on different planets, a trade was performed to identify the best level of commonality. The result: complete vehicle-level commonality seriously compromises both vehicles because of differences in requirements, e.g. the MEV has an aerobrake while the LEV does not. Very high subsystems commonality is recommended; same engines, same propulsion system components, largely common avionics and power, same crew modules (assuming crew sizes are the same -- if not, common crew module subsystems and components). This commonality approach was reflected in development cost analyses.

1.8.7.2 Alternative Crew Modules

This was not a trade study but a parametric data base for LEV, LTV and MTV crew modules and ECCVs for a range of crew sizes and occupancy durations. External dimensions and masses were provided.

1.8.7.3 Habitat Trade

This trade addressed the Mars transfer vehicle long-duration crew habitat. Crew sizes from 4 to 12, durations from 100 to 1000 days, and various configuration arrangements including unitary and clustered modules were evaluated. A broad spectrum of criteria was applied. Unitary modules were preferred over clusters because of mass and crew amenities advantages. The best diameter (of three evaluated: 4.4 m, 7.6 m, 10 m) was 7.6 meters except for a crew of 12, where a 10-m diameter module is slightly better. Longitudinal rather than transverse internal arrangement was preferred. An internal pressure bulkhead provides redundant pressure volumes. Parametric data on mass versus crew size and duration were developed.

1.8.7.4 Radiation Assessment

Crew radiation hazards were evaluated for Mars missions. A storm shelter in the MTV habitat is necessary to provide adequate protection from solar flares (well-known result). On all Mars profiles except conjunction fast transfers, predicted crew dose from galactic cosmic rays exceeds current exposure standards unless additional shielding is provided (relatively new result, not reflected in "90-day Study"). The conjunction fast transfer only provides relief from the cosmic ray problem if the crew spends nearly all the Mars vicinity time on the surface where the atmosphere provides shielding. Mass impact of the solar flare storm shelter (small volume) is minimal, while the mass impact of cosmic ray shielding is expected to be significant. Cosmic ray shielding requirements need further evaluation.

1.8.7.5 Large Crew Size

Crew sizes up to 32 were examined for impact on the study Mars architectures. The nuclear propulsion systems adapted to large crews better than non-nuclear systems.

1.8.7.6 Rescue/Abort

A semi-quantitative mission risk analysis described earlier (rigorous method; best-judgement reliability numbers) was applied to several of the Mars mission profiles. The principal driver on likelihood of crew loss was critical maneuvers without abort or rescue modes, for example Mars ascent with only a single Mars Excursion Vehicle on the mission. Mars mission profiles should be designed with careful attention to abort and rescue modes, especially where these can be provided at modest cost.

1.8.7.7 Mars Transfer Systems

The results of this trade are shown earlier in this executive summary as results of architectures analysis.

1.8.7.8 Artificial Gravity

Artificial gravity configurations were developed for the major propulsion options. Mass

estimates and complexity comparisons were made. The cryogenic/aerobraking system incurs a large complexity penalty for a tether system, but modest (~10%) mass penalty. The nuclear rocket incurs least penalties in both areas - mass a few percent and very little complexity penalty since artificial gravity is accommodated by making the central truss longer and rotating the vehicle end-over-end when not thrusting. The nuclear and solar electric systems use a rotating joint. The nuclear electric system uses the reactor/power generation assembly as counterweight. We evaluated using the entire solar electric vehicle as counterweight in an "eccentric rotator" configuration, but doubt this will really work. Consequently, the nuclear electric mass and complexity penalties are modest, while the solar electric mass penalty appears large since we presently know of no reasonable solution except to employ an inert counterweight of mass about equal to the long-duration crew module.

1.8.7.9 Operational Orbit Selection

High thrust systems need the operational Mars orbit to be tailored to each mission for minimum performance penalty. Low-thrust systems adapt well to any chosen Mars parking orbit. Attempts to find a "standard orbit" have so far been unsuccessful. The only architecture that benefits from a standard orbit is the full-cycler system, which needs a node in Mars orbit.

Our recommended lunar architecture employs a direct mode with no lunar orbit operations.

1.8.7.10 MEV Propellants

The predicted level of cryogenic insulation performance leads to a preference for hydrogen/oxygen propellants for surface stay times to 600 days. Storable propellants result in greater IMLEO, but the penalty can be accepted. If storables are used, metal-gel fuels and/or pump-fed storable engines are strongly recommended. Methane/oxygen was not a winner in any of the cases, but may be beneficial for Mars atmosphere in-situ propellant in view of simpler production and liquefaction processes.

1.8.7.11 Aerocapture GN&C

We conducted extensive simulations of aerocapture, including trajectory design and two

types of GN&C simulations, with a range of Mars atmosphere models including MARS-GRAM and various synthetic models of atmosphere irregularities. The result was that adaptive guidance systems are indicated as able to deal with likely Mars atmosphere uncertainties with L/D in the range 0.5 to 1.0. A caution: present estimates of Mars' atmosphere uncertainties are little more than guesses. Precursor missions, at least Mars Observer and probably additional measurements on later missions, will be required to develop high confidence in the engineering practicality of Mars aerocapture.

1.8.7.12 Structures and Materials

High performance structures and materials have great payoff in view of the difficult performance requirements of SEI missions. This is especially true in aerocapture and aerobraking, but also important in vehicle connective structure such as intertanks, thrust structures, and landing legs. Routine operational use of aerocapture on Mars missions will also require major improvements in aero-heating TPS materials, e.g. from 1800K to 2500K.

1.8.7.13 Landing Site Access

Tailored Mars parking orbits (for high thrust transfer systems) can have periapsis location at latitudes far from the equator, e.g. ±40°. Depending on the direction of flight at periapsis, significant cross-range can be needed to reach surface sites within an expeted range of requirements. No site accessibility requirements have been specified. We assumed a requirement to reach any latitude within ±20° of Mars equator from any likely parking orbit. The preliminary result is that a lander L/D at least 1 is needed. Further work is required to address avoidance of crossing the terminator into the night side of Mars, and to examine a broader range of parking orbits.

Low-thrust systems can enter parking orbits tailored to landing site access; little cross-range is required. Surface site access for the Mars direct and NTR-dash modes remains to be investigated; the access problem is believed to be less severe than for elliptic parking orbits for high-thrust systems.

The lunar direct mode recommended in this report can reach any lunar surface site.

1.8.7.14 High L/D

We examined blended wing and bent biconic shapes for high L/D aerocapture and landing. The blended wing shape is preferred in a scenario where on-orbit assembly is permitted. If on-orbit assembly is not available, as might be the case for a Mars precursor mission, the bent biconic provides roughly equivalent performance at slightly greater mass, and presents more difficult packaging problems.

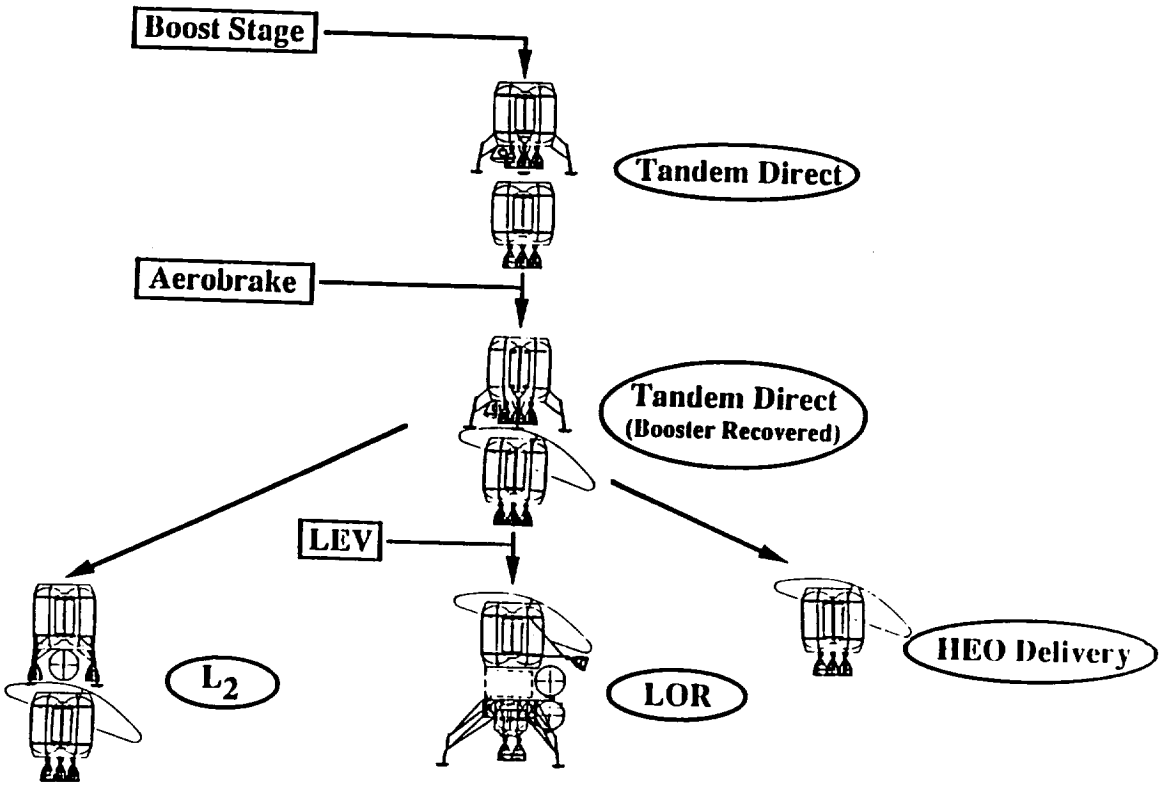
1.9 Conclusions and Recommendations

The real SEI program will not follow any particular long-range scenario that can be defined now. Clearly, an ambitious program can start small and evolve. The potential for evolution to a larger scale must be included in technology selection decisions for small and moderate-scale programs. Programs of different scale at the Moon and Mars are logical. Lunar and Mars exploration will probably start with modest objectives and evolve as a result of early mission achievements and scientific findings. An ambitious lunar program will significantly defer human Mars missions under likely funding constraints. Similarly, an ambitious Mars program will limit lunar activities to high-priority scientific objectives. As more is learned about the benefits of lunar and Mars activities, shifts in emphasis are likely. The recommended SEI program strategy is evolutionary and keeps options open. With these considerations in mind, seven specific recommendations are presented below.

1.9.1 Specific Recommendations:

a. Begin the manned lunar program with a tandem-direct expendable system, Figure 1-35. The benefits are:

(1) The system can be designed to eliminate on-orbit assembly. One docking or berthing is required.



LUNAR TRANSPORTATION FAMILY EVOLUTION
FIGURE 1-35

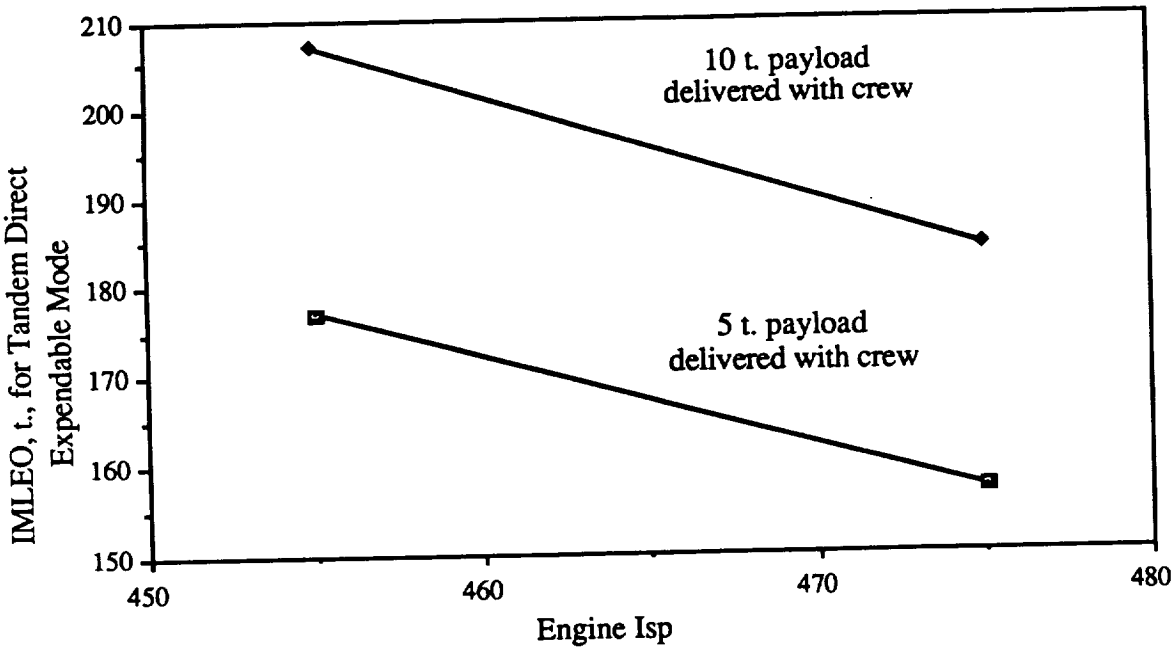
(2) The number of development projects is minimized compared to any other architecture thus far identified. This minimizes front-end cost, and offers reasonable expectation of manned return to the Moon by 2004 under likely funding constraints.

(3) Flight mechanics constraints for LOR operations are avoided.

(4) The tandem-direct LTV is a starting point for evolutionary paths to all other identified lunar architectures.

(5) A lunar aerobrake can be tested on the unmanned booster stage of the tandem direct system, accomplishing flight testing on a stage otherwise expended, at little performance cost, without risk to the crew.

b. Invest in cryogenic storage and management technology and in a 30K-class advanced expander cryogenic engine with 10:1 or better throttling capability, Figure 1-36. These activities are candidates for advanced development. Benefits are:



LUNAR MODE SENSITIVITY TO ISP
FIGURE 1-36

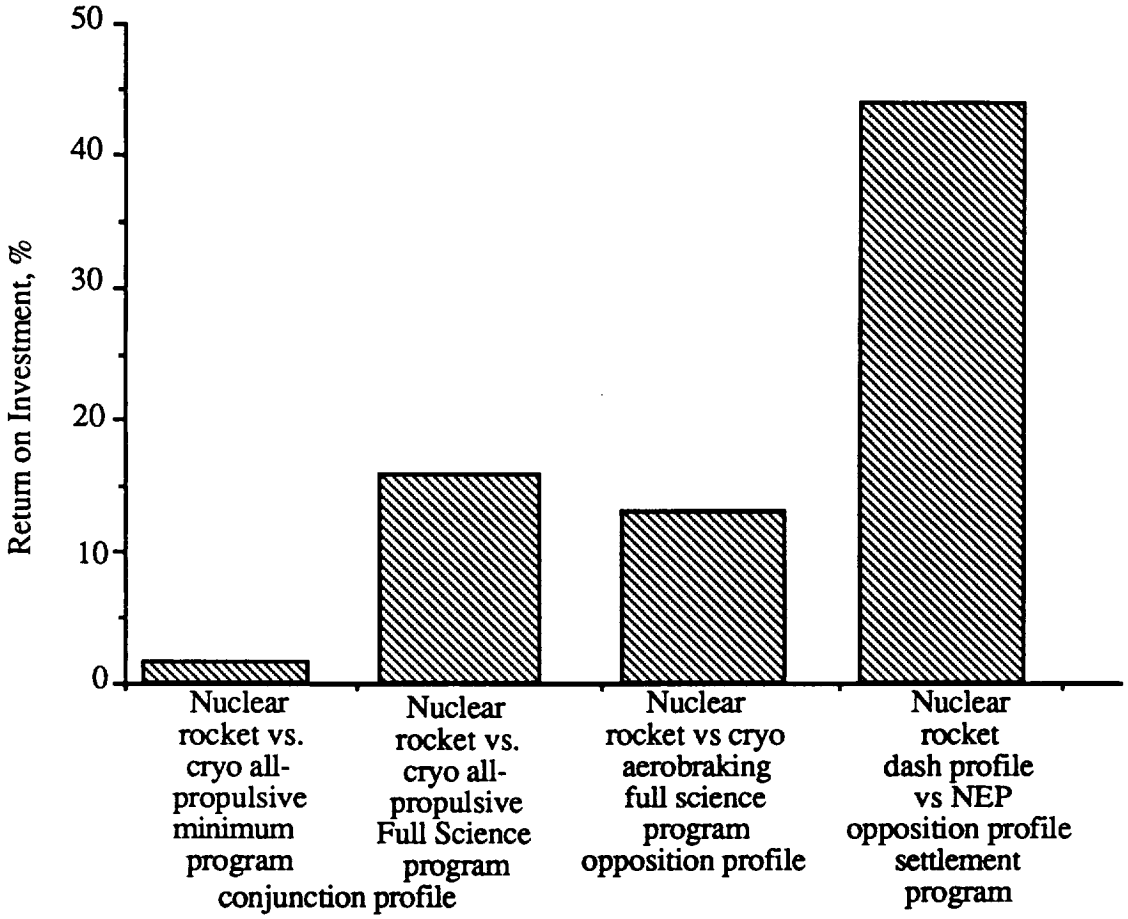
(1) Without advanced development of a low-boiloff flight-weight cryogenic insulation system, the lunar program may be forced to select a storable propulsion system for lunar vicinity operations. The cost impact is many billions of dollars.

(2) An advanced expander engine offers about 20 seconds' Isp gain over a modified RL-10, and can incorporate and demonstrate advanced health monitoring and maintainability features essential for Mars missions.

c. Baseline nuclear thermal rocket propulsion for Mars. Initiate a technology advancement program with emphasis on high-performance fuels and full-containment ground test facilities. Benefits are:

(1) The nuclear thermal rocket is indicated as the most economic and flexible Mars transfer propulsion system over a wide range of program activity levels, Figure 1-37.

(2) Nuclear rocket vehicle performance is sensitive to specific impulse. The Isp gain for carbide fuels is well worth the technology investment.



ROI COMPARISON FOR NTR

FIGURE 1-37

(3) Development and qualification testing of a nuclear rocket will require a proven test facility technology that fully contains, and removes radioactivity from the hydrogen working fluid.

(4) Nuclear rocket performance enables a modest lunar program and significant Mars exploration with about six launches per year of a 100-tonne class HLLV.

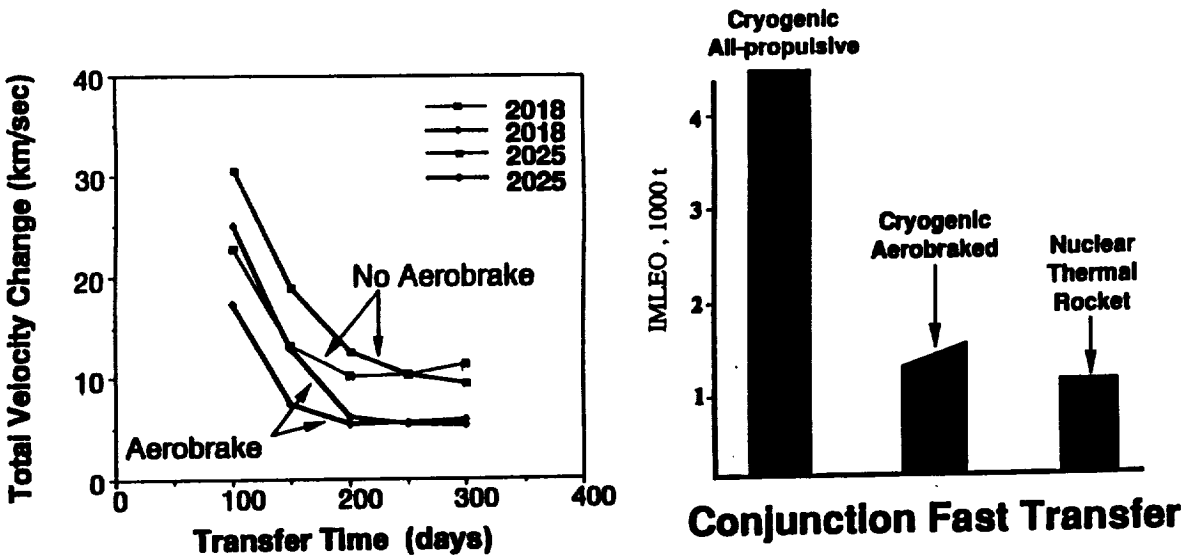
(5) The nuclear rocket baseline offers reasonable expectation of an initial manned Mars mission by 2010, under likely funding constraints, assuming a modest lunar program.

d. Accelerate aerobraking technology for Mars aerocapture as a backup to the nuclear rocket, targeting a decision between the two in the 1996-2000 time frame. Benefits are:

(1) There is enough doubt regarding NTR performance and cost, especially for test facilities and testing, that a backup is appropriate at this time.

(2) Aerobraking is needed for Mars landing. While the technology challenges are less daunting than for aerocapture, they nonetheless merit a technology program.

(3) Aerobraking technology keeps other options open, including Mars direct, cycler orbits, and the NTR-dash profile identified as one of the most promising in the event Mars exploration advances to a need for a crew rotation and resupply mission profile, Figure 1-38.



MARS AEROCAPTURE BENEFIT FOR CONJUNCTION FAST TRANSFER
FIGURE 1-38

(4) Aerobraking is economic for lunar transportation at a lunar flight rate of two or more per year.

e. Perform aerobrake tests on the LTV booster, to put the technology on the shelf for Mars application. Benefits are:

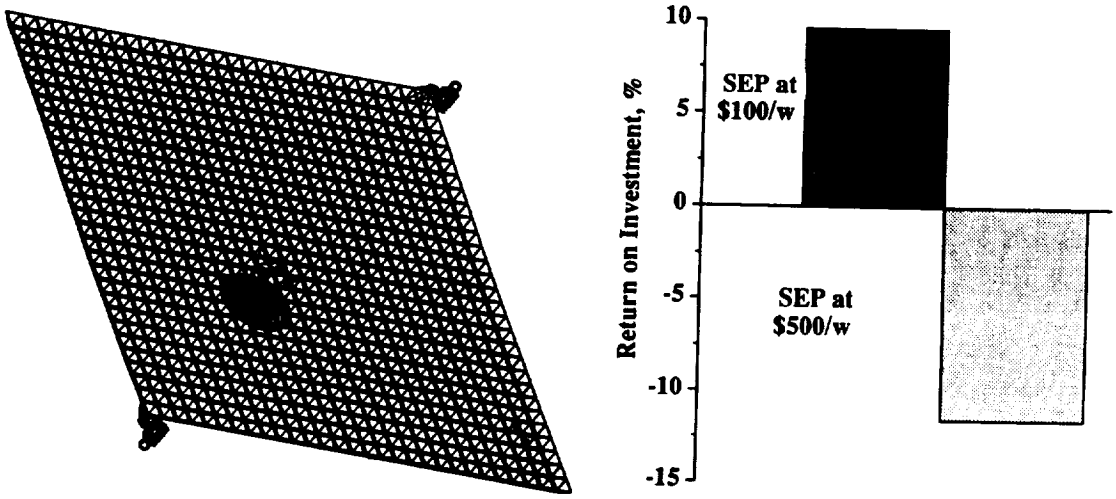
(1) If the lunar program grows to high activity levels, the lunar aerobrake is economically justified.

(2) A space-assembled aerobrake is needed for Mars landing.

(3) Assembly of a lunar aerobrake appears to be relatively straight forward.

(4) Aerocapture technology is needed as a backup to the selected preferred Mars architecture, and is also needed for the NTR-dash mode recommended for Mars base crew rotation and resupply.

f. Designate solar-electric propulsion as a "dark horse" for Mars transportation, and conduct a technology advancement effort aimed at removing the barriers to a high-performance, economic SEP system Figure 1-39. The targets are: (i) light weight, high performance arrays with improved radiation hardness; (ii) automated production technology capable of producing the arrays at about $100/watt; (iii) robotics technology capable of constructing the large SEP space frame truss and deploying the arrays and power collection and processing systems; and (iv) long-life, high power density, efficient electric thrusters. Benefits are:



ROI COMPARISON SEP AND NTR FOR FULL SCIENCE PROGRAM
FIGURE 1-39

(1) If safety considerations preclude operation of nuclear propulsion systems in low Earth orbit, SEP is the only option offering significant economies over cryogenic/aerobraking.

(2) If the low-cost array target is achieved, SEP has a large economic advantage over nuclear electric propulsion.

(3) SEP is the most likely architecture (of those presently identified) for eventual private sector use for Mars settlement.

(4) SEP technology has possible derivative benefits, e.g. power beaming to planet surfaces.

g. Continue the present emphasis of the nuclear space power program on near-term systems applicable to planet surface power, but augment with (i) further studies to better understand the probable cost of nuclear power systems suitable for electric propulsion, and (ii) modest funding of high-leverage high-performance power conversion technology. Benefits are:

(1) The DDT&E and production cost estimates made in this study, high enough to eliminate nuclear electric propulsion as a top contender, are clearly very preliminary. As NEP systems are better understood, cost estimates may come down (see Figure 30).

(2) High-performance power conversion technology is necessary to keep the option open. High-leverage technology advancements can be made with modest funding.

1.9.2 Indications for Lunar Architectures

Lunar transportation operations include those serving lunar missions, and for some architectures, those serving Mars missions. The L2 libration point as a Mars node offers performance advantages for low-thrust systems and for cryogenic systems using lunar oxygen, but does not have a return-on-investment advantage. For NEP and SEP the Moon itself is not involved; the L4 or L5 points, far from the Moon, may serve about as well.

However, if high-thrust systems are used for LEO-libration point operations, availability of lunar gravity assist favors L2 over the others. The L2/lunar oxygen option for Mars involves the Moon as a source of propellant.

Considering all the candidate architectures, there are seven relevant lunar mission modes; the first three apply to all Mars architectures and the last four only to those involving the L2 node. With suitable tailoring of requirements, all modes can be served by two vehicle sizes, an LTV-like vehicle at about 110 to 130 t. propellant load and an LEV-like one at about 25 to 30 t. propellant load. Modes staging at low lunar orbit need one of each size. Direct and L2 modes to the lunar surface use the larger size; crew missions to L2 need only the smaller size. Considering all modes, both vehicle sizes need to be compatible with an aerobrake and with lunar landing.

The maximum lunar surface payload delivery capability occurs with the tandem LTV direct mode. The booster LTV can be recovered in LEO with very little penalty; the upper stage LTV stays on the lunar surface with the landed payload. The payload capability of this mode is about 45 t., enough to land a complete integral "campsite" lunar surface habitat, suitable as an overnight shelter for a crew of four.

An LTV mode staged at L2 is moderately efficient for lunar oxygen delivery to L2. About 65 t. to LEO serves to deliver 95 t. of lunar oxygen to L2, with ordinary rocket propulsion; no mass driver or other exotic device is needed. This leverage for oxygen delivery makes the L2 node attractive for Mars missions; without lunar oxygen, the L2 node is not performance-wise competitive.

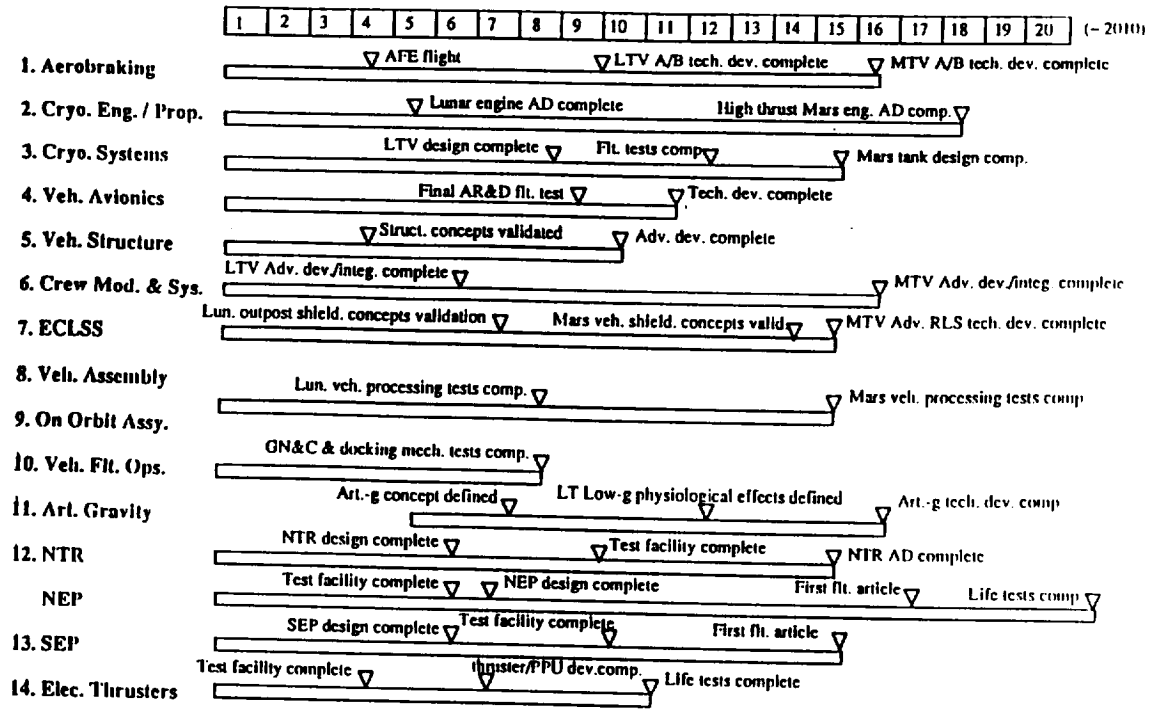
1.9.3 Technology Advancement and Advanced Development

Our current recommendations for technology advancement and advanced development, with schedules and funding estimates are shown in Figures 1-40 and 1-41. The funding level averages about $300 million per year. If we consider the median (full science) program as representative, the technology/advanced development program is about 2% of the life cycle cost of the program to 2025, a modest investment.

Technology Category	1	2	3	4	5	6	7	8	9	10	11	Total
1 - Aerobraking* -Technol.	1	6	10	5	5	8	10	10	8	5		68 M
- Adv. Dev.	0	0	30	30	55	20	30	65	65	40	30	400 M
2 - Cryogenic Engines / Prop.	0	30	30	30	20							110 M
- Adv. Dev.	0	0	99	71	65	50	65	65	50			465 M
3 - Cryogenic Systems - Tech.	5	5	5	5								20 M
- Adv. Dev.	0	10	10	20	50	50	50	110				300 M
4 - Vehicle Avionics/Software	2	5	5	5								17 M
- Adv. Dev.	0	0	0	25	45	40	40	40	40	40		270 M
5 - Vehicle Structures - Tech.	3	7	5	5	7	7	5					39 M
- Adv. Dev.	0	0	15	17	11	10	15	15	15	10		108 M
6 - Crew Modules & Systems	0	0	3	3	3	5	5	5	3			27 M
- Adv. Dev.	0	0	15	20	10	10	15	20	20	10		120 M
7 - Environ. Ctrl. & Life Supp.	0	0	3	5	5	10	10	5	5			43 M
- Adv. Dev.	0	0	6	10	10	15	30	30	30	20		151 M
8 - Vehicle Assembly - Tech.	5	5	5	5								20 M
- Adv. Dev.	0	5	40	40	40	40	40	40	10			255 M
9 - Orbit Launch & Checkout	5	5	5	5								20 M
- Adv. Dev.	0	4	15	16	5	10	10	10	10	5		85 M
10 - Vehicle Flight Operations - Adv. Dev.	0	0	9	15	10	15	15	15	10	5		94 M
11 - Artificial Gravity - Tech. - Adv. Dev.	0	0	0	2	5	10	10	10	10	3		50 M
12 - Nuclear Propulsion												
NTP -	0	10	15	20	20	20						85 M
NEP -	0	15	20	30	30	30	20	20				165 M
13 - Solar Electric Ion Prop.	2	8	10	15	15	10						60 M
Array manufac. Tech. -	0	0	0	30	30	30						90 M
14 - Electric Thrusters	0	5	10	20	20	20	10					85 M
Tech. Development Total	23	120	365	449	461	410	380	460	276	138	30	3112 M

TECHNOLOGY/ADVANCED DEVELOPMENT FUNDING ESTIMATES

FIGURE 1-40



TECHNOLOGY DEVELOPMENT SCHEDULES OVERVIEW
FIGURE 1-41

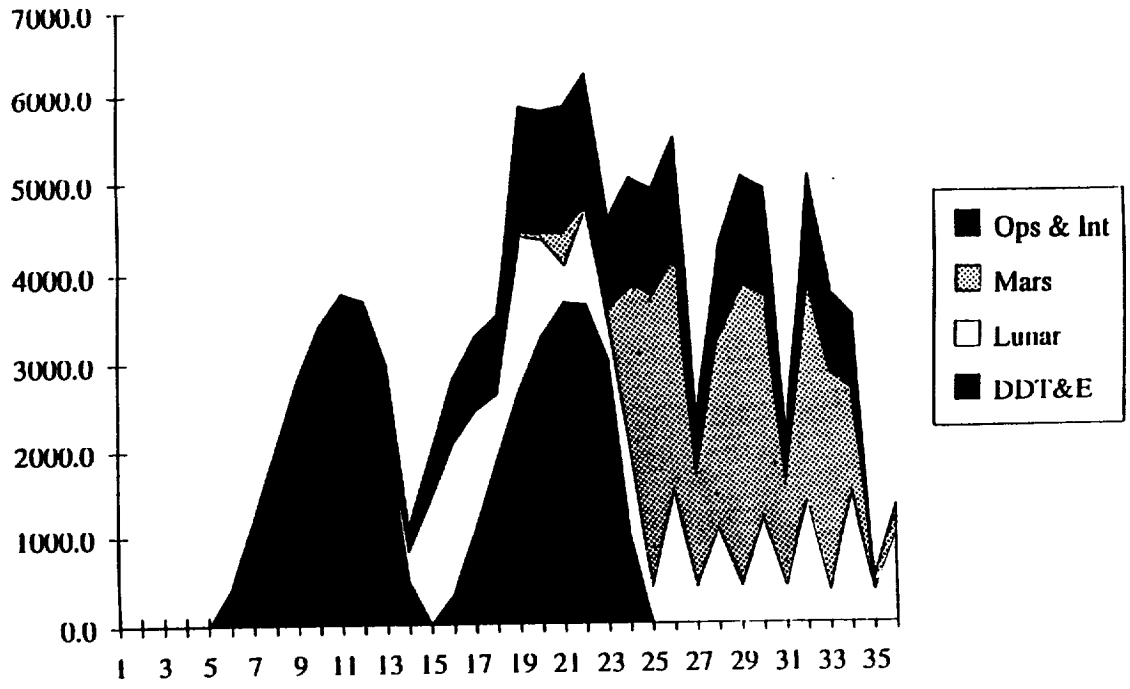
1.9.4 Minimum Program Life Cycle Cost Spread

The minimum program life cycle cost spread peaks between five and six billions per year, Figure 1-42. The deep valley between lunar and Mars peaks indicates that the Mars program should occur earlier in this program. The minimum program involves relatively modest investments in surface systems and falls well below the SEI funding wedge implied by the Augustine Committee recommendations (Reference 17).

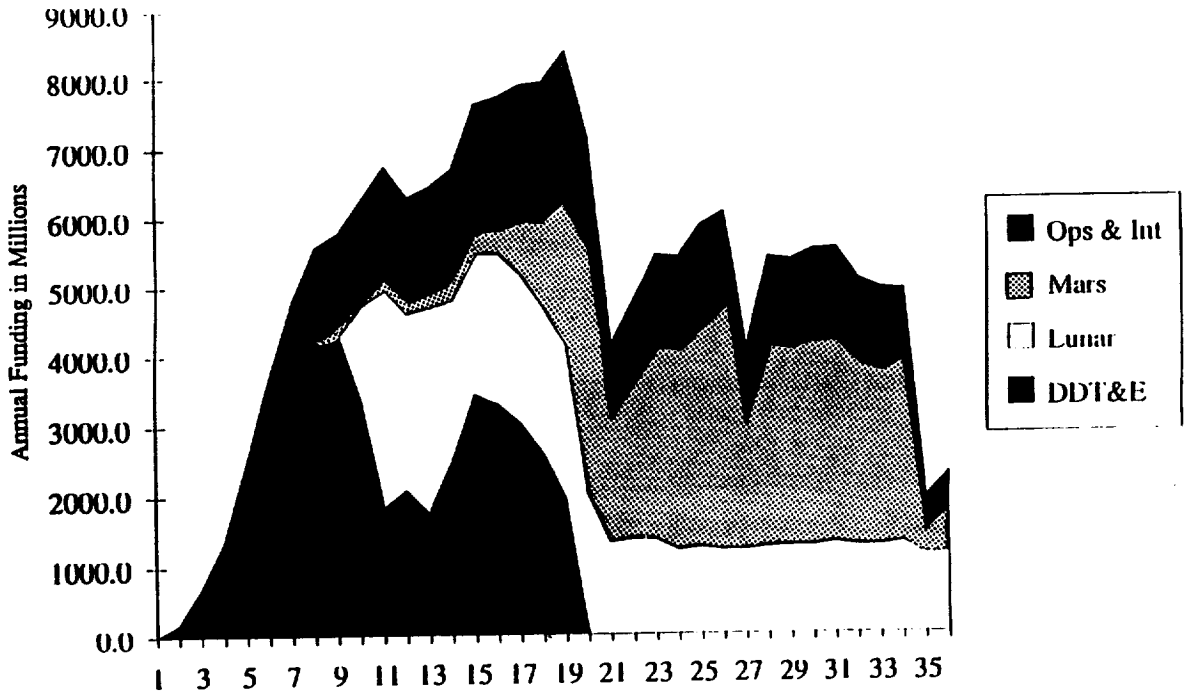
1.9.5 Median (Full Science) Program Life Cycle Cost Spread

The median life cycle cost spread peaks at about eight billions per year, Figure 1-43. With addition of likely surface systems costs, this program probably exceeds the Augustine guidelines during the peak years. The median program has the capability of providing a science and exploration output many times that of the minimum program. Lunar human presence grows from an occasional 45 days to permanent presence of six people, and Mars surface time grows from about four man-years to about 30. In other words, a roughly 50% increase in cost leads to about an order of magnitude increase in exploration and science potential.

C-2



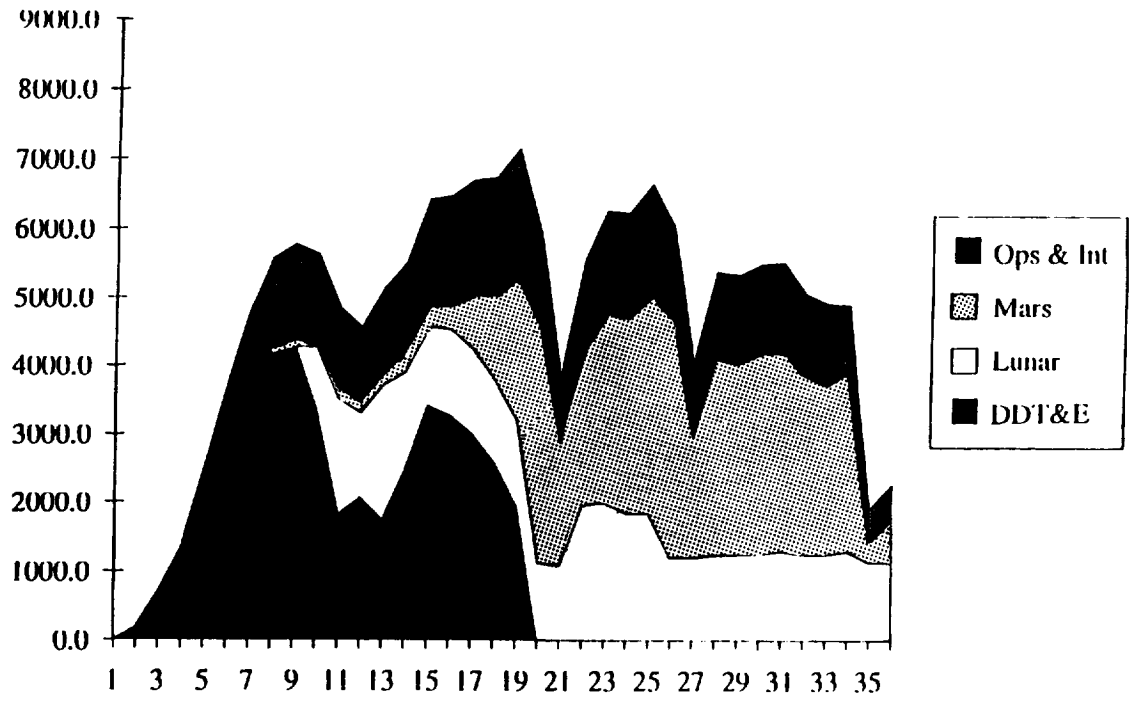
MINIMUM PROGRAM FUNDING ESTIMATE
FIGURE 1-42



FULL SCIENCE PROGRAM FUNDING ESTIMATE
FIGURE 1-43

1.9.6 Median (Full Science) Program Life Cycle Cost Spread for Reduced Early Lunar Program

By deferring major lunar activities, the median program is brought within the Augustine guide lines, Figure 1-44. Permanent human lunar presence is delayed until after the Mars DDT&E peak. The early lunar program is like the minimum scenario, i.e. man-tended astrophysics observatories.

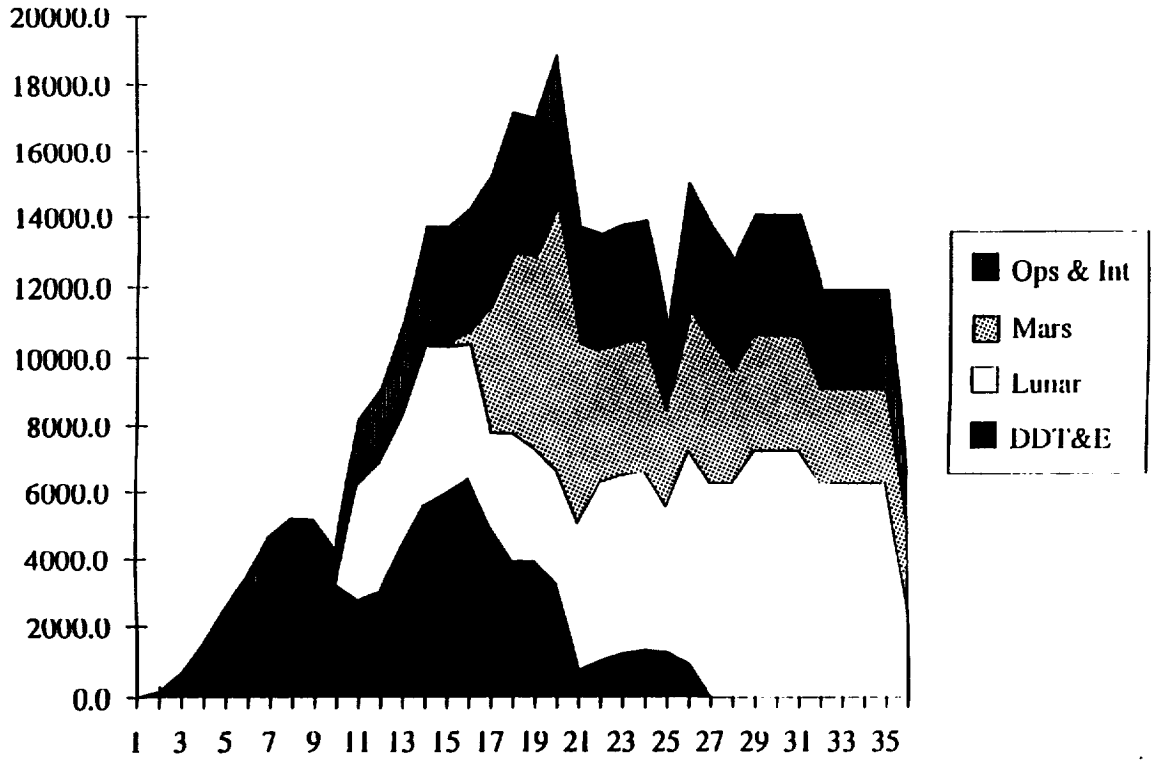


FULL SCIENCE PROGRAM FUNDING ESTIMATE WITH REDUCED LUNAR
FIGURE 1-44

Another way to level the funding profile for the median program is to defer Mars by a few years. The reference median program achieves a Mars landing in 2010 (2009 departure). Deferral to about 2016 would probably smooth out the funding profile much as did the reduction of the early lunar program. Our view was that getting to Mars early was more important than an early buildup to permanent lunar presence. The partially deferred lunar program still achieves astrophysical observatories early, but defers permanent human presence until after the major Mars mission DDT&E is complete.

1.9.7 Industrialization and Settlement Cost Spreads

Our maximum scenario involved simultaneous industrialization of the Moon and progress towards settlement of Mars, Figure 1-45. As the cost spread shows, this is clearly beyond the funding levels recommended by the Augustine Commission. Both premises of this scenario (i.e. industrialization and settlement) suggest significant private sector involvement. An additional investment on the order of $100 billions over about 20 years builds upon a plausible public-sector program of science and exploration to create a program involving the private sector for industrialization and settlement. This amount is more than the private sector investment in the Alaska oil pipeline by a factor of three to four, and less than the private investment in oil supertankers since the closure of the Suez Canal.



SETTLEMENT/INDUSTRIALIZATION PROGRAM FUNDING ESTIMATE

FIGURE 1-45

The economic potentials of lunar and/or Mars industrialization and settlement are presently not at all understood. We have made an effort at estimating the costs. We have little idea as to the eventual payoffs.

1.9.8 Open Issues

It is not presently possible to provide a comprehensive list of remaining architectural issues. Those that emerged in the last six months of study are:

a. The risk versus cost tradeoff for Mars trip time: How much should we be willing to pay to reduce trip time?

b. The cost and risk tradeoffs for the Mars direct and cycler architectures; these need further analysis and evaluation.

c. Importance of galactic cosmic ray radiation dose to overall mission risk and the best engineering solutions to controlling radiation dose risk, i.e. shielding versus reduction of transfer time.

d. Full understanding of the nature, cost and facilities for vehicle assembly and turn-around operations in low Earth orbit, and how to minimize impact of these factors by proper design of the elements of transportation architectures.

e. Mission profile, design and operational strategies to deal with the problem of Mars opportunity launch windows.

1.9.9 Conclusions

The basic in-space transportation architectures for Mars, i.e. cryogenic/aerobraking, nuclear thermal rockets, and nuclear and solar electric propulsion, are all competitive under plausible circumstances.

Before ultimate selections among these architectures can be made, better definition is needed as to the nature and activity level of the lunar and Mars exploration and development programs that will be a part of the Space Exploration Initiative. We also need to conduct continuing design, performance, operations, cost, and architectural studies as well as technology advancements to better define technical performance capabilities, risks, operational problems, and development and operations costs.

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2. CONCEPTS & EVOLUTION

2.1 Introduction

2.1.1 Three SEI Program Scales

The Space Transfer Concepts and Analysis for Exploration Missions (STCAEM) study partitioned the field of possible implementations of SEI into three representative scales, as explained in Section 1: (1) a minimal-scale program which meets the "lunar stay" and "Mars mission" requirements specified by the President in 1989; (2) an aggressive science program which picks up the former milestones, while focusing on science return as recommended by the Augustine Commission (Reference 17); and (3) a large-scale program which reaches eventually for industrialization of the Moon and the settlement of Mars. These three classes of programs provided the calibrating context within which the current study developed in-space transportation element concepts and evolution scenarios. All transportation systems are driven by physical constraints of their destinations, as well as by issues of scale -- the amount of payload and the frequency of transport. The difference between our minimal and large reference program scales represents roughly an order of magnitude in exploration accomplishment. We have developed vehicle (element) concepts capable of covering this entire range, and have analyzed evolution scenarios both spanning and exceeding it.

2.1.2 Technical Architecture Primary Requirements

During the course of the STCAEM study, and particularly during the 90-Day Study (Reference 5), many SEI (then HEI) transportation requirements were generated by NASA's Office of Exploration Level II. These are reported as appropriate and necessary in various sections of this report, as well as in References 7-11. Here, space only permits a summary discussion of the top-level requirements adopted by the study as they evolved. The concepts developed and analyzed ultimately were to accommodate the in-space transportation functions required to support the buildup of a permanent presence on the Moon and initial human exploration of Mars. Stated simply then, our chief requirement was to deliver cargo reliably to the surfaces of the Moon and Mars, and to get people to

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those places and back safely. The focus provided by SEI definition in the summer of 1989 precluded the need for the current study to examine vehicles in support of missions to other destinations. Planet surface system characteristics and Earth-to-orbit (ETO) launch vehicle characteristics were adopted as needed for manifesting purposes, largely intact from other sources. No design work was performed for these two categories. In addition, the mission planning horizon was limited to the year 2025, about 35 years from now.

The chief NASA - Level II requirement governing the dimensions of the vehicle concepts we developed came to us during the 90-Day Study, and was a crew size of 4 for Mars missions. As discussed in Section 1, a later, simple skill mix analysis for these long-duration missions showed that doubling up on critical skills (for redundancy), given reasonable expectations of how many skills each crew member could become expert in, requires in fact a minimum of 6 - 7 crew members for Mars missions. For the sake of consistency, our vehicle concepts are shown comparable to the 90-Day Study results, sized just for four crew. Effects of larger crew sizes are discussed in Section 2.3.

2.1.3 Concept Development Methodology

A vehicle concept emerges gradually through the iterative combination of requirements analysis, subsystems analysis, mass synthesis, performance analysis and configuration design. Because of the cascading, cause-and-effect nature of specific technical decisions in this cyclic process, the ability for a particular concept to remain fully parametric is incrementally lost, sacrificed for depth of detailing. The need to penetrate deeply even at the conceptual stage is twofold: (1) to uncover subtle integration interactions whose ramifications fundamentally revise the concept as they reflect the information hierarchy; and (2) to enable the production of graphical images of the concepts capable of being communicated widely but grounded firmly in engineering detail. If circumstances allow the concept development process to engage many cycles of reflexive adjustment, from requirements all the way down through subsystem detailing, the design oscillations subside eventually and the product that emerges is a robust and defensible concept. Basic differences in problems posed and solutions engineered lead concept developments in different directions. "Like" problems and solutions gravitate together; their recombination and resolution results in distinct, identifiable vehicle concepts which constitute vehicle archetypes. A concept is archetypal if it spawns concept progeny whose ancestry is clear, and if in so doing its salient features recognizably survive subsequent refinement, development and scaling. The ultimate purpose of the current Concepts and Evolution

tasks was to generate, analyze, evaluate and describe such vehicle archetypes, and the role they could play in human space exploration missions.

The architecture analysis identified seven major classes of transportation architecture for SEI lunar and Mars missions. Some are derived from different propulsion technology candidates; some are derived from distinct mission philosophies independent of propulsion method; most have many sub-options. Vehicle archetypes are keyed more closely to propulsion method than to mission mode, however, so we found that all seven SEI transportation architectures can be accomplished by derivative combinations of just five archetypal Mars transfer vehicle (MTV) concepts, two archetypal Mars excursion vehicle (MEV) concepts, and one archetypal lunar transportation family (LTF) concept. The concept evolution of these archetypes is outlined in Section 2.2.

2.1.4 Design and Neckdown Criteria

The concept development was punctuated by four "neckdowns", which winnowed down the option candidates generated at each successive level of detail throughout the study. The four neckdowns were intended to result in: (1) feasible options, based on promising propulsion technologies capable of performing SEI-class missions - the advanced propulsion candidates which survived are shown in Figure 2-1; (2) preferred options, representing the handful of candidates whose performance and technological readiness were judged to warrant detailed study - a scaled overview of STCAEM vehicle concepts is shown in Figure 2-2; (3) integrated concepts, vehicle archetypes developed sufficiently to uncover their major integration concerns and architectural context - as an example, an iterated design for a Mars nuclear thermal rocket is shown in Figure 2-3; and (4) detailed concepts, based on the reconciled integration of traded subsystems - a detail of the power conversion plumbing designed for nuclear electric propulsion is shown in Figure 2-4. The 90-Day Study occurred such that the first two neckdowns were effectively reversed; cryogenically propelled, aerobraking technology was necessarily preferred at that time, due to depth of understanding. The study rounded out the picture by completing all four neckdown activities.

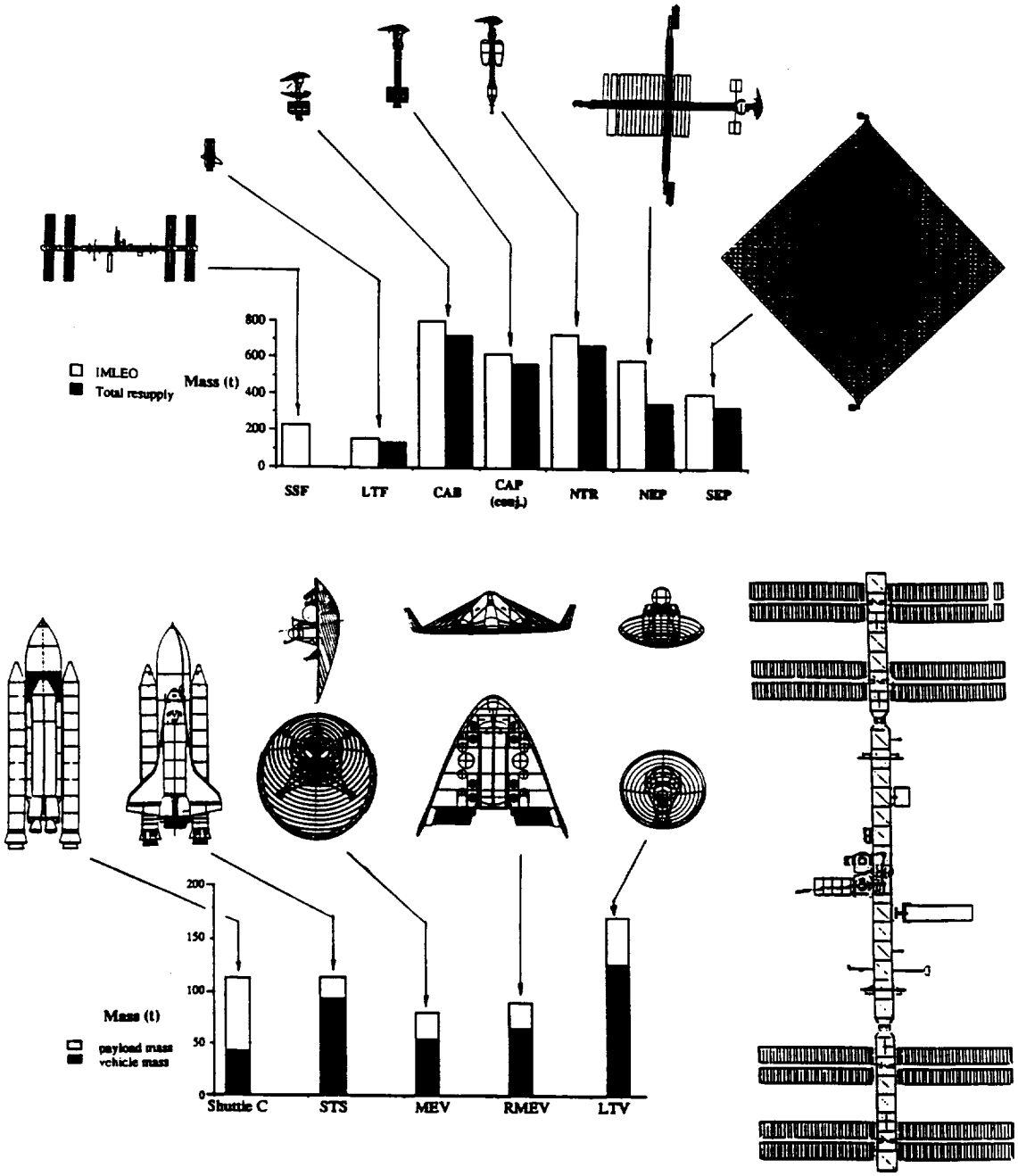
	REJECT	REJECT WITHOUT PREJUDICE	CONTINUING WORK
Solar Electric Propulsion (Ion)			X
Nuclear Electric Propulsion(Ion)			X
Arcjet	X		
Resistojet	X		
MPD		X	
Solar Lightsail	X		
Laser Lightsail	X		
α-Particle	X		
Mirror Matter			
Mirror Matter Hydrogen		X	
Solid Core NTR		X	X
Gas NTR (GCR)			X*
Magnetic Fusion		X	
Inertial Fusion		X	
Catalyzed Fusion		X	
Fusion Hydrogen		X	
Solar/Laser Thermal	X		
Laser Detonation Wave	X		
Pulsed Fission	X		
Scramjet Gun	X		
Railgun Rocket	X		
Mass Driver Rocket	X		
Flinger	X		
Orbital Tethers			X

* Continuing study work due to high level of interest in SEI community

ADVANCED PROPULSION OPTIONS DEEMED PRACTICAL FOR MARS MISSION STUDY

FIGURE 2-1

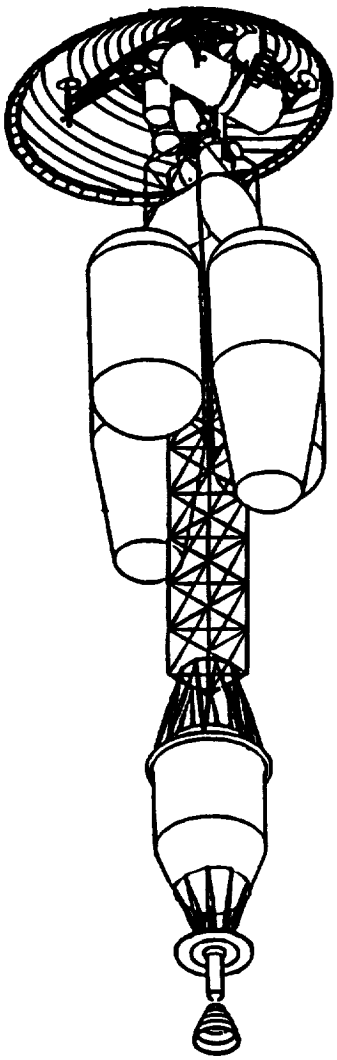
The program architecture analyses of various technology options for SEI missions led to the conclusion that the most general discriminators, cost and risk, are driven by technical discriminators different than, for instance, initial mass in low Earth orbit (IMLEO). These can be grouped into three broad categories: feasibility, flexibility, and multi-use design. As indicated above, feasibility was the first filter for all concepts considered. Flexibility has three components: (1) robustness, which is the ability to perform nominally despite variable or unanticipated conditions; (2) resiliency, which is the ability to recover from accidental delays or mishaps; and (3) evolution, which is an adaptation over time to changing requirements. Flexibility is thus a measure of a program's technical strength and



PRIMARY VEHICLE CONCEPTS COMPARED TO SPACE STATION, (TOP) LUNAR/MARS TRANSFER VEHICLES, (BOTTOM) AEROBRAKED VEHICLES

FIGURE 2-2

safety in the face of variable extrinsic factors. Multi-use design has two components: (1) re-usability, which means using the same hardware item more than once; and (2) commonality, which means using the same hardware design in more than one setting. Multi-use design is thus a measure of a program's cost-effectiveness and intrinsic longevity. Because of the leverage these two key architecture drivers have on decreasing life cycle cost and improving the return on investment, they were paramount in interpreting the results of the technical trade studies, and figured prominently in the development of element concepts.



Element	*mass (kg)*
MEV desc aerobrake	7000
MEV ascent stage	22464
MEV descent stage	18659
MEV surface cargo	25000
MEV total	73118
MTV crew hab module 'dry'	28531
MTV consumables & resupply	5408
MTV science	1000
MTV crew habitat system tot	34939
MTV frame, struts & RCS inert wt	5593
Reactor/engine weight	9684
Radiation shadow shield weight	4500
EOC propellant (dV=2654 m/s)	27756
TEI propellant (dV=3900 m/s)	59245
EOC/TEI common tank wt (1)	13845
MOC propellant (dV=3870 m/s)	151680
MOC tanks (2)	25572
TMI propellant (dV=4201 m/s)	286146
TMI tanks (2)	43092
ECCV	0
Cargo to Mars orbit only	0
IMLEO	*735190*

2016 opposition with Venus swingby

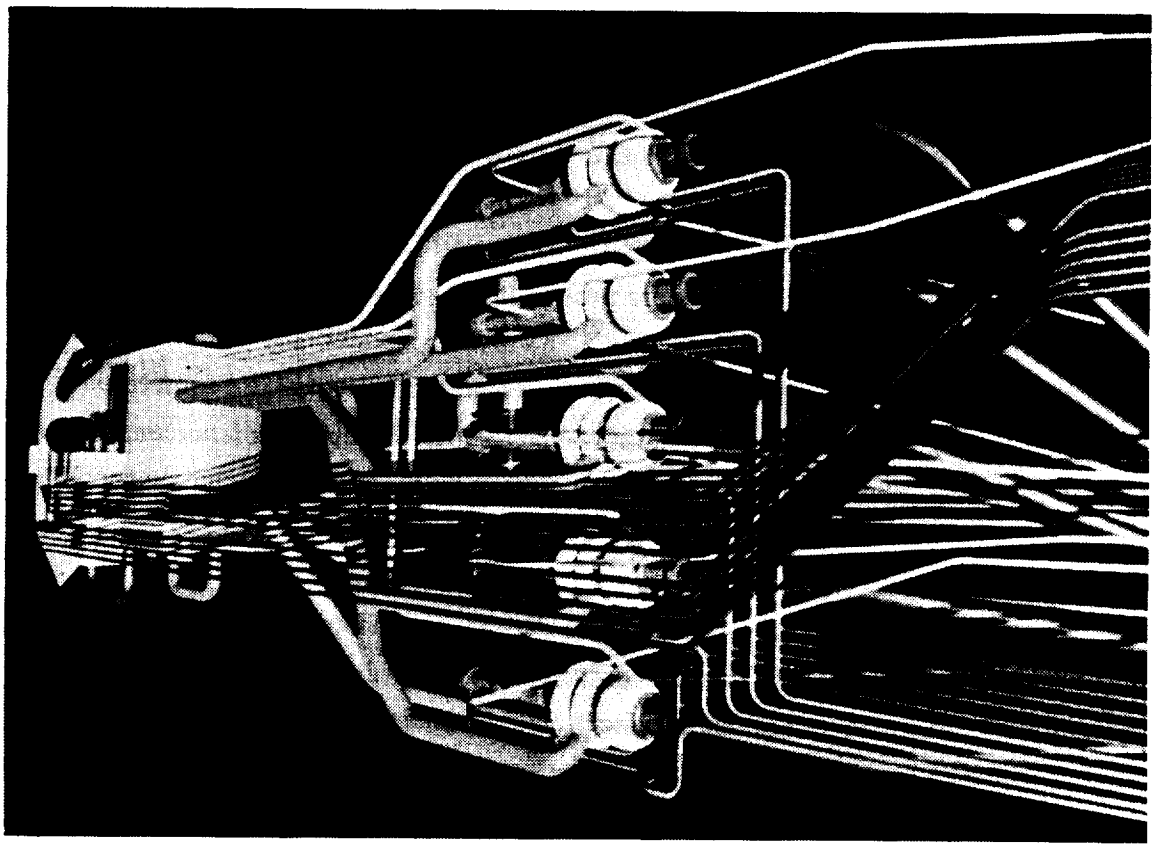
434 day mission time

INTEGRATED CONCEPT FOR A NUCLEAR THERMAL PROPULSION VEHICLE
FIGURE 2-3

2.2 Results

2.2.1 Lunar Transportation

During the 90-Day Study, NASA's two Office of Space Flight (Code M) Space Transfer Vehicle (STV) contractors supported development of SEI lunar transportation concepts. That work treated lunar SEI missions (and evolution to the support of later Mars missions) as the far end of a more near-term STV program, most of whose missions were satellite delivery and servicing requirements derived from Civil Needs Data Base (CNDB) projections. This study's contribution to that work was primarily crew system design, since this was recognized as offering potential for commonality with crew cab design for Mars excursion vehicles (MEVs).



DETAILED SUBSYSTEM CONCEPT FOR A DYNAMIC NUCLEAR POWER CONVERSION SYSTEM

FIGURE 2-4

Later, we began to address the complete design of a lunar transportation system. Because of our Mars concept experience, our perspective was particularly sensitive to evolutionary

systems; the approach of looking back from a Mars mission perspective is thus complementary to that of the parallel NASA studies. Our effort was guided by attention to two broad drivers. The first of these drivers is the technical requirements whose resolution had proved so intractable for earlier concepts:

a. State-of-the-art understanding of constraints imposed by the detailed geometry of aerobraking upon Earth return: non-symmetrical relative wind configurations for lifting flight profiles; off-axis placement of composite mass-center (CM); and changing mass-balance conditions due to sequential propellant expulsion.

b. The need to accommodate "mixed" payloads in a reasonable lunar exploration program: versatility in the delivery of a wide variety of heavy-cargo payload manifests, which rarely if ever can be divided into two or four equally heavy pieces; cargo processing and loading requirements in LEO; cargo exchange between transfer vehicles and excursion vehicles; cargo offloading on the surface of the Moon; cargo placement on manned flights; crew access to the surface, and shirtsleeve (IVA) exchange of crew between transfer and excursion vehicles.

c. Provision for transfer of cryogenic propellants: a typical scenario is supplying LH_2, brought from Earth by a transfer vehicle, to a reusable lander based in low lunar orbit (LLO). Cryogenic propellants are baselined, of course, because of the requirement for high-thrust propulsion for planetary landing and ascent. (The use of nuclear thermal propulsion for lunar transfer is potentially attractive, but still involves cryogenic propellant management.)

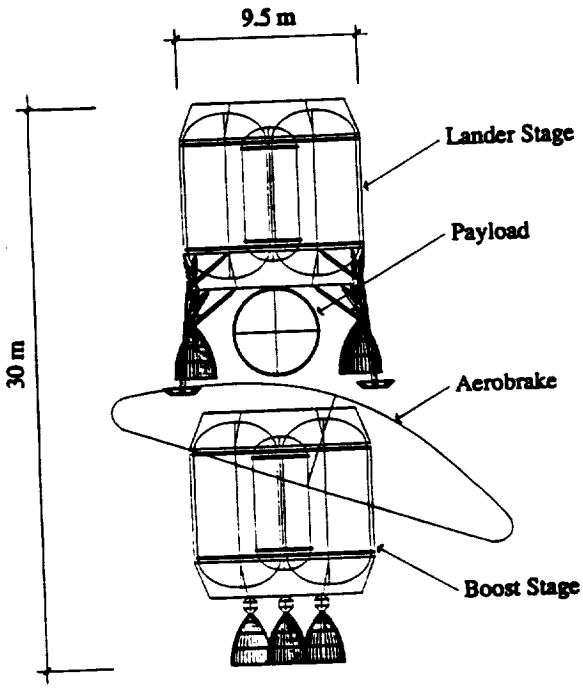
d. Potential for full system reusability: designs which drop tanks are better for limiting aerobrake size, but have negative cost implications for advanced cryogenic storage technology, and negative operational implications via the accumulation of empty tanks in cis-lunar space and on the lunar surface. Mission modes which posit multiple annual flights for several decades drive us to consider full reusability.

The second broad driver is vehicle adaptability. Over several decades of lunar operations, many mission modes should be accommodated. What is needed is not so much a single vehicle or pair of vehicles, but rather an evolving lineage of vehicles, fabricated on long-lived production lines, which can be adapted gracefully and economically to handle contemporary requirements. Two observations keyed this investigation:

a. Lunar flight hardware decisions will probably be made before final site selection decisions. This means that the lunar transportation architecture should not constrain site selection to less than potentially global access. Many possible mission modes must be preserved by the architecture.

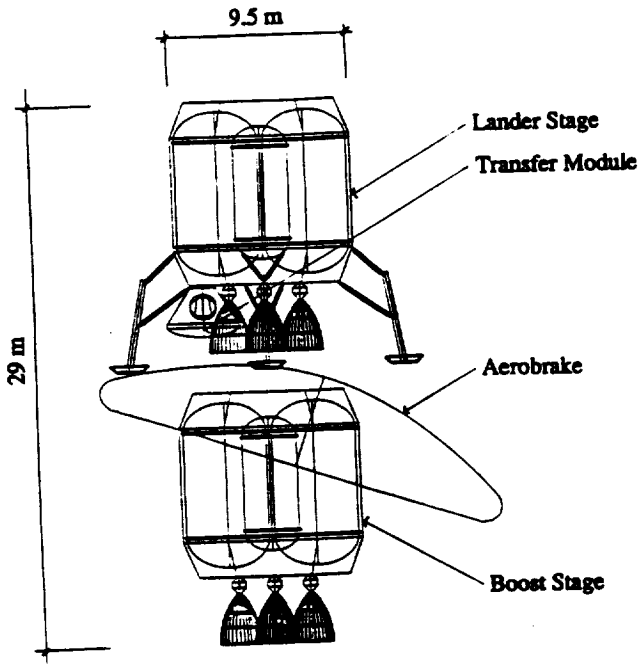
b. An early lunar surface operations capability can be obtained by using a tandem-direct flight mode, in which one lunar transfer vehicle (LTV) boosts another, "campsite" LTV, to a fractional-orbit, direct landing on the Moon. The crew would be sent separately on an identical profile, returning directly to Earth's surface in their heat-shielded crew capsule. No LLO operations, aerobrake, LTV recovery, or space station rendezvous upon return would be needed, nor would a specialized lunar lander (LEV).

What resulted was a lunar transportation family (LTF) concept (Figure 2-5 shows some key members of this family) consisting of various "models" of two basic, cryogenic vehicle "chassis": an LTV with 110 t propellant capacity, and an LEV with 25 t capacity. Particular vehicle combinations from this evolutionary family can handle many distinct mission modes (Figure 2-6 lists just 11 of them), to provide versatile, flexible service for decades as mission requirements evolve. Our preferred evolutionary scenario for the LTF is shown in Figure 2-7. For instance, the addition of an aerobrake would permit unmanned recovery of the boost-stage LTV, providing invaluable flight qualification experience for later man-rating. (Such an aerobrake can be essentially the symmetrical central core of the asymmetrical Mars-class aerobrakes discussed later, as the L/D requirement is only about 0.25 for lunar missions; aerobrake technology evolution is then enhanced.) A heavy-cargo lander would be a modest upgrade to the campsite vehicle design. If the scale of the exploration architecture justified the more efficient lunar-orbit-rendezvous (LOR) mode, a dedicated lunar excursion vehicle (LEV) could be introduced. For fully reusable operations, a version of the campsite habitation module would provide crew support during the extended wait required by return orbit phasing. If electric propulsion (SEP or NEP) Mars missions were operated efficiently through a lunar libration point, LTVs could support these as well as lunar operations. Also, LTVs could supply the final capture propellant to a nuclear thermal rocket vehicle returning from Mars. Lunar transportation operations can be upgraded to the use of lunar-derived oxygen (LLOX) with this family of vehicles. All combinations of crew and cargo manifests identified so far for lunar support, and all lunar-related SEI missions identified so far, can be accommodated by the lunar transportation family (LTF).



Mass Statement

Boost Stage Inerts	10,638
Boost Propellant	108,950
Aerobrake	3,000
Lander Stage Inerts	11,113
Lander Propellant	92,603
Payload	60,000
IMLEO (in kg)	286,304

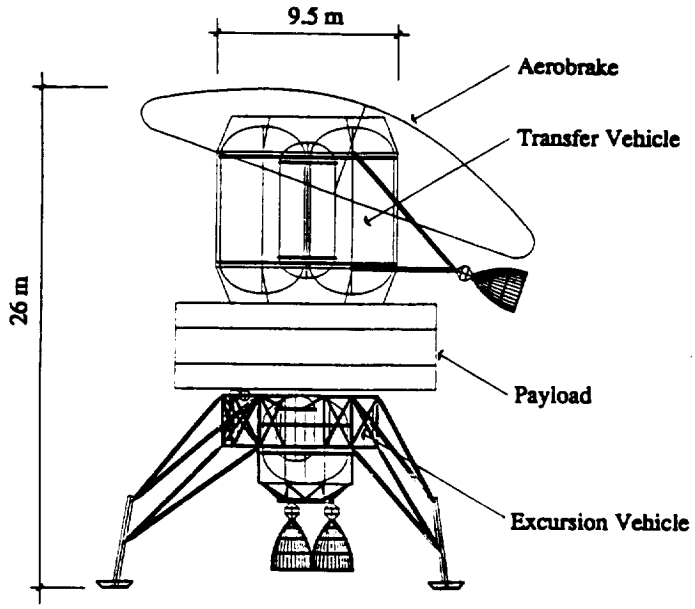


Mass Statement

Boost Stage Inerts	10,638
Boost Propellant	78,647
Aerobrake	3,000
Lander Stage Inerts	11,113
Lander Propellant	84,436
Transfer Module	13,633
Payload	5,000
IMLEO (in kg)	206,467

A FEW KEY MEMBERS OF THE VERSATILE LUNAR TRANSPORTATION FAMILY

FIGURE 2-5

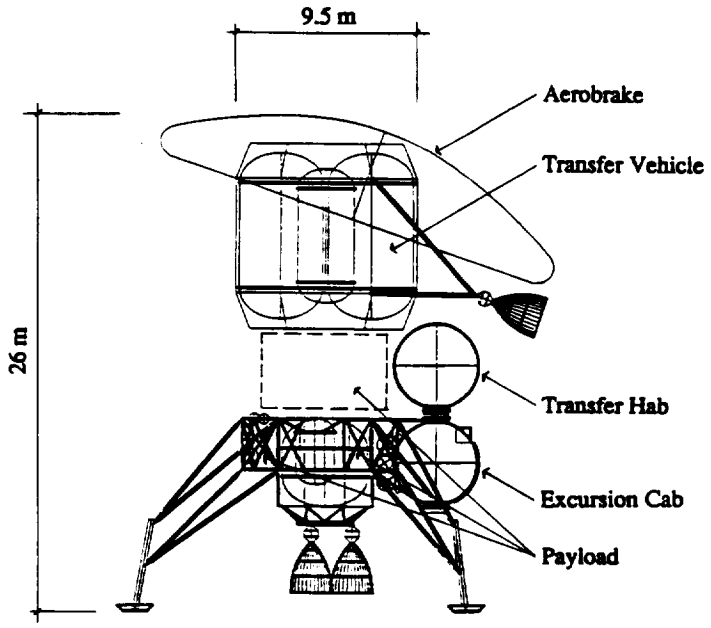


Mass Statement (LEV Delivered)

Transfer Vehicle Inerts	10,638
Transfer Propellant	107,719
Aerobrake	3,000
Excursion Vehicle Inerts	5,592
EV Propellant	23,151
Payload	24,000
IMLEO (in kg)	**174,100**

Mass Statement (LEV Refueled)

Transfer Vehicle Inerts	10,638
Transfer Propellant	107,100
Aerobrake	3,000
Resupply Tanks	2,377
Resupply Propellant	23,765
Payload	25,000
IMLEO (in kg)	**171,880**



Mass Statement (LEV Delivered)

Transfer Vehicle Inerts	10,638
Transfer Propellant	106,602
Aerobrake	5,000
Transfer Hab	7,500
Excursion Vehicle Inerts	5,592
EV Propellant	19,768
Excursion Cab	4,000
Payload	8,000
IMLEO (in kg)	**167,100**

Mass Statement (LEV Refueled)

Transfer Vehicle Inerts	10,638
Transfer Propellant	102,184
Aerobrake	5,000
Transfer Hab	7,500
Resupply Tanks	2,100
Resupply Propellant	20,997
Payload	10,000
IMLEO (in kg)	**158,419**

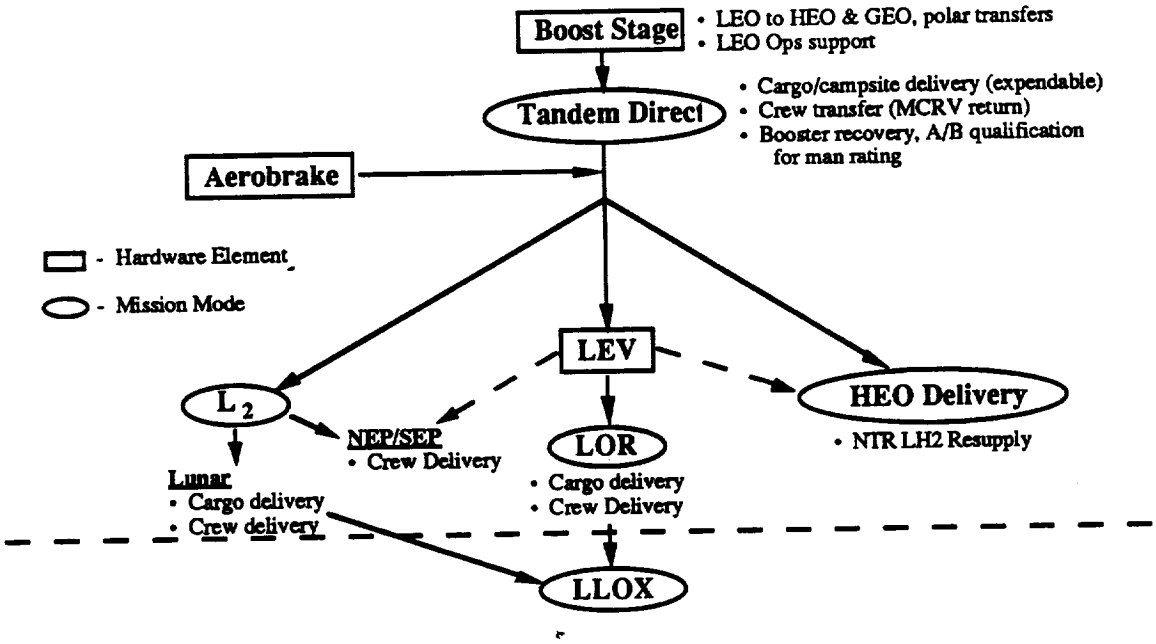
A FEW KEY MEMBERS OF THE VERSATILE LUNAR TRANSPORTATION FAMILY (CONTINUED)

FIGURE 2-5

Mission Mode/Date \ System	110 t Propulsion Stage	25 t Propulsion Stage	Transfer Hab	Crew Cab	Aerobrake	Propellant From Earth (kg)
Tandem Direct Cargo / 2000	2	—	—	—	(1)	201,553
Tandem Direct Crew / 2001	2	—	1	—	(1)	163,093
Tandem Direct Large Cargo / 2002	2	—	—	—	(1)	201,553
LOR Cargo / 2004	1	1	—	—	1	130,865
LOR Crew and Cargo / 2004	1	1	1	1	1	123,181
LOR Cargo using LLOX / 2009	1	1	—	—	1	65,109
LOR Crew using LLOX / 2009	1	1	1	1	1	73,274
Cargo Delivery Through L_2 / 2010	2	—	—	—	1	159,153
Crew Delivery Through L_2 / 2010	2	—	1	1	1	159,145
NTR Resupply / 2015	1	—	—	—	1	42,021
NEP/SEP Crew Delivery / 2015	1	—	1	—	1	33,556

ELEVEN OF THE PRIMARY LTF MISSION MODES OF INTEREST

FIGURE 2-6



INCREMENTAL EVOLUTIONARY APPROACH RECOMMENDED FOR LUNAR TRANSPORTATION CAPABILITY

FIGURE 2-7

2.2.2 Habitation Systems

Definition of crew habitable systems proceeded during the entire study because of their high mass leverage on integrated vehicles, and concentrated on building up mass manifests from the subsystem level. Many of these data were adapted directly from contemporary data generated in the Space Station *Freedom* program, with adjustments made for differences in mission requirements, operating conditions and projections of second-generation improvements (particularly with regard to subsystem masses). The earliest Mars crew size selected was five; the 90-Day Study later pared this down to four, used as a reference minimum for most of the study.

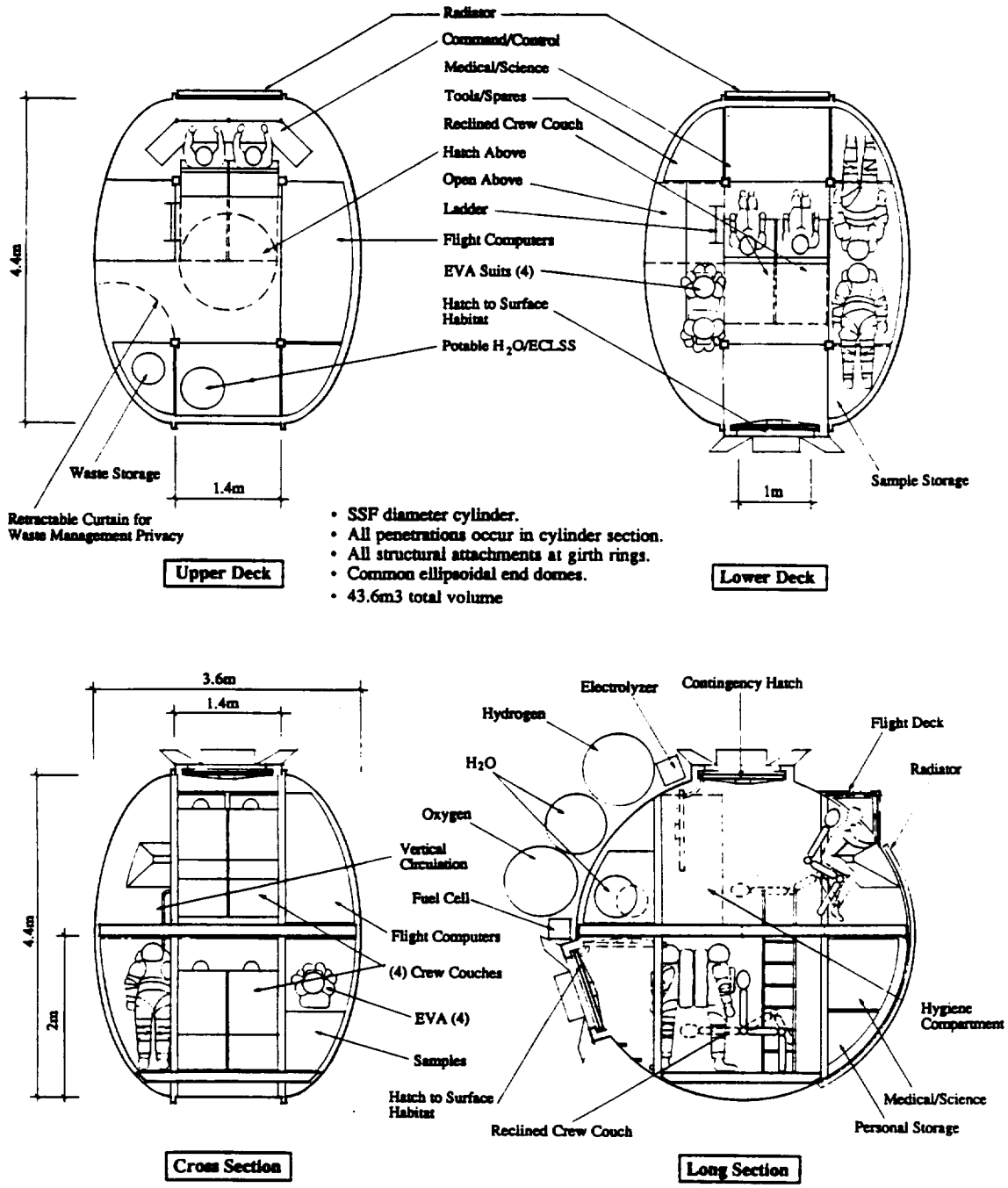
An aspect of our approach was to discover how the small, crew "cab" concepts required for lunar transfer, lunar landing, and Mars landing could be driven toward some common design, in the service of the multi-use program philosophy. We developed a baseline structure concept for them derived from Space Station *Freedom* habitation module technology, but with evolutionary structural variations to reduce mass (unpenetrated, ellipsoidal end domes; removable LEO debris shielding that "stays home" when the mission begins; and a thin skin pressure-stiffened for ETO launch). Varying the 4.4 m-diameter barrel-section length can accommodate a wide range of crew sizes and flight durations. Specifying open-loop life support systems appropriate for short-duration cab use and shortening the barrel section to the minimum required for hatch penetrations leads to an EV (descent, ascent and short surface stays) cab suitable for up to 8 crew, shown in Figure 2-8. Sizing the module for 3 - 4 week durations, adding a radiation storm shelter and increasing ECLSS closure leads directly and simply to an LTV crew hab.

The problem of long-duration space habitation incurred by Mars transfer (an order of magnitude longer duration than SSF duty tours) requires a fundamentally different solution than simple evolution of SSF modules. Since the pressure vessel itself represents a relatively modest fraction of the mass, and a small fraction of the cost, of habitation systems, this does not mean an abrogation of commonality.

The high mass leverage of Mars mission habitats argues strongly against clusters of several small modules; and the availability of heavy-lift launch permits integration of large, unitary modules on the ground. We performed a major trade study aimed at determining a habitation system concept appropriate for integration into Mars mission vehicle archetypes.

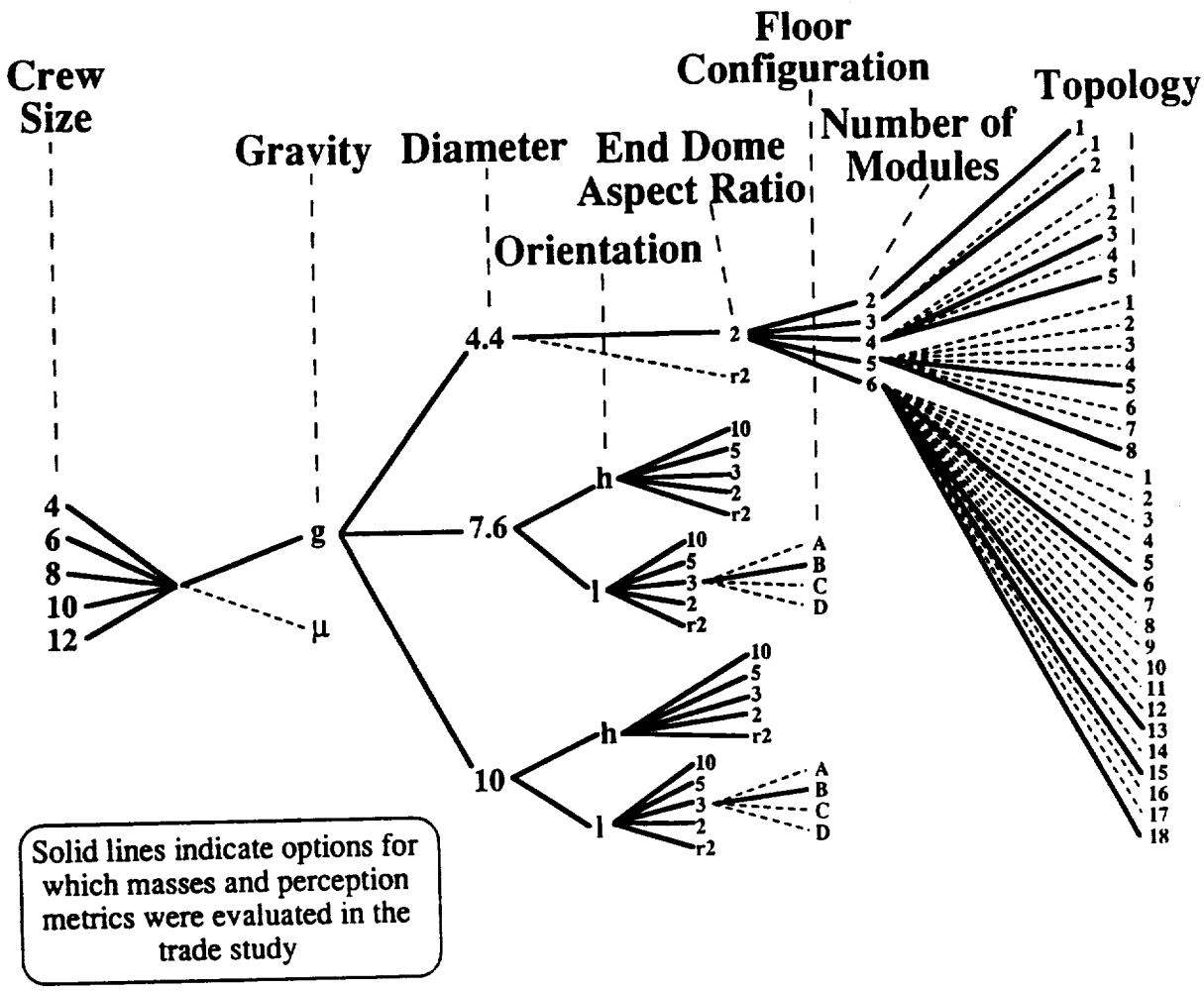
We examined crew sizes ranging from 4 to 12; module diameters ranging from 4.4 to 10 m; module end-dome aspect ratios ranging from 2 to 10; microgravity and artificial gravity conditions; vertical and horizontal module orientations; internal floor and pressure-bulkhead arrangements; 34 module clustering topologies, for clusters of from 2 to 6 modules, using 7 topology metrics; internal spatial characteristics using 10 geometry metrics; crew perception by soliciting 56 individual responses to 6 reference layouts; and 5 construction technology options based on 15 candidate materials systems. In all, 1480 distinct options were considered, and necked down to 150 prime candidates, as diagrammed in Figure 2-9.

These were compared in four categories: (1) functionality, or how well they would accommodate crew activities and safety; (2) integration, or how well they would work with other vehicle systems and the rest of the SEI architecture; (3) cost, measured by mass, manufacturability, ease of outfitting, and commonality potential with surface base applications; and (4) perception, or how comfortable they would appear for human occupants. Thirty "finalist" module system candidates were fully mass-estimated, including internal outfitting subsystems.



COMMON SHORT-DURATION CREW MODULE

FIGURE 2-8



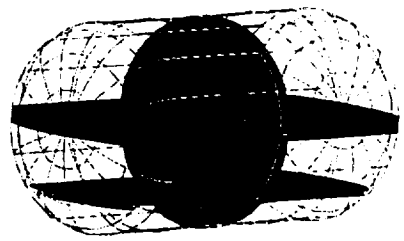
SOME KEY TRADE OPTIONS EXAMINED FOR HABITATION MODULES
FIGURE 2-9

The preferred concept shown in Figure 2-10, is 7.6 m in diameter; has 2:1 aspect-ratio end-domes; is oriented "horizontally" with a cross-section, bisecting pressure bulkhead; has two longitudinal floors, and can accommodate up to 12 people in stretched versions before becoming unwieldy in length. Given a reasonably sized HLLV, such a habitation system could be launched intact, already fully integrated, to LEO ; versions sized for up to 10 crew could be landed on the Moon, ready to use, by our LTF.

Major Features

- 7.6 m diameter
- 2:1 aspect ratio, unpenetrated end domes
- Cross-section, bisecting bulkhead
- Diametral tension - tie, deep floor
- Extensive commonality across architecture: g-field optimized

6 Crew Configuration



PREFERRED REFERENCE LONG-DURATION CREW MODULE CONCEPT
FIGURE 2-10

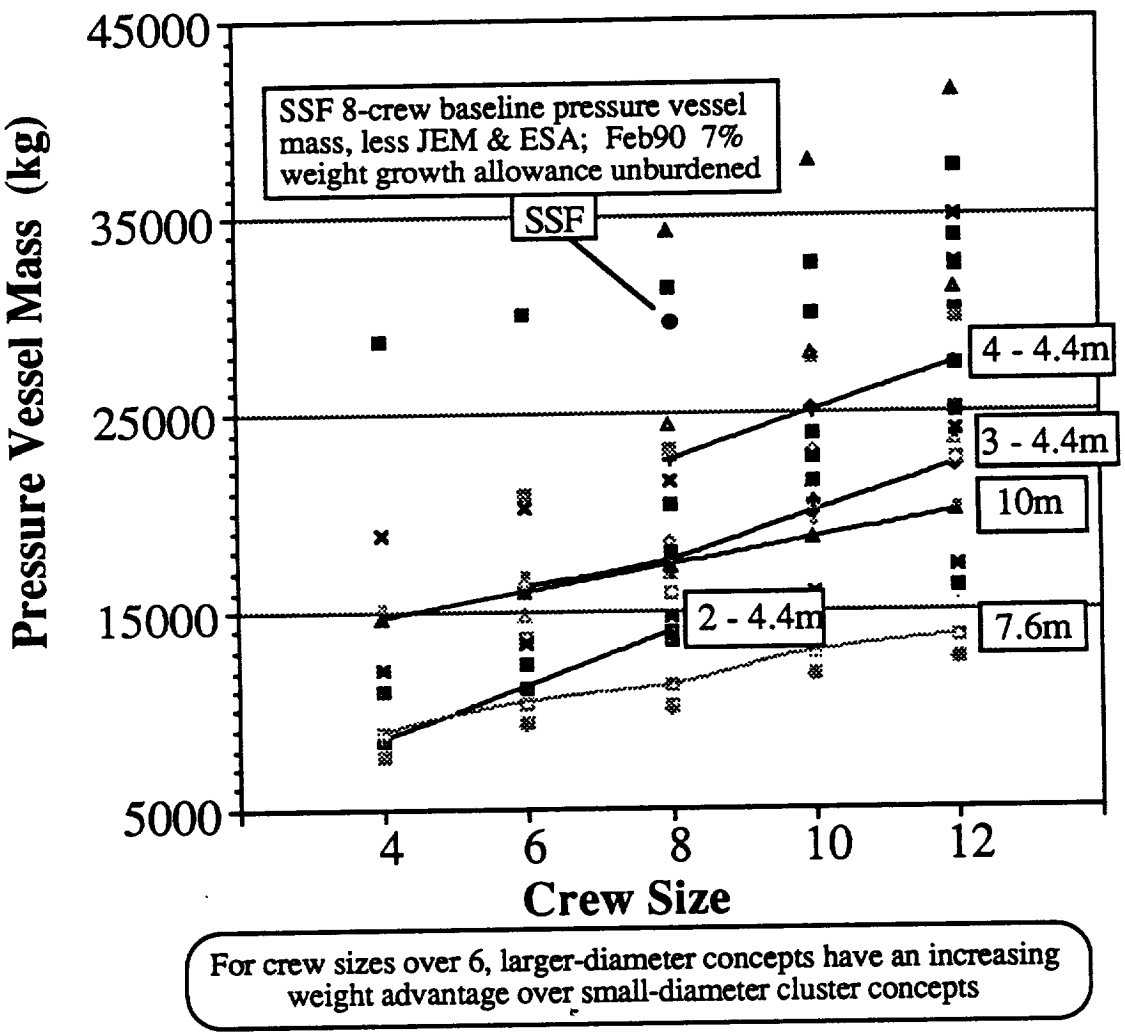
In terms of functionality, the unitary vessel minimizes leakage and parts count and permits a wide variety of internal outfitting designs. Its diametral main floor maximizes nominal floor area and facilitates a mass-reducing tension-tie structure system analogous to pressurized jetliner cabins. The compact domain is good for access-time contingencies, and the horizontal orientation has the best overall multi-floor efficiency in gravity fields for a range of crew sizes. The pressure vessel concept offers less intrinsic wall area for mounting equipment than do smaller-diameter options, but this can be compensated by internal partitioning design.

Considering integration, the concept minimizes orbital assembly operations required, simplifies ground-installation of internal subsystems, is compatible with a launch shroud size likely available even early in SEI, facilitates vehicle packaging behind or inside aerobrakes, and can accommodate very large crews through simple clustering.

In terms of crew perception, our survey results showed that for iso - volume options, technical professionals tend to perceive larger diameter concepts as more spacious. The horizontal module's overhead barrel vault remains proportionately invariant with crew -- and hence module -- size and is thus superior to the overhead dome available in vertically-oriented modules. The preferred concept has a better plan aspect ratio than smaller-diameter options; however, it does suffer from a low "circulation option boredom" score and a low intrinsic number of distinct spatial units. While these are potentially important for long-duration missions, both can be compensated by internal outfitting.

Cost-wise, the selected concept suffers from fewer size-related fabrication complications than do the larger-diameter options. Well-understood welded-metal technology could be

used for its fabrication, but the concept represents a prime opportunity for materials and process improvements to be brought on line. Specifically, a filament-wound technique using continuous SiC fibers in a plasma-sprayed Al matrix appears promising. The selected concept was among the lightest-mass options investigated, as shown in Figure 2-11; in particular it is far better than cluster options. Finally, the most far-ranging cost-reduction leverage of all may be the concept's straightforward adaptation to planet surface base applications.

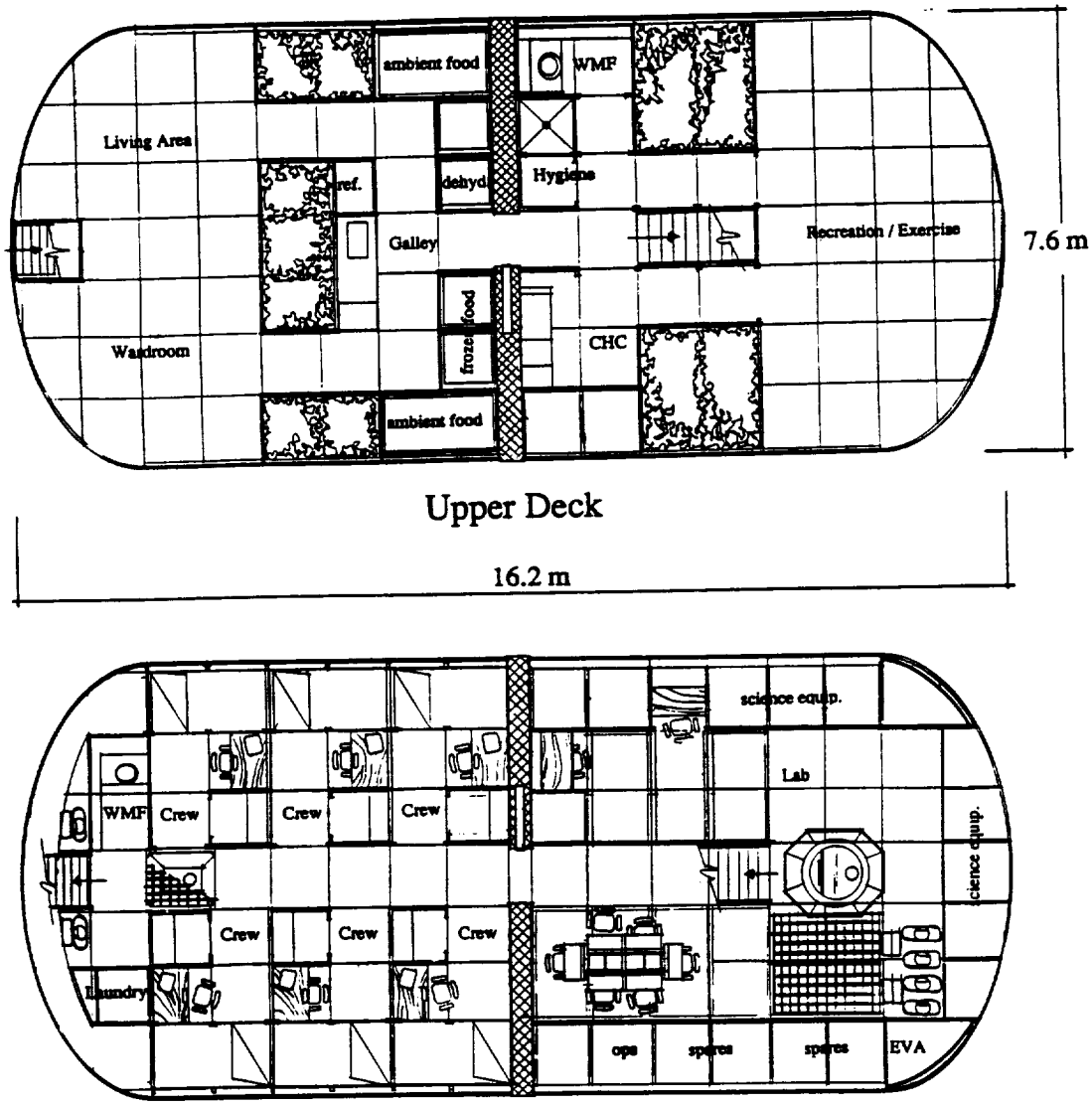


PRESSURE VESSEL MASS COMPARISON OF CANDIDATE CONCEPTS
FIGURE 2-11

The preferred long-duration habitat module concept was incorporated into all MTV archetypes developed. Our subsystems definition of the interior of the reference module presumes the crew size of six our Mars mission skill mix analysis recommended. The initial structure concept presumed in the habitat trade study (which kept internal equipment inertial load paths distinct from the pressure-containing module skin) was retained for further study. Skin thickness required for minimum-gauge considerations keeps this from being the minimum-mass approach, but the clear structure concept did permit a quick interior design model to be developed. All loads are brought to the bisecting bulkhead via the two floors (which incidentally stiffen both sides of the bulkhead against the contingency pressure load), which are tied together by a shear-panel arrangement of the partition walls dividing up the lower deck. Thus the entire module interior is a deep beam/frame cantilevered from the bulkhead.

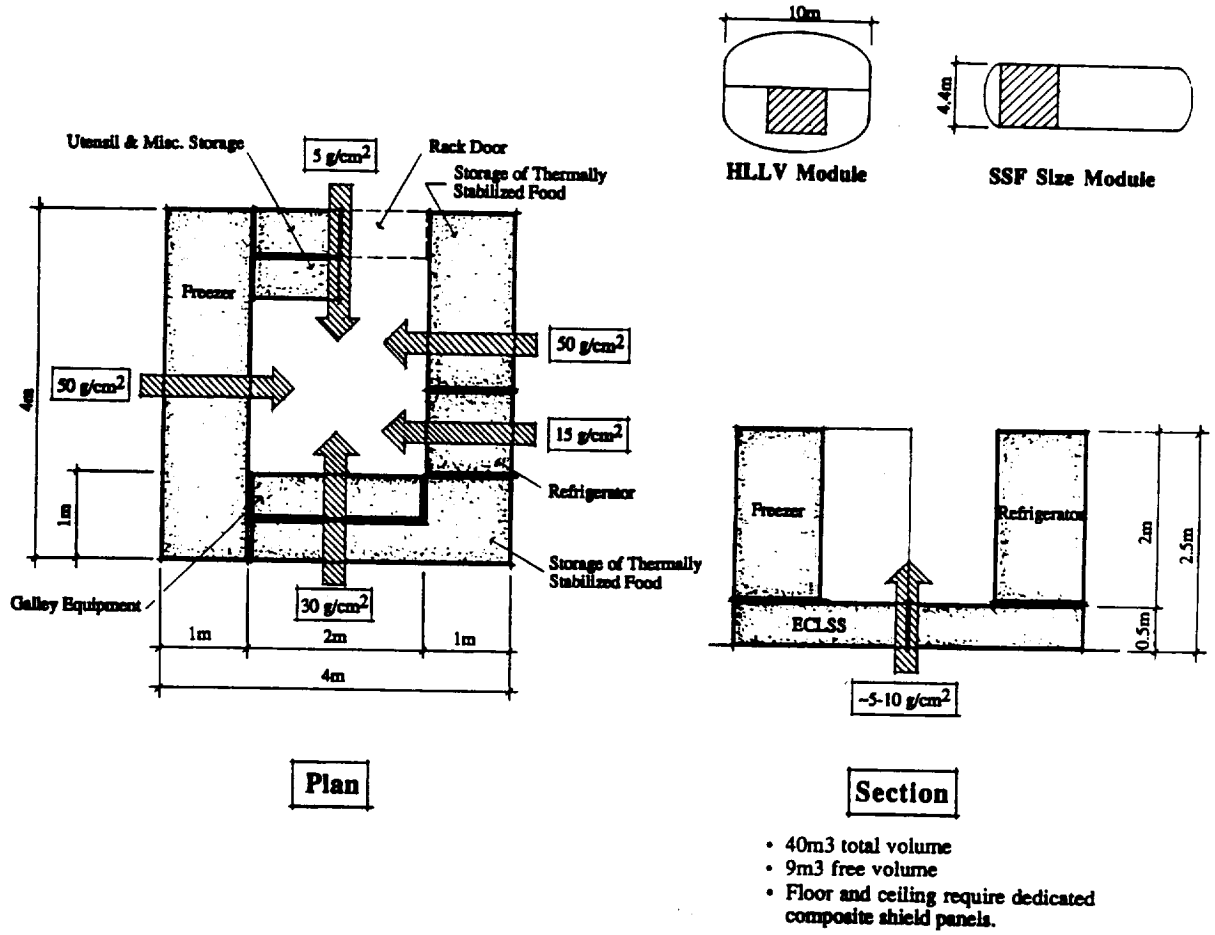
Expanding on work done for the habitat trade study, a crew activity functional analysis was performed, followed by proximity analysis to establish interconnection and interference requirements among the allocated functions. These data were worked into a successfully zoned layout (Figure 2-12): crew cabins and vehicle operations "downstairs", on opposite sides of the bisecting bulkhead; and galley, living, major hygiene, athletic and recreational areas "upstairs" in the higher-ceilinged spaces. The module primary and secondary structure, internal architecture, furniture, equipment and supplies were computer-modeled using a limited palette of representative materials, to prepare for detailed radiation analysis. Detailed results are expected to indicate promising bulk-mass radiation protection geometries practical for long-duration missions. For this initial assessment, no dedicated "storm shelter" was designed in; however, sufficient consumables, equipment and structure are configured around the galley area to constitute the beginnings of an "extended storm shelter." Detailed radiation transport analysis will determine what more needs to be done for adequate crew protection from short-term solar proton events (SPE) as well as long-term galactic cosmic ray (GCR) flux.

Earlier in the study, we had developed a concept for a dedicated storm shelter combined with the galley, which pioneered the synthesis of two requirements: (1) crew-carrying missions outside the protective geomagnetic field require bulk shielding to attenuate high-energy protons from SPEs; and (2) materials rich in hydrogen and other light elements are the best proton attenuators, and long-duration missions require many tonnes of



INTERIOR LAYOUT SUITABLE FOR RADIATION DOSE MODELLING
FIGURE 2-12

consumables for the crew (over 2 t/man for conjunction-class transfers). We were led naturally to the development of a galley concept with well-packaged food and equipment stowage, which could double as a storm shelter (Figure 2-13), and augmented only where strictly necessary by dedicated radiation shielding. In fact, this coincidence of the need for shielding mass and the need for consumables tends to argue against the conventional wisdom that advanced ECLSS closure including food growth is required for Mars-class missions. The mass saved by food closure might be required anyway for shielding. Detailed radiation transport modeling is expected to help resolve this issue.



SHIELDING MASS AVAILABLE FROM A MARS MISSION-CLASS GALLEY
FIGURE 2-13

2.2.3 System Commonality and Evolution

Because commonality has high leverage for program cost-effectiveness, STCAEM developed an evolutionary strategy for facilitating, and benefitting from, commonality. The greatest commonality challenge is between early systems designed for lunar missions and later systems designed for Mars missions. Because of similar flight regimes, crew systems requirements and durations, excursion vehicle crew cabins for lunar and Mars uses proved most amenable to strict commonality; for conceptual purposes the LEV and MEV crew cabs can be considered just slightly different "models" of the same element.

An effort was made to extend the commonality approach to entire excursion vehicle designs for the Moon and Mars. This extreme degree of commonality was conceptually feasible only for a particular size class of vehicle (a 25 t propellant-capacity vehicle typical of LEV designs is comparable to a so-called "mini-MEV," which would take 3 crew and 1 t payload to the surface of Mars). Except for this special case, the differing gravity levels of the Moon and Mars, and configuration complications arising from aerobraking upon descent at Mars, tend to drive LEV and MEV designs apart. The LEV and MEV are likely to come on line years apart in any case, and we found a more productive way to introduce commonality.

We found it useful to consider commonality at three program levels: (1) the mission design level, using the same mission design to accomplish different programmatic architectures (a problematic category because mission designs are by definition tied to unique requirements); (2) the functional element level, using end-items from the same production line to fill different roles within a given mission design (the most appropriate example we developed is the evolutionary LTF described earlier); and (3) the performing subsystem level, using system assemblies or components from the same production line in different functional elements (a sensible way to standardize industry, get predictable performance and facilitate product longevity). This latter approach, applied to engines, sensors, processors, some structural components and modular life support hardware, shows great promise for cost-effectiveness and preserving program resiliency. At the component level, extensive commonality can be worked into the fabric of SEI, as shown in Figure 2-14.

Another application of this subsystem/component commonality-for-evolution approach is the potential use of hardware systems developed mostly for lunar transportation (and augmented by long-duration crew systems) for early Mars missions staged out of high-orbit node locations like Earth-Moon L_2. Earth-departure (TMI) delta V for this mission mode is such that the need for a large TMI stage is obviated altogether, as is the need for a large cryogenic space engine. The use of chemically-propelled technology for Mars missions thus permits early accomplishment if the lunar systems are evolved carefully.

2.2.4 Mars Transfer Propulsion Options

The 90-Day Study precipitated a reversal of the first (feasible concept) and second (preferred concept) neckdowns described earlier, since NASA was required at that time to focus immediately on candidates already known to be feasible. In order to proceed with

	SSF	SEI LEO Ops	Lunar Transfer	Lunar Excursion	Lunar Surface	Mars Transfer	Mars Excursion	Mars Surface
Vehicles								
Lunar Transfer	✕	⊘	—	●	✕	⊘	⊘	✕
Lunar Excursion	✕	⊘	⊘	—	✕	○	●	✕
Mars Transfer	✕	✕	✕	✕	✕	—	✕	✕
Mars Excursion	✕	✕	✕	⊘	✕	○	—	✕
MCRV	—	●	⊘	✕	✕	⊘	✕	✕
Vehicle Systems								
Main Propulsion	✕	⊘	—	⊘	✕	TBD	⊘	✕
ACS Propulsion	✕	⊘	—	⊘	✕	⊘	⊘	✕
Aerobrake	✕	✕	⊘	✕	✕	⊘	—	✕
Propulsive Power	✕	✕	⊘	✕	⊘	—	✕	⊘
Housekeeping Power	—	⊘	⊘	⊘	⊘	⊘	⊘	⊘
Communications	○	⊘	⊘	⊘	⊘	—	⊘	⊘
Avionics	○	⊘	—	⊘	○	⊘	⊘	○
Robotics	○	⊘	⊘	⊘	⊘	—	⊘	⊘
Thermal Control	—	⊘	⊘	⊘	⊘	⊘	⊘	⊘
Structures	—	○	⊘	⊘	⊘	⊘	○	⊘
Crew Systems								
Small Habitat	—	⊘	⊘	⊘	⊘	⊘	⊘	⊘
Large Habitat	○	⊘	⊘	⊘	●	—	⊘	●
Airlock	○	⊘	⊘	—	⊘	⊘	⊘	⊘
Suit	○	⊘	⊘	—	●	⊘	⊘	○

— Exemplar ● Element ⊘ Subsystem ⊘ Component ○ Technology ✕ None

EXTENSIVE COMMONALITY OPTIONS AVAILABLE TO SEI
FIGURE 2-14

the performance analysis of advanced propulsion options after the 90-Day Study, to-scale initial configuration concepts were developed for: NTR (solid-core nuclear thermal rocket), GCR (gas-core nuclear rocket), NEP (nuclear electric propulsion, and SEP (solar electric propulsion). Conceptual designs were then engineered for all of these except GCR after the feasibility neckdown verified their promise. A significant accomplishment of the current study was to demonstrate that trip times available with electric propulsion through good mission design make it a viable and competitive candidate for crew-carrying missions as well as cargo missions. This result substantially broadened the range of options available for Mars-class SEI missions.

Combining the lessons learned from engineering those vehicle concepts with the results of our transportation architecture assessment led us directly to a vital conclusion of the STCAEM study regarding Mars transportation systems: All the prime propulsion candidates "win" on some scores but none win on all scores; how the candidates compare depends critically on what the program is trying to accomplish, as compared in

Figure 2-15. If early accomplishment, with current technology, in a small-scale program is paramount, cryogenic-all-propulsive (CAP) looks best. If fast trip times, and a wide range of payload sizes is paramount, NTR looks best. If very large payloads, many trips, and even missions beyond Mars are important, NEP looks best. And if high performance, extreme redundancy and non-nuclear technology are paramount, SEP is the clear winner. STCAEM was tasked with driving to comparable levels of detail vehicle concepts based on the prime propulsion technology candidates. It is not possible at this time to perform a "final neckdown" to a single, preferred mission system, because SEI lacks sufficient architectural specification so far. Once exploration activity priorities are established, and crew sizes, payload manifests, flight rates and the number of missions are determined, then

		NTR	NEP	SEP	CAP	CAB
Flexibility (robustness, resiliency, evolution; preservation of options)	Launch window size	✕	●	●	✕	✕
	Insensitivity to opportunity	○	●	●	✕	○
	Insensitivity to variable payload mass	○	●	●	✕	✕
	Insensitivity to parking orbit	✕	●	●	✕	✕
	Intrinsic system redundancy	✕	○	●	✕	✕
	Capability for trans-Mars missions	○	●	✕	✕	✕
Multi-use design (commonality, re-use; establishment of infrastructure)	High-power surface system commonality	✕	●	●	✕	✕
	Lunar crew transportation commonality	●	✕	✕	●	●
	Lunar cargo transp. commonality	●	●	○	○	●
	Lunar return/MEV aerobrake commonality	✕	●	●	✕	●
	MTV re-usability	○	●	●	●	○
	In-space infrastructure buildup	○	●	●	✕	✕
Integration	Trip time	●	○	✕	○	○
	Ease of MTV launch packaging	●	○	✕	●	●
	Simplicity of in-space dry-dock operations	○	✕	✕	●	●
	Safety for EVA crews	✕	✕	●	○	○
Programmatics	Capability for early Mars	○	✕	✕	●	○
	Low acquisition cost	○	✕	✕	●	○
	Technology readiness	○	✕	○	●	○
	Commercial technology potential	○	○	●	○	○

Key: ✕ comparatively poor ○ moderate ● good

PROGRAM PERFORMANCE MATRIX FOR THE FIVE MARS TRANSFER PROPULSION OPTIONS

FIGURE 2-15

the performance and design integration data can be used to match an appropriate vehicle archetype to those requirements. As discussed in Section 4, NTR presently appears to be the most versatile baseline for a reasonably-sized SEI program.

2.2.5 Mars Transportation Vehicle Concepts

Analysis of aerobraking resulted in two performance ranges of interest for Mars entry (hypersonic L/D = 0.5, and L/D = 1.0), as well as the use of high-energy aerobraking (HEAB) for capture at Mars. Two archetypal MEV concepts were developed by STCAEM, based on the "low" and "high" L/D performance ranges analyzed. In addition, five archetypal MTV concepts were based respectively on: cryogenic/aerobraking (CAB), cryogenic all-propulsive (CAP), nuclear thermal rocket (NTR), nuclear electric (NEP), and solar electric (SEP) propulsion technologies.

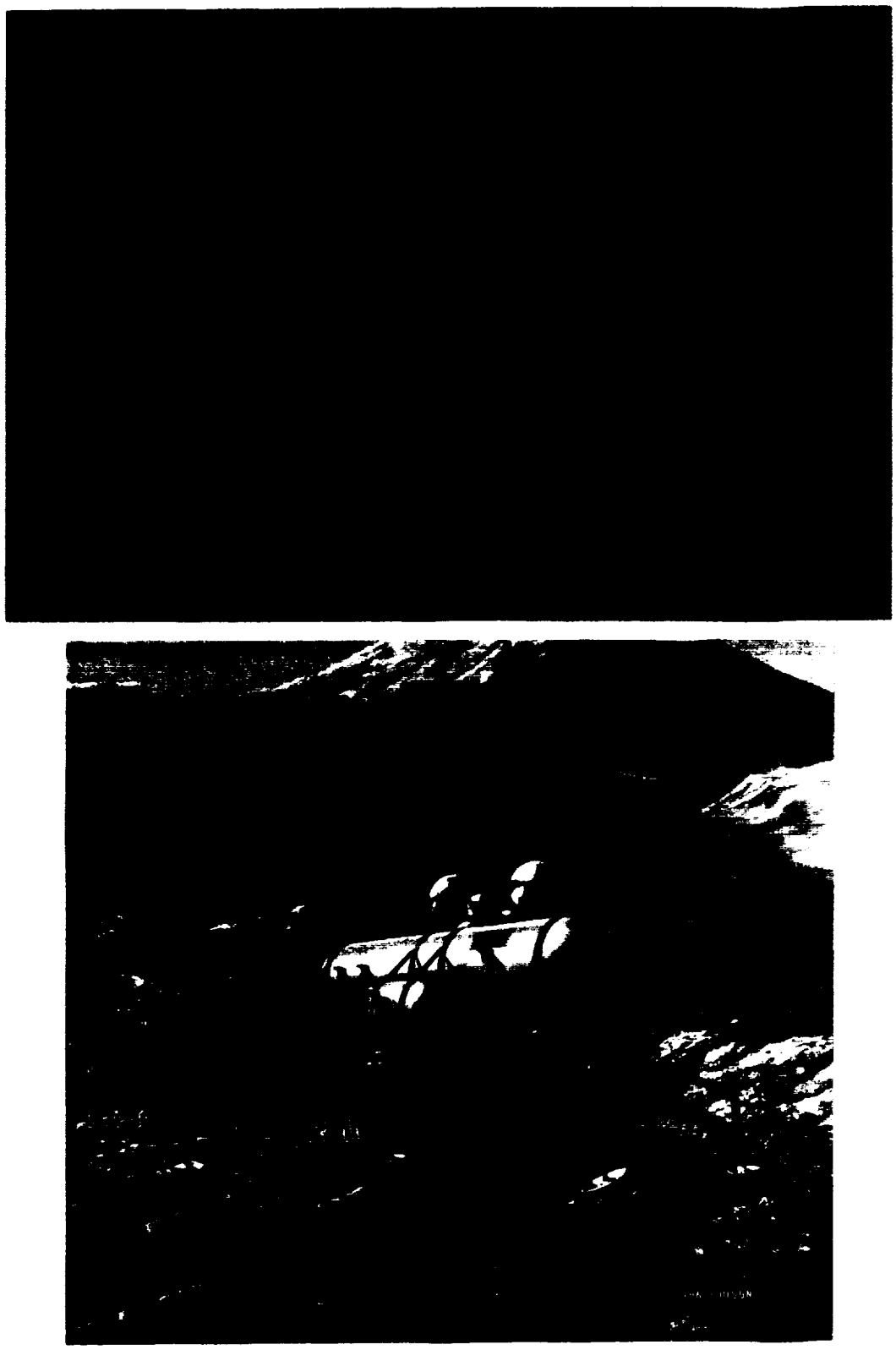
2.2.5.1 Low-L/D Mars Excursion Vehicle (MEV)

The MEV archetype development closely followed requirements generated by the 90-Day Study. It was originally conceived as a means of delivering 4 crew and 25 t of undefined payload to the surface of Mars. However, the specification of crew cab provisions, the analysis of vehicle mass balance, and consequently the configuration design of the vehicle all depend on specifics of the payload manifest. We assumed a 20 t reference surface module as an integral part of the MEV. This led to a "Mars campsite" design (Figure 2-16) intended to support a crew of four for 30 - 60 d. Chief departures from the lunar campsite mode of operation were:

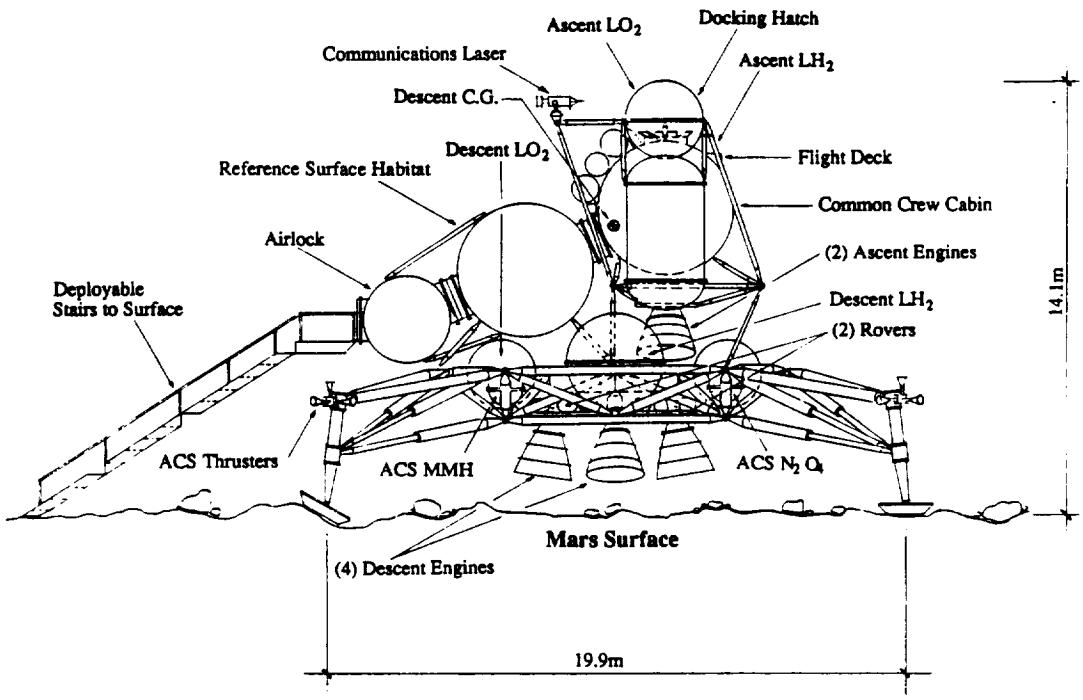
a. The MEV arrives with the crew already on board, and so is capable of a fully self-contained mission.

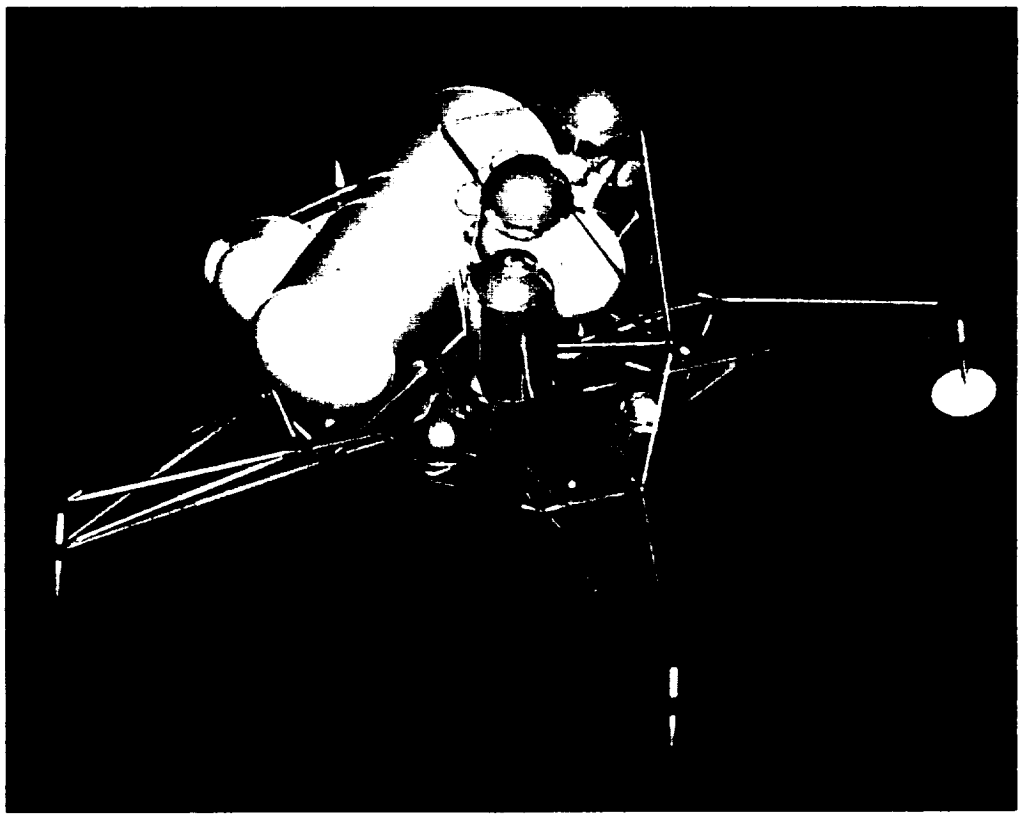
b. The MEV also brings with it an ascent vehicle (MAV) with a separate propulsion system, configured optimally for the ascent phase (or ascent after breakaway from the descent stage during a descent abort). The crew cab for the MAV is the operations bridge for the MEV during all its mission phases.

c. The MEV is configured for packaging within an L/D = 0.5 aerobrake. For CAB missions, this brake captures the as-yet unmanned MEV into Mars orbit autonomously, before rendezvous with the MTV, and is used again for the descent. For CAP and other types of missions with propulsive Mars orbit capture, this brake is used only for descent. In all design cases, terminal descent engines are extended through ports in the windward surface of the brake at low Mach number, and the brake is jettisoned subsequently, prior to touchdown.



LOW L/D MARS EXCURSION VEHICLE (LANDER)
FIGURE 2-16



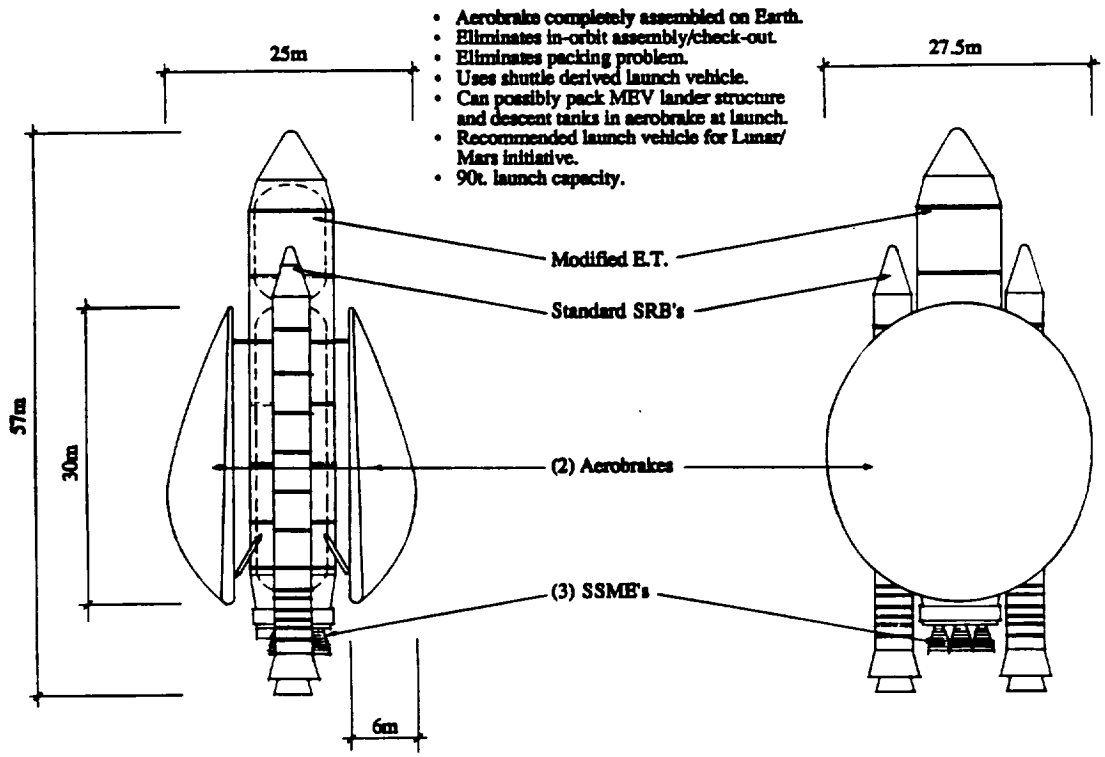


LOW L/D MARS EXCURSION VEHICLE (LANDER) (CONTINUED)
FIGURE 2-16

The MEV configuration was developed to permit later removal and relocation of the surface habitat module, with the aid of surface construction equipment. A variant of the MEV, without either surface module or MAV, was analyzed for delivery of heavy cargo on unmanned missions. A quick assessment was made of the feasibility of re-using an MEV, presuming *in situ* production of oxygen and retention of the aerobrake until touchdown. The assessment indicated that: (1) additional brake hatches appeared necessary for landing gear deployment, crew egress, and cargo offloading; and (2) a lightweight top-shroud appeared advisable due to aerodynamic drag on ascent, and to permit the crew bridge to protrude beyond the presumed wake-protection limit for direct surface viewing during terminal approach. Configuration options for a "split-stage" MEV, in which the same, or a portion of the same, propulsion system is used for ascent as for terminal descent, were also investigated, and shown to be simple variations of the archetype.

Our baseline aerobrake assembly concept presumed robotic-mediated final assembly of pre-finished, rigid aerobrake segments at Space Station *Freedom*. Packaging such segments efficiently by nesting them in an ETO launch shroud is challenging because of: (1) the aerobrake's asymmetrical, deep-bowl shape, in which the maximum depth of a typical "slice" is comparable to reasonable shroud diameters; and (2) the aerobrake's lip, required for both aerodynamic performance and structural stiffening around the free brake edge. Subsequent manifesting analysis, in which segments were configured according to an initial rib-and-spar structure concept, indicated that two ETO flights would be required to launch a single aerobrake in several pieces. Such extremely volume-limited and volume-inefficient manifesting is an unacceptably poor use of the expensively developed capability that a heavy-lift ETO system represents.

In response to this manifesting problem, STCAEM proposed the "integral launch" concept, in which a fully assembled, integrated aerobrake is launched externally, mounted on the side of the launch vehicle exactly analogous to current STS operations (Figure 2-17). The low-L/D brake is comparable to the STS orbiter in linear dimensions, and is light enough to launch two at once, with capacity to spare for other, shrouded payload as well. Ascent performance of such a flight configuration was analyzed and found acceptable; the critical question is whether ascent loads would size the aerobrake structure out of the competitive mass range for the mission itself.



INTEGRAL LAUNCH CONCEPT FOR LARGE, RIGID AEROBRAKES
FIGURE 2-17

Our structural analysis indicates that since the deep bowl-shaped aerobrake tends to load as a doubly-curved shell, it may be possible to construct an actual "aeroshell" without resorting to ribs and spars or some other articulated skeletal structure system. The shell would be made of a relatively thin honeycomb-type material system with integral thermal protection system (TPS). However, lip buckling would still require a stiff rim, probably facilitated by a closed-tube-section structure. Such a brake may be lighter, and certainly simpler, but the thickened rim may still cause packaging problems due to nesting interference.

2.2.5.2 High-L/D Reusable Mars Excursion Vehicle (RMEV)

The RMEV archetype development occurred in response to three drivers:

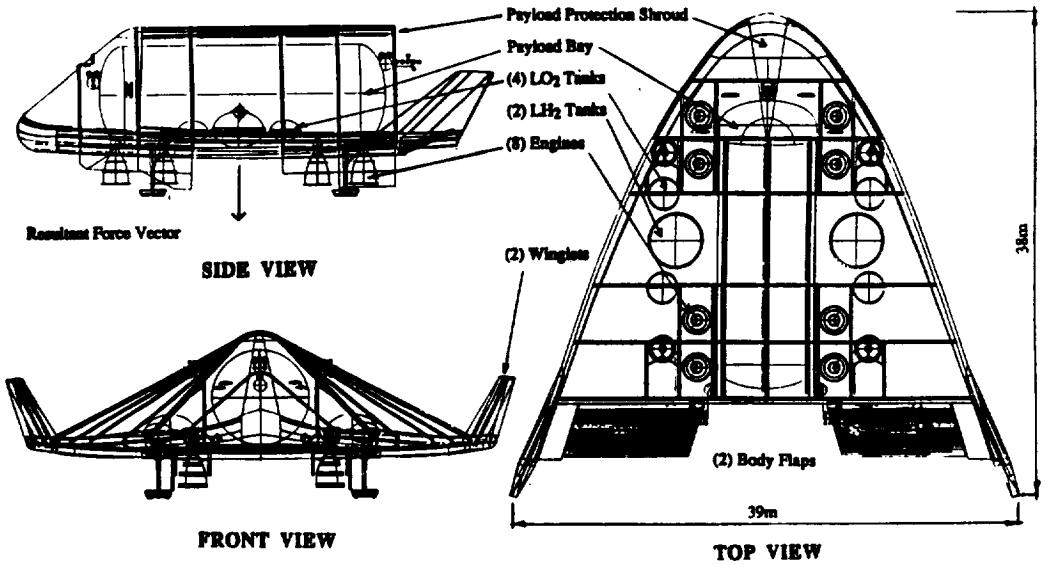
a. Analysis so far indicates that $L/D = 0.5$ is sufficient for controlling an aero-vehicle at Mars. However, the existence of some mission design studies in the literature which advocate $L/D > 1.5$ for Mars, combined with our preliminary understanding of controllability under Mars conditions, make it important to know in detail how different the

configuration constraints imposed by higher L/D would be from those imposed by the lower L/D (which by 1989 had come to be regarded as appropriate).

b. As the 90-Day Study stimulated thinking about what the purpose of SEI Mars surface missions should be, concern developed that global, or at least wide, access to the surface of Mars was potentially important. High-thrust Mars transfer propulsion systems (chemical or NTR) tend to be mass-constrained by arrival and departure vector geometry to certain parking orbit conditions. Although there is no lack of interesting (scientifically important) landing sites accessible from the periapsis of any orbit at Mars, the fact that performance-optimized parking orbits are unique for each high-thrust opportunity causes a site-access problem if returning to the same surface site is required (e.g., for base buildup and operations). Thus for high-thrust transfer propulsion options particularly, an ability to achieve cross-range on lander entry may be important. High L/D enables greater cross-range capability.

c. Certain Mars lander issues not imposed as requirements during the 90-Day Study required analysis and design validation. Developing a new MEV concept (Figure 2-18), substantially different from the baseline MEV, allowed us to investigate those issues simultaneously and thoroughly. Specifically, we addressed: (1) a deep aerobrake structure concept, of interest for maximum structural efficiency and therefore reduced brake mass; (2) the ability to deliver large-envelope cargo manifests, represented in our design by a long-duration surface habitat module sized for 10 crew; and (3) re-usability of the MEV, based on *in situ* production of cryogenic propellant.

The vehicle shape represented by the RMEV has applications for other interesting mission modes, concepts for which have yet to be investigated in detail. Three examples are: (1) a smaller RMEV, sized commensurately with the MEV to be a modest cargo-delivery vehicle; (2) a direct-landing MTV, whose return propellant would be manufactured *in situ* on Mars; and (3) re-usable aerobraked "taxi" vehicles capable of performing the Earth-Mars cycler embark/debark function.

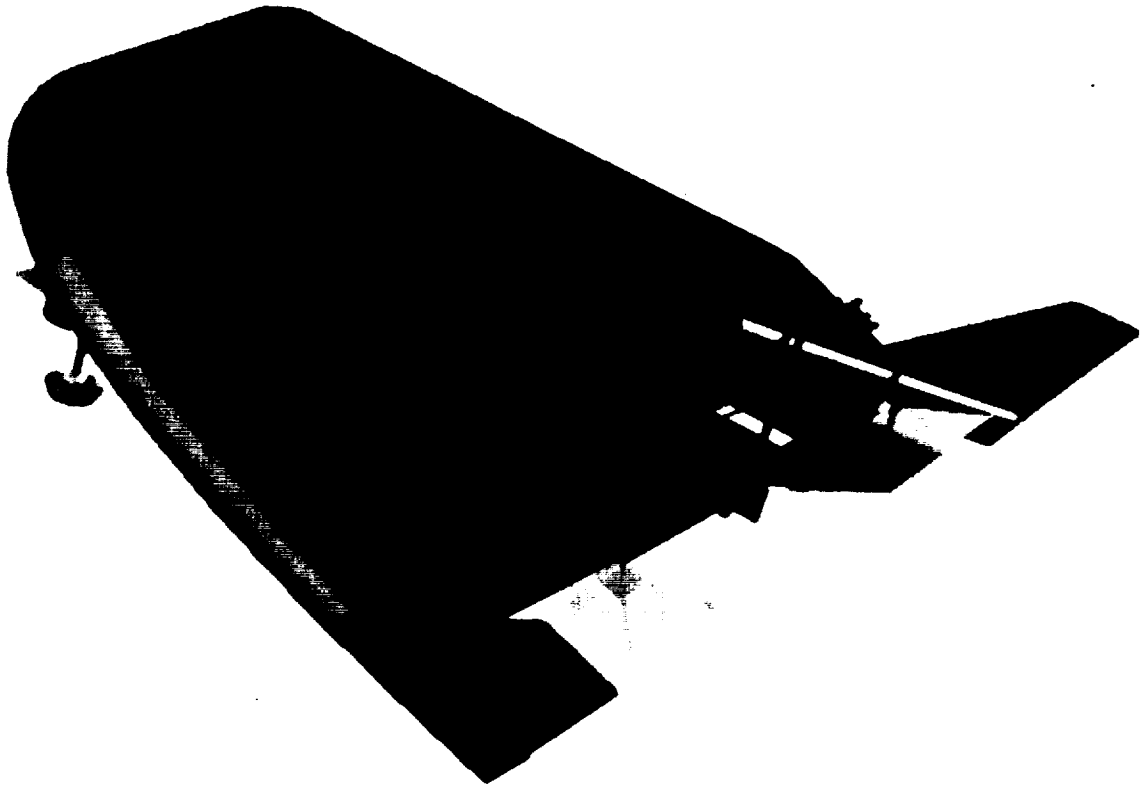


HIGH L/D MARS EXCURSION VEHICLE (LANDER)
FIGURE 2-18

2.2.5.3 Cryogenic/Aerobraked Mars Transfer Vehicle (CAB)

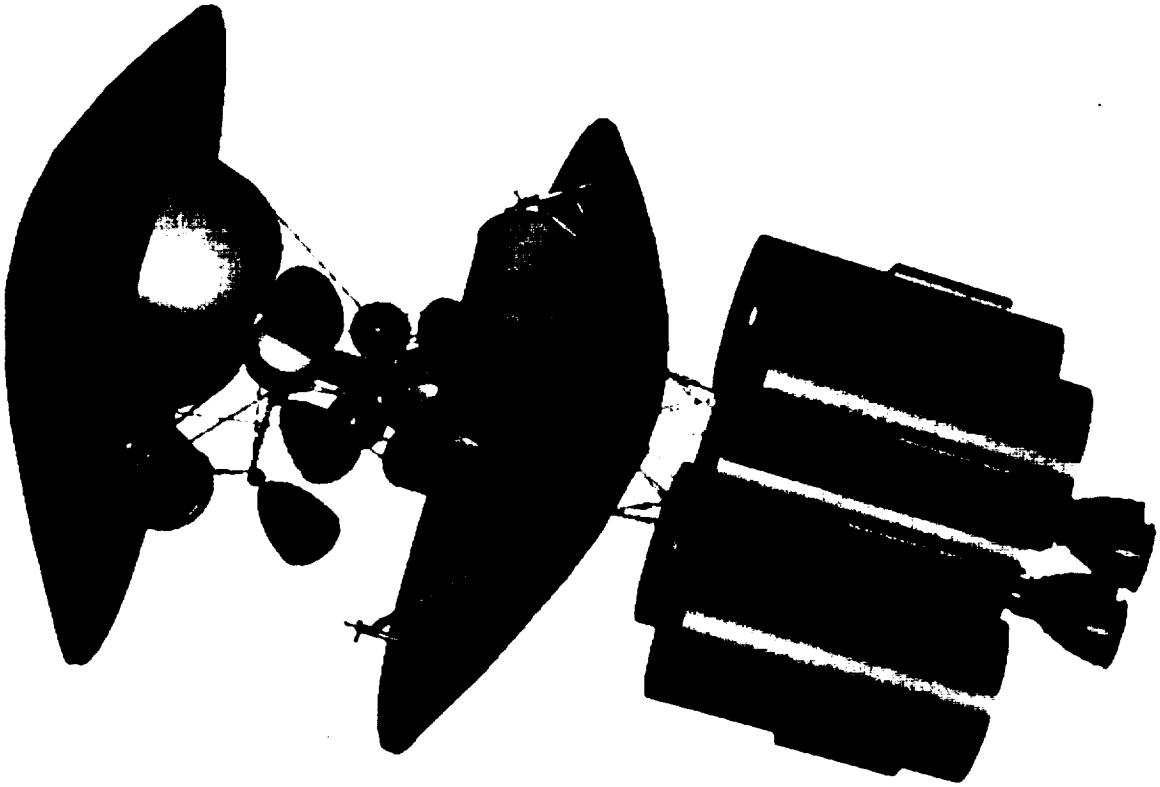
NASA selected cryogenic chemical propulsion, augmented by aerobraking for capture and landing at Mars, as the opposition-profile baseline for the 90-Day Study. The archetype which first resolved the dominant configuration complications for CAB Mars missions already existed (Reference 18). With this foundation the 90-Day Study was able to progress rapidly into performance, subsystem, operations and programmatics analyses. The 90-Day Study exercise in turn enabled refinement and validation of the CAB archetype, Figure 2-19. The major drivers for the CAB archetype are:

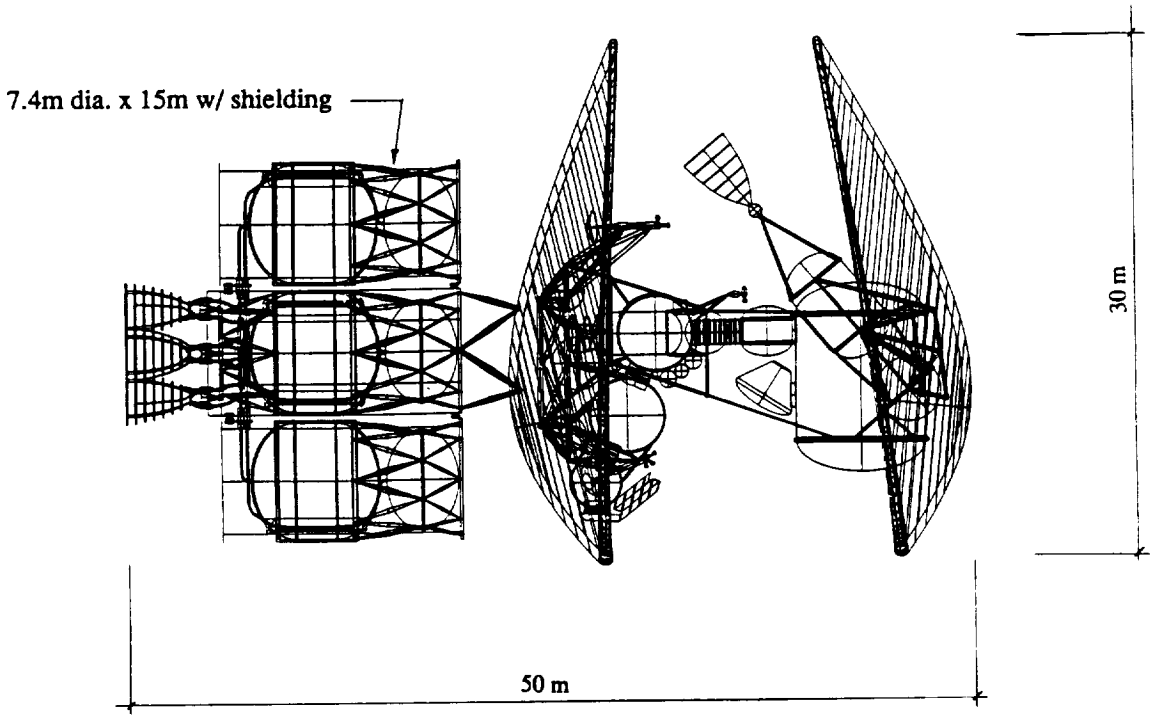
a. High-thrust chemical propulsion: engine-out design to accommodate shifting vehicle mass center as the mission progresses, given the constraints of engine clustering and limited gimbal angle; propulsion system geometry for in-flight testing before critical mission maneuvers; and avoidance when possible of aerobrake penetrations.



HIGH L/D MARS EXCURSION VEHICLE (LANDER) (CONTINUED)

FIGURE 2-18





CRYOGENIC/AEROBRAKED MARS MISSION VEHICLE CONCEPT
FIGURE 2-19

b. High-energy aerobraking: contemporary understanding of aftbody wake closure geometry, and aerodynamic simulation-based constraints on mass center location; mutual independence of MEV and MTV during final approach to Mars space, since each is captured separately; packaging of the entire MTV system in as small a capture aerobrake as possible; potential requirement for MTV brake retention and re-use for Earth capture upon return.

c. Rotating artificial gravity: physiological constraints drive the CAB archetype toward deployable tether schemes because of the effort to make the aerocaptured vehicles as compact as possible. This makes the physical arrangement of the MTV systems difficult, given both a requirement to maintain all habitable volumes (both the MTV habitat and the MEV crew cab and surface module) contiguous during transfer, and the fact that the only rotation countermass available on the return leg is the empty MTV TEI propulsion system.

d. Modular vehicle design: maximize system commonality desired to standardize integration and operations protocols, and especially to accommodate the widely varying energy (propulsive) requirements of opposition-class missions. The opposition missions were designed to collect most of the energy difference in the TMI delta V. This burden was more easily accommodated by the TMIS, which became a highly modular vehicle system.

e. Robotic-mediated operations: facilitating machine access into the densely packaged systems of the CAB vehicle, and designing provision for robotic EVA maintenance during the mission, is a tough but essential requirement. We baselined an operations concept in which manipulator systems could travel around the rims of the rigid aerobrake structures, both to assist in assembling the vehicles at Earth and to service them en route.

A concept called the "Shuttle-Z 3rd Stage" was detailed in response to a Level II trade request. This is a modular version of the TMIS, in which each section uses its engine twice (once for ETO orbit insertion and again for the Mars departure burn). The fundamental problem with the scheme is that, with engines located on each TMIS section instead of clustered in the center, mass-balanced engine-out on TMI is not possible without the addition of an extremely long (120 m) truss to separate the TMIS from the payload mission vehicles.

A configuration trade analysis revealed that avoiding the need for Mars orbit rendezvous upon arrival between a separate MEV and MTV by configuring one large, aerocaptured vehicle was not practicable (either a very large aerobrake, or a reconfigurable cryogenic propulsion system, appeared necessary).

An Earth-Mars cycler vehicle capable of providing periodic transfers between the two planets is one potential mission mode addressed by our architecture assessment. Such a vehicle could take a variety of forms, but for SEI-class missions, the basic function could be accomplished by a variant of on the CAB vehicle. For the conventional cycler profile, aerocapture energies for the "taxi" craft needing to get into parking orbits at Mars are quite high. Re-usable vehicles for this job would probably require heavy and/or complex thermal protection systems.

2.2.5.4 Cryogenic All-Propulsive Mars Transfer Vehicle (CAP)

The CAP archetype is fundamentally a variation of the CAB archetype, but is reported here as a separate archetype because its mission philosophy is quite distinct. The CAP concept was developed in response to two drivers:

a. Exploration of alternative purposes for SEI Mars missions led, after the 90-Day Study, to more in-depth discussions of the merits of conjunction *vs.* opposition profiles. Initial presumptions favored short total mission durations; this approach remained typical after the FY88 and FY89 OEXP study cycles, in which very short, compressed opposition or "split-sprint" mission modes figured prominently. However, given the 30 - 60 d Mars staytime they realistically allow, the ratio of usable surface time to total mission time for opposition profiles is about 10%. After the 90-Day Study, this was recognized more widely as permitting only a relatively disappointing science return on a large engineering investment, exacerbated by the possibility of extrinsic events (like Martian dust storms) precluding landing altogether. By comparison, the same ratio for a typical conjunction mission is about 30%. The top-level costs associated with exploiting the greater opportunity to do in-depth science proffered by conjunction missions are two: (1) the requirement for more elaborate surface payload manifests to support that science (and the crews to conduct it) for year-long stays; and (2) the greater risk to mission completion incurred by having the crews and hardware spending almost 3 yr in deep space instead of about 1.5 yr.

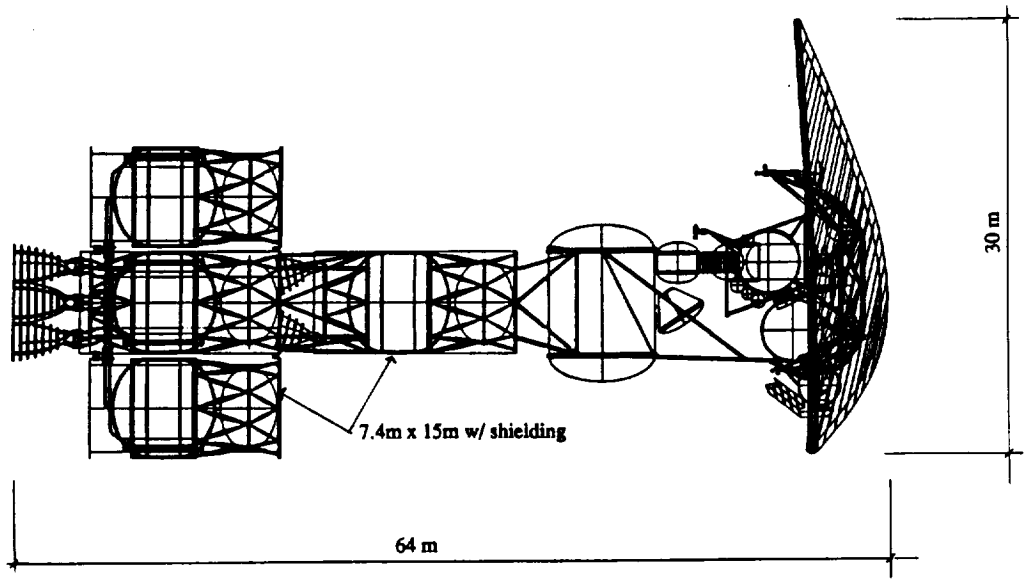
The conjunction profile offers benefits in addition to greater surface staytime. First, the opportunity variation in mission energy requirements is much reduced for the conjunction case, so that mission hardware can be more consistent from one opportunity to the next. This would minimize the actual program upset resulting from a missed opportunity. Second, having of order 300 d available at Mars would permit more flexible mission design. For example, rather than spending the entire staytime on the surface, the mission might carry multiple landers each destined for short visits to widely separated surface sites (or crew rescue at a given site). And finally, although conjunction missions are roughly twice the length of opposition missions, the bulk of that difference can consist of time spent on the surface of Mars, under the radiation shielding afforded by the Mars atmosphere. The actual in-space transfers are about equal in length outbound and inbound, and their total is less than the total in-space transit time for typical opposition missions. Thus in scenarios required to minimize astronaut exposure to in-space galactic cosmic radiation (GCR), well-designed conjunction missions are of great interest. (Trip times can be shortened further still, until the so-called "conjunction fast transfer" mission energy requirements approach those for opposition missions.)

Conjunction low-energy missions do not benefit from HEAB, so these missions need only carry aerobrakes for entry and landing. Performing Mars capture with cryogenic chemical propulsion leads to three fundamental distinctions between CAP (Figure 2-20) and CAB concepts:

a. The MTV and MEV(s) are captured together, precluding the possibility of failure to rendezvous and consequent elimination of landing attempts.

b. The Earth-departure (TMI) stage grows into a multi-staged propulsion stack, with TMI, deep-space burn (DSB), and Mars arrival (MOC) stages. This changes the overall aspect ratio of the all-up vehicle, making it longer, which has implications for attitude control and debris shielding in LEO.

c. Relaxing the requirement for the MTV to be an aerobraked vehicle means that the systems constrained in the CAB case to be packaged behind an aerobrake can be distributed differently. Thus the Mars-departure (TEI) propulsion system can be combined with the MOC system and placed at the opposite end of the vehicle from the MTV habitation system and payload. This in turn means that rotation for artificial gravity can be accomplished



CRYOGENIC ALL PROPULSIVE MARS MISSION VEHICLE CONCEPT
FIGURE 2-20

simply by configuring a long, lightweight truss between the propulsion system and the payload, and spinning this rigid assembly end-over-end. Tethered solutions are not required because aerobrake packaging is no longer a problem. This last set of CAP consequences departs from the CAB concept sufficiently for their resolution to constitute a distinct vehicle archetype.

2.2.5.5 Nuclear Thermal Rocket

Nuclear thermal rocket propulsion had a long, successful development history in support of post-Apollo human space exploration, and still occupies a uniquely validated position as a candidate for advanced propulsion for SEI Mars missions. Consequently it was the first advanced propulsion option investigated in depth as a consequence of the 90-Day Study. Much of the NTR technology discussion centers on sophistication, ranging from resurrections of the original NERVA design, through upgrading with modern materials,

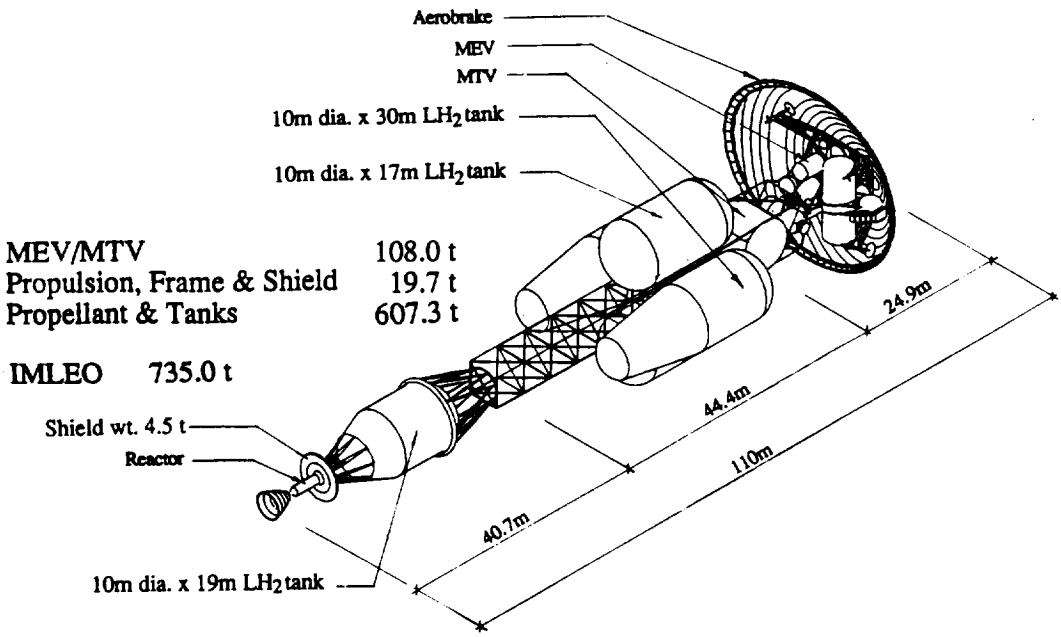
and further through new particle-bed designs with enhanced Isp, to liquid-core and gas-core rockets. However, this wide range of alternatives is satisfied by one NTR archetype; to first order, only the amount of liquid hydrogen propellant varies among them. The overall vehicle configuration, requirement for shadow shielding of the payload from scattered radiation, long-term storage of LH_2, and solutions for providing artificial gravity, all remain constant. The NTR archetype has existed since the NERVA days; our work has provided analysis of four particular enhancements (Figure 2-21): (1) the use of a truss spine and drop tanks instead of a large, structural, axial tank to reduce inert mass; (2) the configuration detailing associated with providing dual engines for engine-out reliability; (3) using a single NTR vehicle to deliver multiple landers to Mars; and (4) truss-spine elongation and careful positioning of large drop-tanks around the mass center to accommodate artificial gravity during all coast phases via end-over-end spinning.

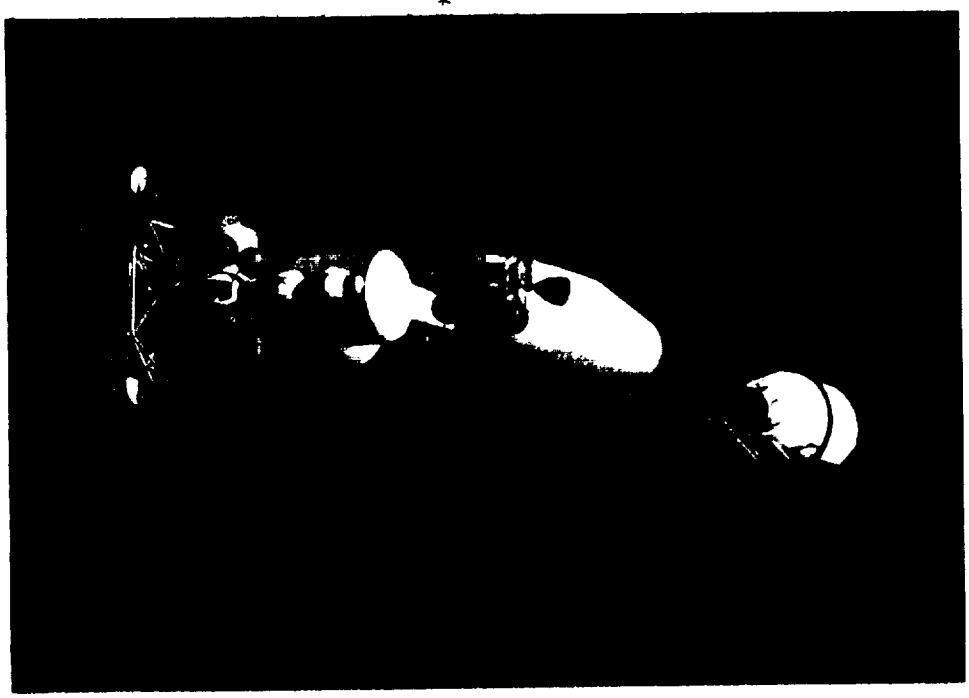
2.2.5.6 Nuclear Electric Propulsion

Nuclear electric propulsion represents a power-rich approach to extremely efficient, low-thrust propulsion for long range missions. The NEP concept archetype we developed specifically addresses several important system interactions:

a. We started with power plant schematics and state-point characterizations from Rocketdyne (Reference 19). To these we added mission performance requirements consistent with the rest of the STCAEM study, and developed a complete vehicle concept that could be modeled, measured and specified in detail. The result is the first NEP concept to detail the power system plumbing, from reactor to radiator, integrated with other subsystem concerns.

b. The high equipment density and challenging operating conditions of a dynamic power conversion system introduces concerns about mission safety due to meteoroid impacts and equipment reliability, respectively. Redundancy solutions to the critical power equipment failure problem (analogous to redundant valving and manifolding for chemical propulsion systems) introduce complex plumbing implications for NEP.





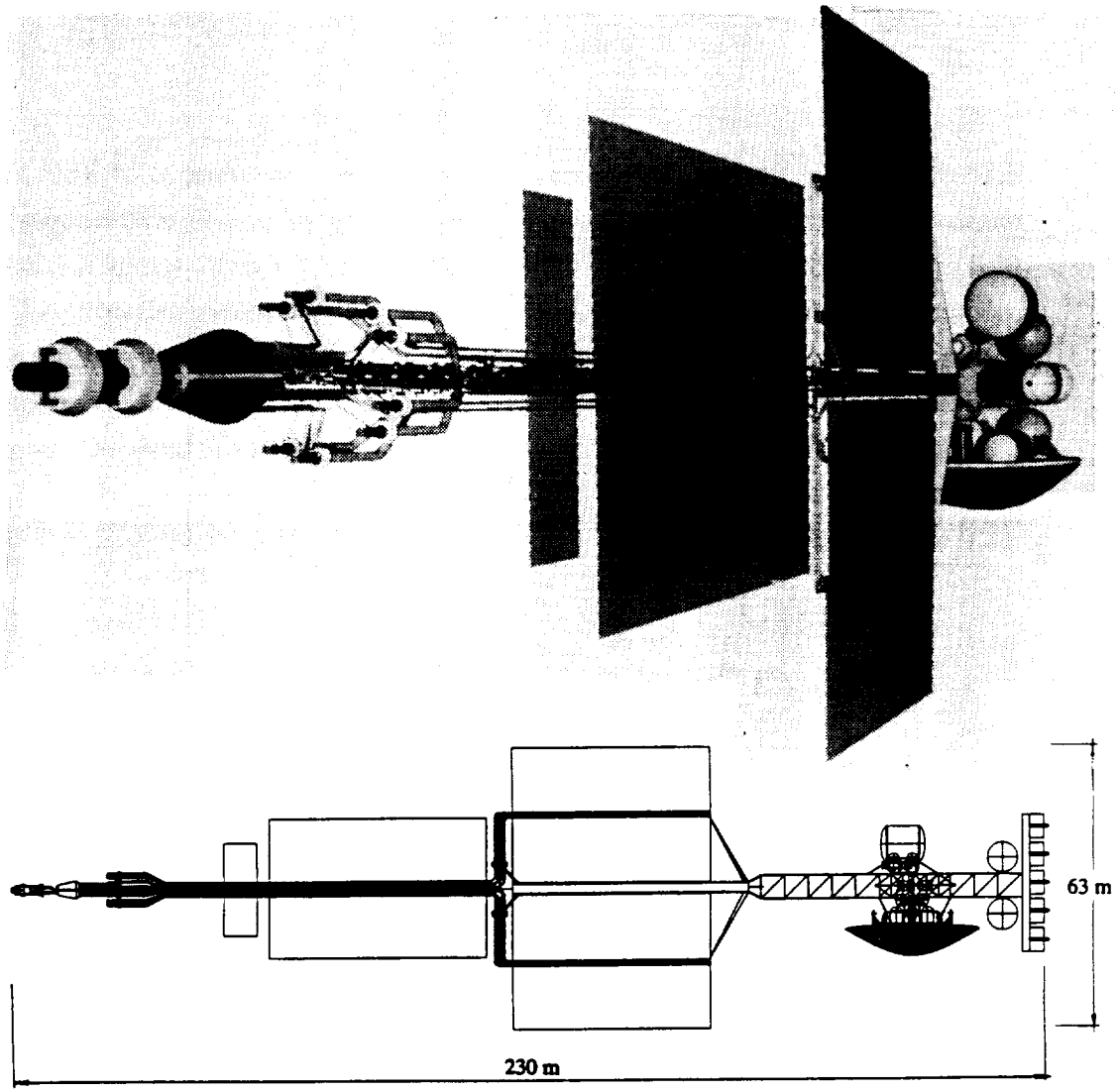
NUCLEAR THERMAL PROPULSION MARS MISSION VEHICLE CONCEPT
FIGURE 2-21

c. The most immediately recognizable feature of NEP initial concepts in the exploration mission literature is their large radiator area, typically shown simply as a conical device following the contour of a protected zone behind a small radiation shadow shield. Engineering analysis to develop a modular, buildable radiator subsystem integrated with the rest of the vehicle, and to minimize shield mass, challenges this simplified picture.

d. The real possibility of an eventual requirement to provide essentially continuous artificial gravity during thrusting portions of an electric propulsion mission leads directly to serious configuration complications. In particular, precession of the angular momentum vector, and transfer of high electrical power levels across rotating joints pose challenging concept and technology problems.

At first we tried to develop a vehicle concept that could easily optimize for both microgravity and artificial gravity mission profiles, by having its thrusters at the center of rotation, thrusting normal to the vehicle's long axis. The geometry requirements proved incompatible, and we subsequently allowed designs for the two types of missions to diverge. The microgravity version became an axial vehicle, with engines at the stern, payload attached around the spine, and power system at the bow. From the reactor's standpoint, the entire vehicle looks like a thin line; this permits a small, carefully-shaped shadow shield to be used, which limits its mass. The artificial gravity version was much more complex, consisting fundamentally of the addition of a cross-axis outrigger amidships for the engines so they could be despun and located near the axis of rotation. This configuration went through many stages during which detailed alternatives were sequentially explored. It is reported on more fully later in this section, along with the other artificial gravity concept development results.

Our archetype (Figure 2-22) features two redundant reactors, five identical power conversion systems (2 of which are spares), and large expanses of stiffened planar radiator panels, comprised of over five thousand identical, finned, 30 m long, liquid-sodium heat pipes. The structural spine of the vehicle is a lightweight truss, along which are arrayed all the armored fluid-carrying loops of the power production and conversion system. The "front end" of the vehicle, containing reactors and dynamic conversion machinery, was configured both to allow straight-line access for robotic maintenance activity and also packaging in a 10 m launch shroud. The detailed design of this power plant as it will appear in the next-generation integrated NEP concept is shown in Figure 2-23. Thus the power system itself can be integrated on the ground, and requires liquid-metal-temperature joining at pipe interfaces to the heat rejection system assembled on orbit. The integration of a large NEP vehicle represents an unprecedented orbital operations challenge, which dwarfs the assembly complexity of SSF. The NEP's superior mission performance thus comes at a high operations and infrastructure price.

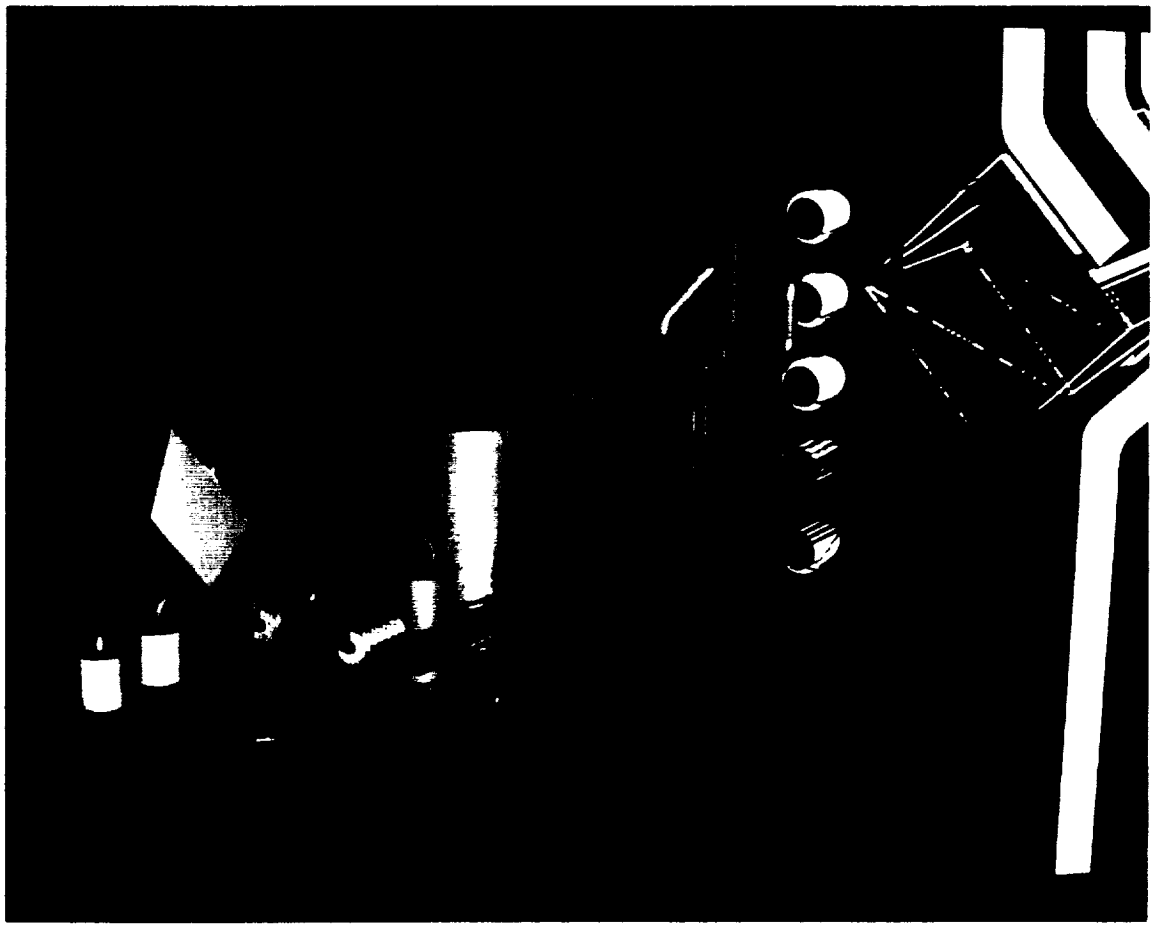


NUCLEAR ELECTRIC PROPULSION MARS MISSION VEHICLE CONCEPT
FIGURE 2-22

2.2.5.7 Solar Electric Propulsion

Solar electric propulsion represents a non-nuclear, "decentralized" and extremely redundant approach to advanced propulsion for SEI missions. It is not, however, a "low-tech" solution to Mars transportation as is commonly held. First, the technology associated with large electric engines is the same for SEP as it is for NEP, because in each case individual 1 MWe engines are ganged together to achieve the appropriate power level. Second, SEP challenges our lightweight, large space structures (LSS) technology more than any other SEI concept. The reference SEP is 203 m on a side, covering an area equivalent to 9.24 football fields; yet its supporting structure must have a mass on the order of only 15 t. Surface accuracy requirements are orders of magnitude less stringent for the SEP photovoltaic (PV) arrays than for high-precision, large space antennas studied in the LSS

literature, but the design, fabrication, deployment and maintenance of SEP-scale LSS remain unvalidated empirically. Third, the size, fragility and unit-repetition appropriate for SEP concepts absolutely requires robotic-mediated maintenance. (However, addressing robotic requirements for SEP may help us face up to the necessity and utility of state-of-the-art automation for other vehicle concepts as well). Finally, high-performance, robust, sufficiently lightweight and low-cost PV assemblies have yet to be demonstrated. The usefulness of SEP hinges critically on our ability to fabricate acres of advanced PV assemblies economically.



NEP POWER CONVERSION PLANT
FIGURE 2-23

Our earlier versions of SEP vehicle concepts presumed motorized unfurling of diaphanous, flexible PV blankets across a skeleton of ribs diagonalized by cables. Engine plume impingement of the arrays and structure was avoided by locating two engine assemblies at opposite ends of a long, truss outrigger. The thermal rejection system for the electrical power management and distribution (PMAD) system was centralized in two areas with

dedicated radiators. Further investigation of current thinking on practical concepts for LSS led us to adopt the area-truss approach as the only way to get requisite stiffness and remain lightweight. The bay size selected was 7 m, as this limits parts count while not exceeding a reasonable span for projected, strengthened PV blanket technology. The blankets themselves consist of an iso-stress mesh of kevlar fibers to which are bonded stiffened, 4 cm advanced tandem solar cells. The need for engine outriggers was avoided by locating the twin engine assemblies at opposite corners of the square vehicle structure. The vehicle thus sweeps back at 45° angles from the nominal thrust line, and our presumed impingement envelope (a combination of ±20° for plume spreading and ± 20° for engine gimballing) was only 40°. To first order, the thrust line is in-plane with the vehicle because the solar arrays must be sun-facing, while the thrust must average tangential to the transfer orbit.

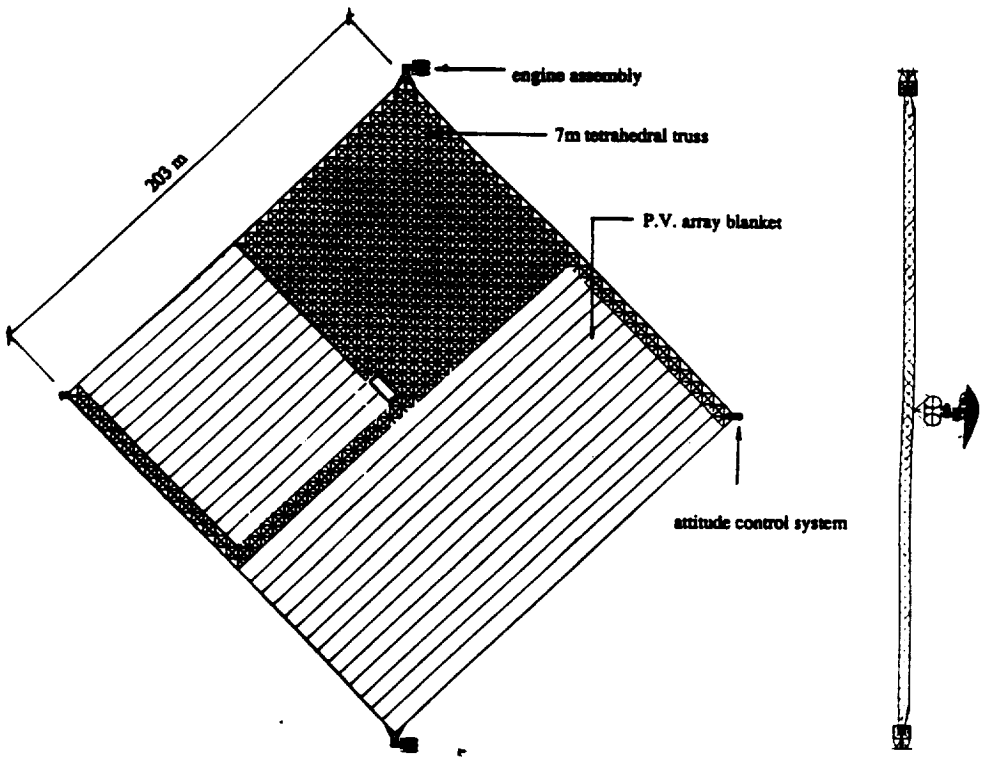
Our SEP reference vehicle (Figure 2-24) has an extremely large number of identical parts, and was developed along with a matching robotic assembly, deployment and maintenance scenario. Two kinds of robots are envisioned: (1) a dextrous truss-builder with the ability to move about the vehicle, top or bottom, inspect critical systems and change out defective components; and (2) an array-paver, capable of accepting cassettes consisting of pre-integrated, rolled PV blankets. The paver would attach the blanket to the vehicle structure, removing and rolling up the blanket's protective packaging sheet as it progressed in one-bay-wide strips. On the SEP's first flight, the paver would deploy the sacrificial transfer array, undeploy it once beyond the van Allen belts, and subsequently deploy the full, main array for interplanetary flight.

2.2.5.8 Artificial Gravity

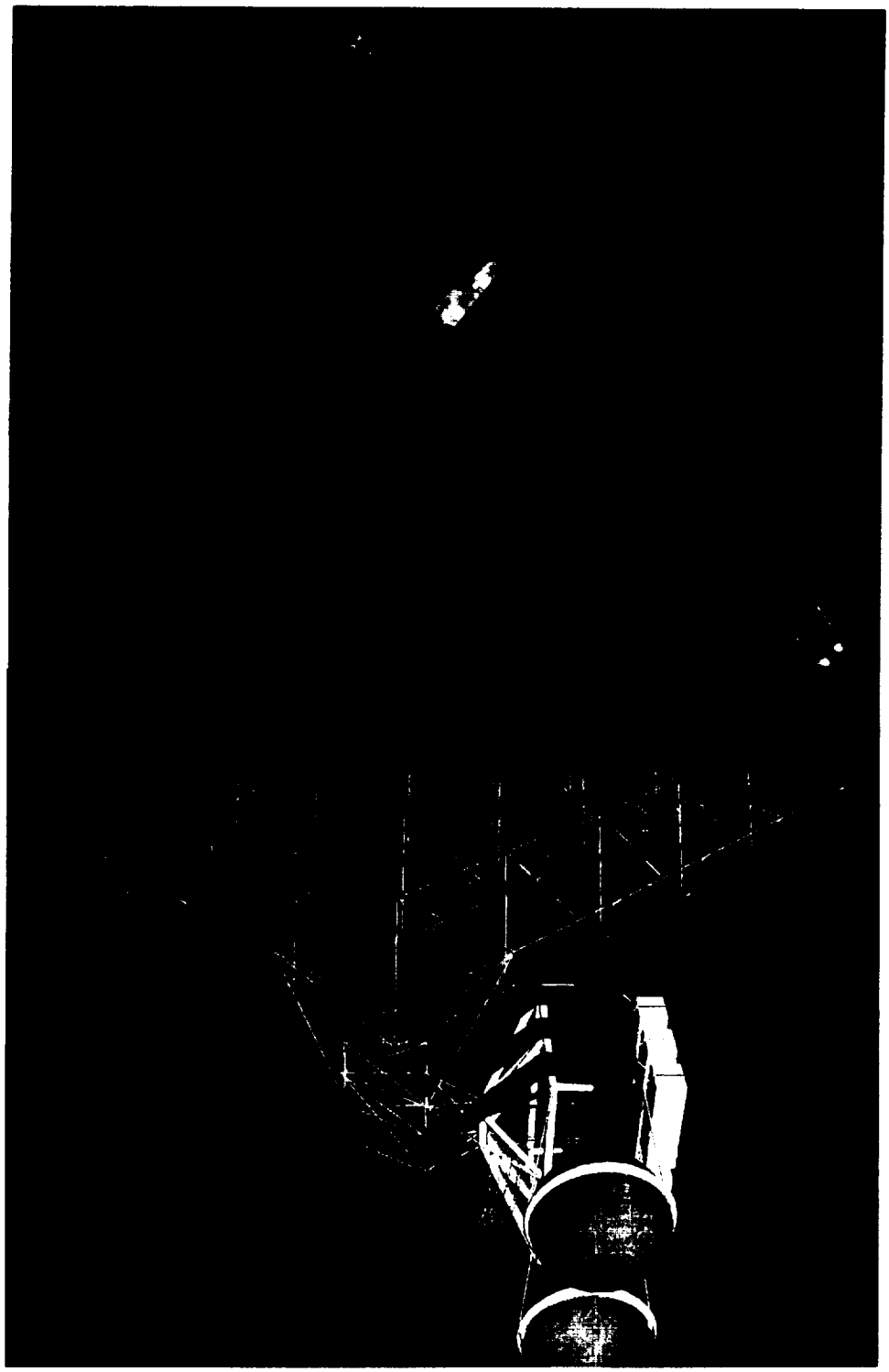
The need for artificial gravity on long-duration interplanetary transfers has not been established. Neither has the lack of such a need, however, so we examined the penalties incurred by requiring continuous artificial gravity en route between Earth and Mars. Various approaches to rotating artificial gravity have been proposed; we assessed all of them, and invented some new ones. The fundamental design problems associated with artificial gravity derive from: (1) the need for a countermass for rotation; and (2) the high mass cost of precessing the angular momentum vector of a system having large rotational energy. Elegant solutions to both are elusive, and vary widely with propulsion option. Secondary complications are communications and navigation pointing, flight structures sized to hang heavy vehicles, and material fatigue due to cyclic loads. The

fundamental operations problems associated with artificial gravity involve crew EVAs during rotation, robotic maintenance in the vehicle's gravity field, crew physiological and psychological responses to a rotating environment, performing minor course-correction propulsive testing the capability prior to departure. Our work has verified that artificial gravity appears feasible for Mars-class missions, for all propulsion options, at fairly modest mass penalties.

The CAP and NTR archetypes accommodate artificial gravity easily. Both are high-thrust systems, so their burn times are extremely short (minutes to hours) compared to coasting transfer time (months). Critical propulsion maneuvers can occur during nonrotating periods of microgravity, at the cost only of spinup/spindown propellant. In general, the propulsion system remaining through the end of the mission can serve as countermass to the contiguously connected habitation systems.



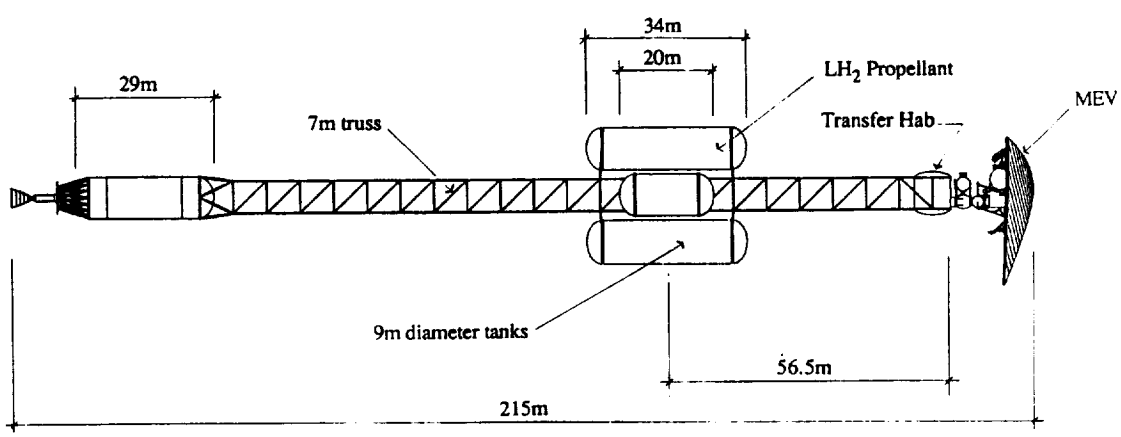
SOLAR ELECTRIC PROPULSION MARS MISSION VEHICLE CONCEPT
FIGURE 2-24



SOLAR ELECTRIC PROPULSION MARS MISSION
VEHICLE CONCEPT (CONTINUED)
FIGURE 2-24

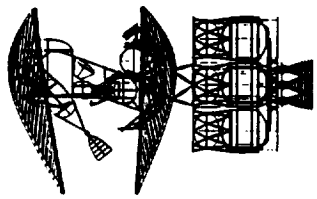
When separated by a lightweight truss (Figure 2-25), they can just spin end-over-end during coast phases to provide sufficient gravity at a comfortable spin rate with acceptable vestibular disturbance (we baselined 1 g to insure full conditioning for surface activity upon arrival at Mars, and 4 rpm maximum spin rate, which together lead to a 56 m separation between the hab and the center of mass). The additional mass of the truss and propellant for a few budgeted spinup/spindown cycles is of order 10 % of IMLEO.

The CAB archetype involves more complexity. The MTV habitat must be contiguous with the MEV crew modules, and yet for the return trip the (empty) MTV propulsion system is the only available countermass to the MTV habitat. Thus the MTV hab and the MTV propulsion system must be separated by a few hundred meters; however, the entire MTV must also package behind an aerobrake for capture at Mars. One solution we rejected for mass and habitability reasons splits the transfer habitat system in two halves, held when not aerobraking at opposite ends of a deployable tunnel. A more sensible approach is to use tethers, configuring the MTV systems such that they are properly mass-balanced for propulsive burns and aerocapture, but can slip apart as the tethers are unreeled for artificial gravity, Figure 2-26. The center of rotation provides a convenient location for a despun power/navigation/communications utility.

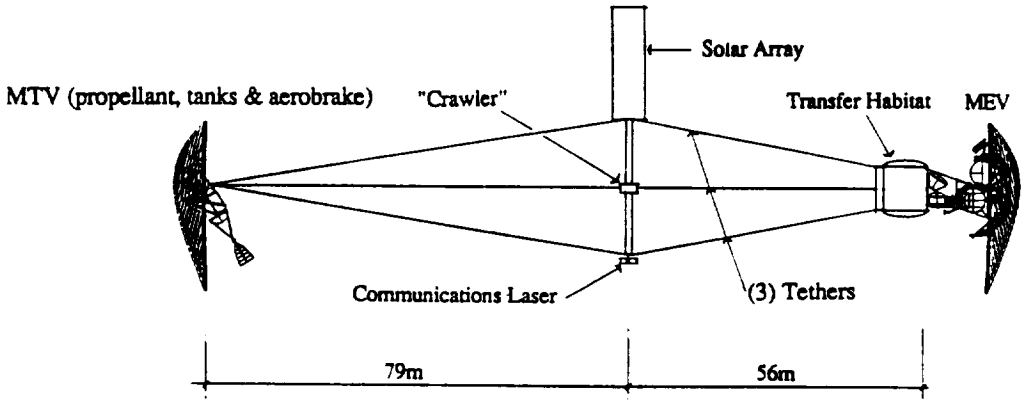


NTR ARTIFICIAL GRAVITY CONFIGURATION
FIGURE 2-25

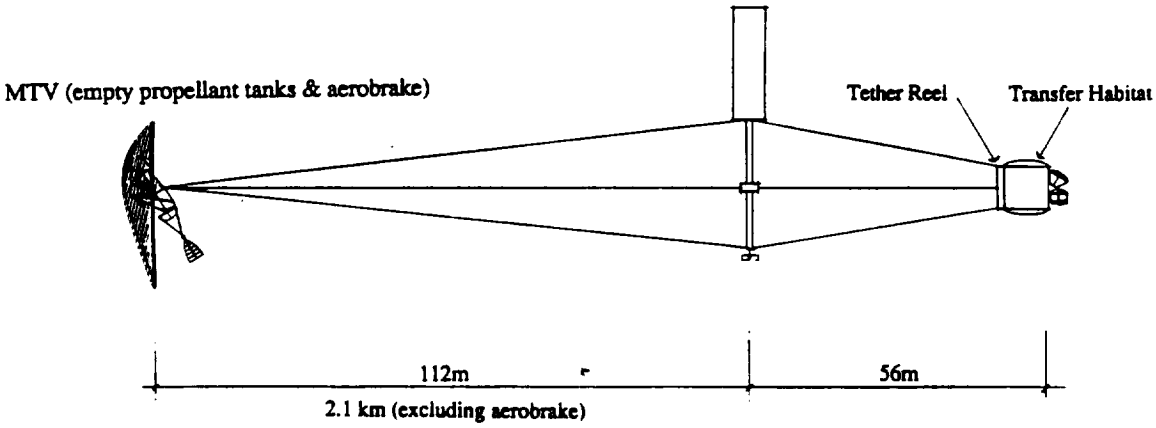
TMI Configuration



Earth to Mars Configuration



Mars to Earth Configuration



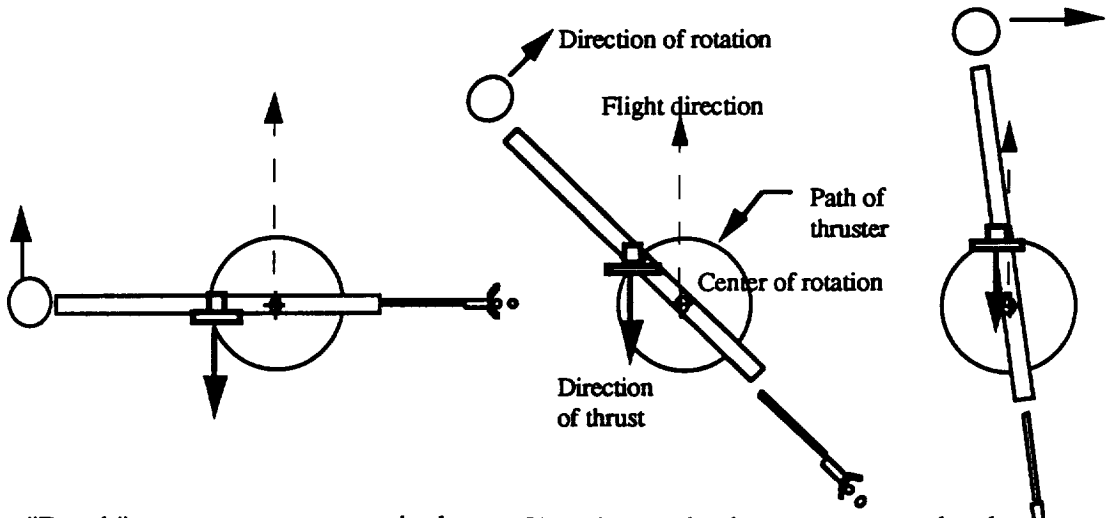
CAB ARTIFICIAL GRAVITY CONFIGURATION
FIGURE 2-26

Vehicles based on electric propulsion pose the toughest integration challenge for artificial gravity. Being low-thrust systems, they must burn for a substantial fraction of the transfer time. One simple approach is to rotate the vehicle only during the mid-transfer coast period (1 - 2 months) and upon arrival at Mars (if a conjunction profile is used to allow long stay times in Mars space). In case intermittent artificial gravity is an insufficient solution, however, it is important to develop full-blown alternatives. The study examined several configuration options. A simple approach would be to keep the thrust vector attitude constant in space, avoiding a need for spin-vector precession. Required thrust vector histories for low-thrust transfers are not completely understood at this time. To first order, however, it appears that repointing would be required, and it is expensive propulsively. We examined using a "cross-product" electric engine located on a long outrigger; even with generous configuration assumptions, the mass penalty is about 10% of IMLEO. If the spin vector is normal to the transfer plane, little repointing would be required, and we selected this option for both NEP and SEP.

We solved the problem of what to use for countermass (particularly acute for the SEP) by baselining a new invention which we call the "eccentric rotator," Figure 2-27. With this approach, everything on the vehicle except the habitable and payload systems *is* the countermass. This leads to the despun electric engines themselves tracing out small circles rather than lying along the spin axis. However, their attitude (all that counts for low-thrust propulsion) can remain constant, and the CM excursion is typically small - of order a few meters for NEP (Figure 2-28) so the gravity loads on the propulsion system are small. Although we developed the artificial gravity NEP as an integrated concept, the dynamics of such rotating vehicles are not yet fully studied. In particular, the practicality of suppressing cross-axis momentum coupling actively, in order to maintain stable rotation, is questionable for spin frequencies of order 3 rpm. Mass penalties as well as trip-time penalties appear small, of order 5% of IMLEO for NEP including a spinup/spindown propellant budget presuming efficient electric thrusting for that purpose.

The SEP eccentric rotator (Figure 2-29) suffers more complications because its distributed structure is so fragile. Effects of the 4 rpm cyclic loading, and the bending moment introduced into the fragile structure by the unbalanced rotor, remain unstudied. Gravity loading of the main truss structure in the eccentric rotator configuration is as high as 0.46 g, and preliminary estimates of the vehicle's structure mass were increased 20% over the microgravity version to accommodate this (because the SEP structure amounts to

only 14% of the vehicle inerts, however, this results in an inerts increase of 2.6%). Solutions using a small countermass deployed on a long tether are probably most appropriate for SEP.



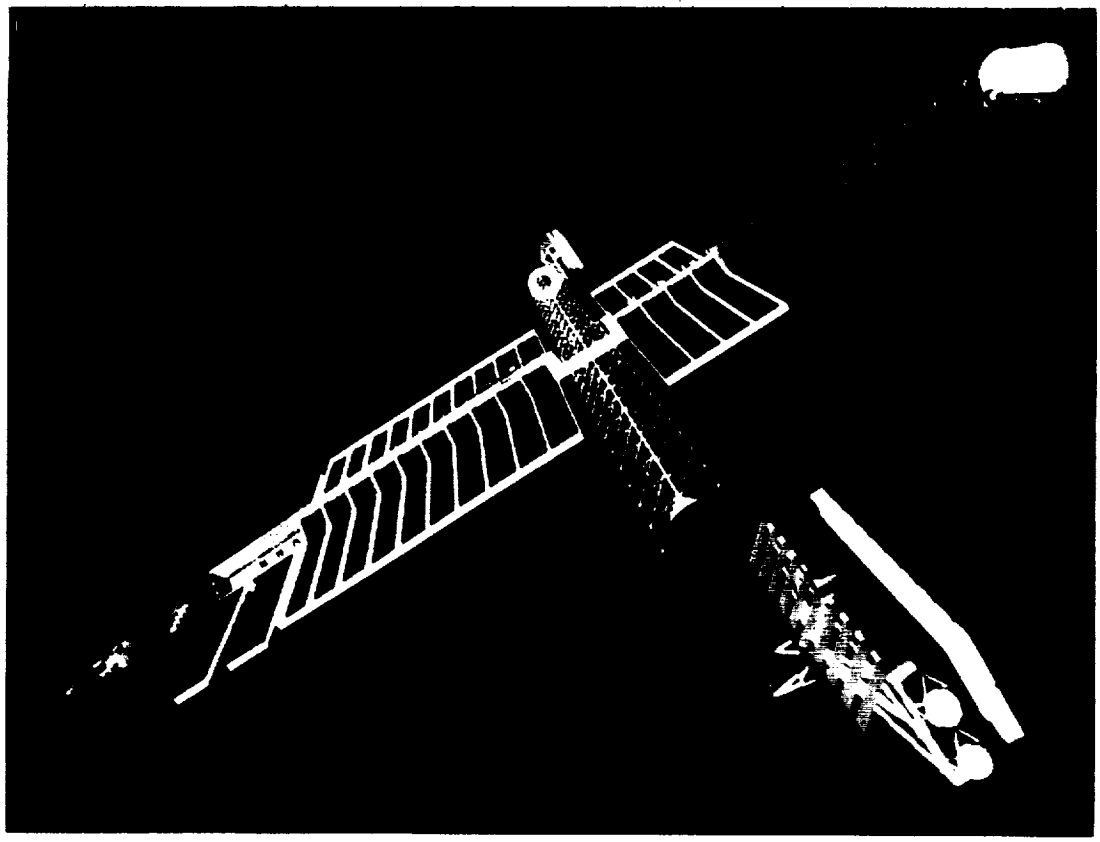
- "Dumb" countermass not required
- Direction of thrust remains constant
- Habitation radius can be fixed
- Changing payload mass accommodated through changing rotation rate
- Minimizes payload structure

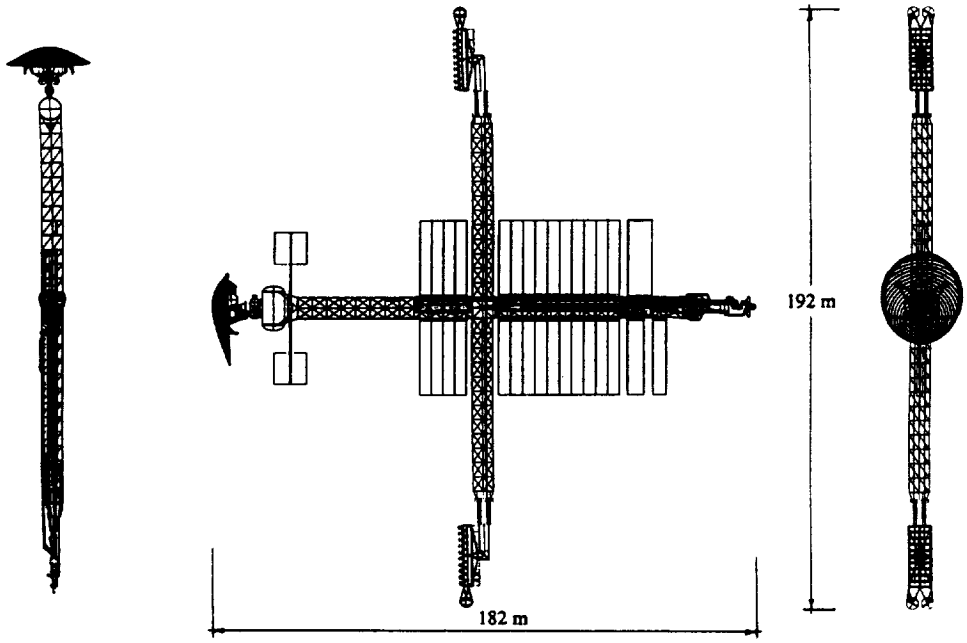
ECCENTRIC-ROTATOR SOLUTION FOR CONTINUOUS EP ARTIFICIAL GRAVITYY

FIGURE 2-27

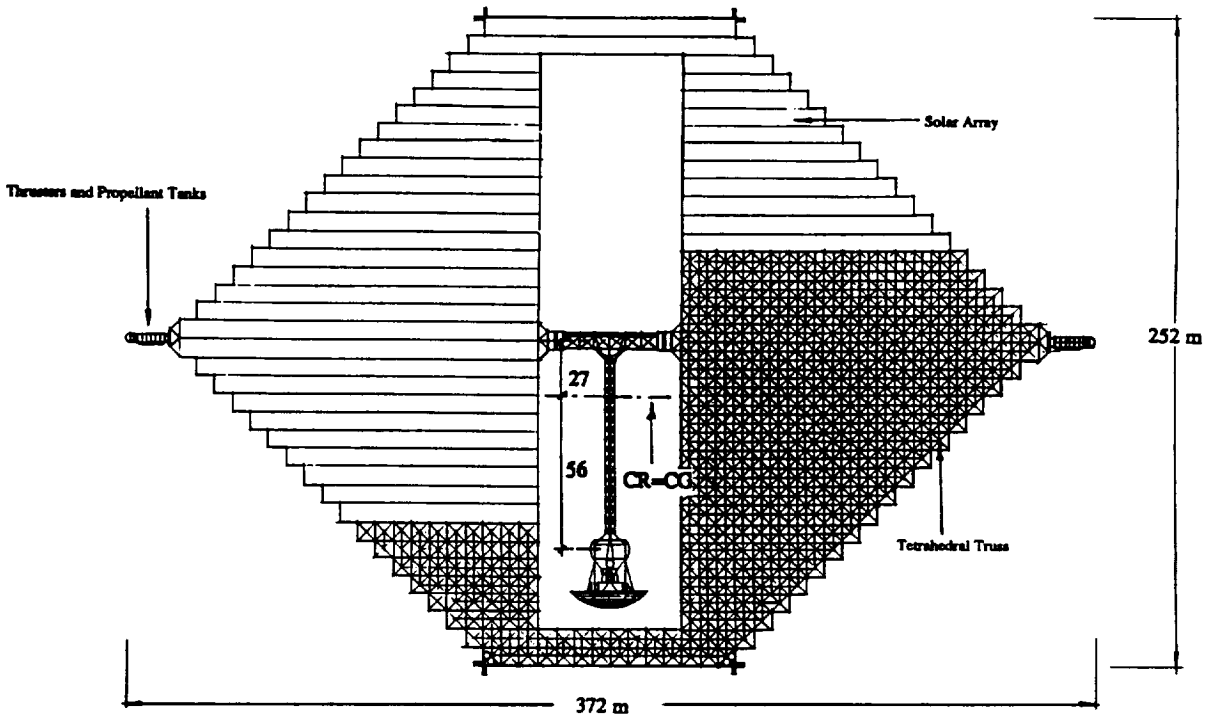
2.2.5.9 Large Crew Sizes

From the time of von Braun's Das Marsprojekt through the 90-Day Study, acceptable crew sizes for Mars-class missions decreased from 70 to 4. That shift can be credited partly to our sobering experience with the true complexity of advanced space exploration; partly to advances in robotic science and the automation of formerly human jobs; partly to attention to real, modern space budgets; and partly to implicit changes in our conception of what the exploration of planets is all about. Nonetheless, STCAEM concluded that a crew size of 4 is too small to be practical or safe. Twice that number, or 8, provides a realistic, minimum skill mix. Our long-duration habitat trade study generated a singular module concept capable of supporting 12, more if clustered together. Twice 8, or 16, may be about right for a bevy of really specialized scientists to investigate Mars during conjunction-class surface stays. And twice that, or 32, approaches a crew size range appropriate for transporting settlers to the red planet. Since we do not yet know what visions will come to





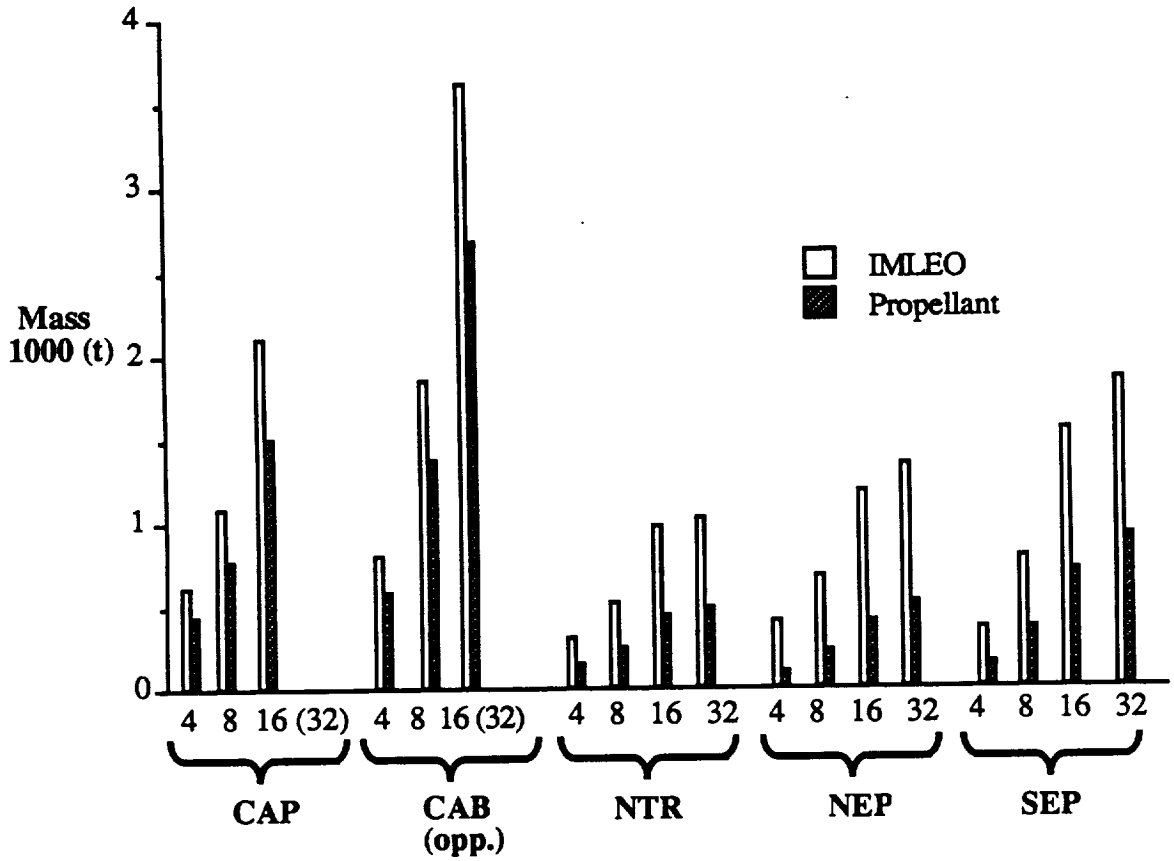
NEP ARTIFICIAL GRAVITY CONFIGURATION
FIGURE 2-28



SEP ECCENTRIC ROTATOR CONCEPT
FIGURE 2-29

guide SEI as it grows, we need to apply our modern understanding of Mars mission technologies, consolidated so far in the STCAEM reference in-space transportation concepts keyed to just 4 crew, to larger crew sizes.

We performed an evolutionary Large Crew Size impact assessment for crew sizes of 8, 16, and 32, looking at Mars mission masses and vehicle strategies for all five prime propulsion candidates, aerobraking constraints, habitat system clustering and staging implications, vehicle configuration impacts, and life support strategies. As expected, advanced propulsion has high mass-limiting leverage for Mars missions, as does the use of conjunction profiles, Figure 2-30. Both CAP and CAB (the latter flying opposition profiles, of course) are not cost-effective for the large payload masses required by large crew sizes. Because SEP power level scales linearly with area, that option appears better suited to flotilla approaches than a "large-vehicle" approach, although this is sensitive to trip time requirements. NEP scales very well, as does NTR. Aerobraking was found to be theoretically feasible at Mars for vehicles of order 64 times heavier than our reference vehicles, assuming similar geometries; therefore, limitations on aerobraked vehicle size are intrinsic, having to do with assembly and trim during flight rather than atmosphere properties. Beyond a certain crew size, (assumed to be between 16 and 32 in this



MISSION MASS SENSITIVITIES TO LARGE CREW SIZES
FIGURE 2-30

assessment), the use of RMEVs pay off. In settlement scenarios where 28 of 32 crew are left at Mars, staging a modular transfer habitat system both reduces return payload and leaves useful habitats at Mars. Clustering large habitat modules together highlights a trade between habitability (gravity-living) and safe-haven splitting (losing too large a percentage of habitable volume through "keystone" module loss). Our vehicle archetypes were configured to be able to accommodate clustered habitat modules as well as multiple landers, even in artificial gravity modes. Like artificial gravity, if large crew sizes become a requirement, mission designs can be found using the concept vocabulary developed by STCAEM to handle them.

Related to large crew sizes is the issue of simply heavy transfer habitat systems. If the

GCR dose problem during interplanetary transfers becomes such a physiological show-stopper that even fast-transfer propulsion technology cannot solve it, then massively shielded habitats may prove to be necessary for preserving crew health. For these heavy payloads as well as those caused by large crew sizes, cycling orbits may provide a practical answer. The technology issues associated with potentially very high aerobraking energies (for capturing at Mars off the cycler) then become paramount.

2.2.5.10 Vehicle Operations Derived Requirements

The vehicle archetypes were developed with thought given to the difficulties inherent in: (1) processing and testing their pieces on the ground; (2) launching them into Earth orbit; (3) assembling and testing them in LEO; (4) maintaining them en route; and (5) refurbishing them upon return. (Preliminary concept options for supporting these activities, and analysis of ground and processing flows, are presented in Section 8) However, at this early stage of SEI, it is vital that accommodation of those support activities not take precedence over vehicle performance in accomplishing the missions for which they were intended. Requirements for processing "nodes" and technologies must be driven primarily by what the transportation-optimized vehicles themselves require. In this regard, three key principles are important:

a. Mars-class vehicles, and even their aerobrakes, are simply too large to be packaged whole inside ETO rockets. In any case, re-usable transfer vehicles require refurbishment upon their return. It is not possible to avoid orbital processing through the "magnum" launch vehicle approach.

b. Because the vehicles must operate successfully for several years with no opportunity for resupply or external assistance if their missions are to be sufficiently safe for humans, sophisticated capability must be built into them for in-flight maintenance, diagnosis and repair. This includes a substantial ability to inspect, manipulate, disassemble, reconfigure and reassemble vehicle systems, some of which are quite large. It also includes a substantial cache of critical spare parts. The key architectural driver flexibility (robustness and resiliency) makes this essential. Given that requirement, the best place and manner in which to test that capability is in LEO, using it to assemble, check out and maintain the vehicles in "dry dock." By the very extremity of these deep-space, crew-carrying missions, very little could be done in Earth orbit that would not have to be feasible also in Mars orbit.

c. LEO is a particularly hazardous place to leave large, expensive or unique space systems, because of its manmade debris environment. Heavy shielding is feasible (and necessary) for a permanent facility like SSF, but cannot be afforded (and is not thought to be necessary at all) in transit to other planets. Thus it falls to the LEO facility, whatever it may be, to protect space transportation vehicles from the unique LEO environment.

Although a variety of debris-shielding schemes have been proposed, decoupling that service from other "dry dock" needs admits more innovative solutions. One is to have removable, re-usable, conformal bumpers which encase the critical systems of a transfer vehicle parked in Earth orbit. Rather than being part of a node structure, they become a part of the vehicle that is left behind, awaiting the next vehicle to be processed, or the departed one to return. Further vehicle integration work must begin to incorporate the very real constraints of what accommodations the ground and orbital capabilities of 2015 will realistically be able to provide in support of exploration missions. Designing for machine-mediated operations, based even on current state-of-the-art automation and robotics, will enhance our concepts' credibility.

2.3 Recommendations for Future Transportation Concept Efforts

Three key areas relating to concepts and evolution require more work:

a. Understanding of how to support advanced SEI missions, both on the ground and in space, needs to be improved. Much opportunity exists for innovative approaches to these costly challenges.

b. In support of that improved understanding, the vehicle concepts require further detail work. More detailed design integration of certain subsystems, notably main propulsion plumbing, attitude control propulsion, bus structure and communications equipment, is needed for the principal concepts. This is necessary to develop technically valid concepts for space assembly, checkout, and launch processing of space transfer vehicles. Enhanced design-for-maintenance is important now that we have a first-pass at integration and performance issues for these archetypes.

c. In support of vehicle integration, it is vital to understand in more depth the proper selection of subsystems for, and configuration of, the habitable elements: LEV/MEV crew cabs, LTV crew habs, MTV hab system, and ancillary items like airlocks and crew return vehicles (MCRVs). Particularly in the case of the MTV habitat, it is time to incorporate quick-turnaround radiation analysis into the iterative concept development cycle.

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3. MISSION ANALYSIS

For the current study our mission analysis investigations are based on the work performed in support of the NASA 90-Day Study for the Lunar /Mars Initiative (Reference 5). The NASA study established the baseline goals, mission vehicle, trajectory set and levied requirements from which a mission set could be derived. This baseline mission set became the starting point for expanding and adapting the in-space transportation system requirements and analyses to vehicles, scenarios and conditions other than those set by the 90 Day-Study. This included optimizations of trajectories for low thrust and high thrust main mission transit vehicles (lunar and Mars) and conditions at capture (vehicle position and GN&C), the Mars excursion vehicles during descent, landing and ascent, the main vehicle and Earth return vehicle arrival plus aerobrake characteristics and vehicle responses.

3.1 Lunar

Part of our support to the 90-Day Study was to conduct a preliminary evaluation of the transfer and trajectory requirements of a lunar vehicle from the Space Station Freedom vicinity to a lunar orbit plane. Because the orbit plane of the Space Station and the orbit plane of the Moon are at different inclinations and rotation occurs at different rates, the transfer plane angle varies in time. A plot of the Moon's out of plane declination in time and the out of plane angle between the lunar and Space Station orbit planes in time will yield a graphic representation of the windows of opportunity where the traces intersect. From this data the required delta V to make the transfer can be calculated for the transit time in days at each angle between the transfer and lunar orbit plane. From the graphic data it can be seen that the opportunities for transfer occur five times every 52 days.Transfers requiring a plane change at the Moon incur a modest cost in delta V, Figures 3-1 and 3-2 (Reference 11).

A preliminary operational sequence was generated for a single mission with a generic abort scenario, Figure 3-3. This assumes that the vehicle leaves from Space Station orbit. The greatest cost in time of transit to the Moon is the wait in orbit for the appropriate lighting conditions to land. With a landing pad with beacons this time may be reduced (advanced missions).

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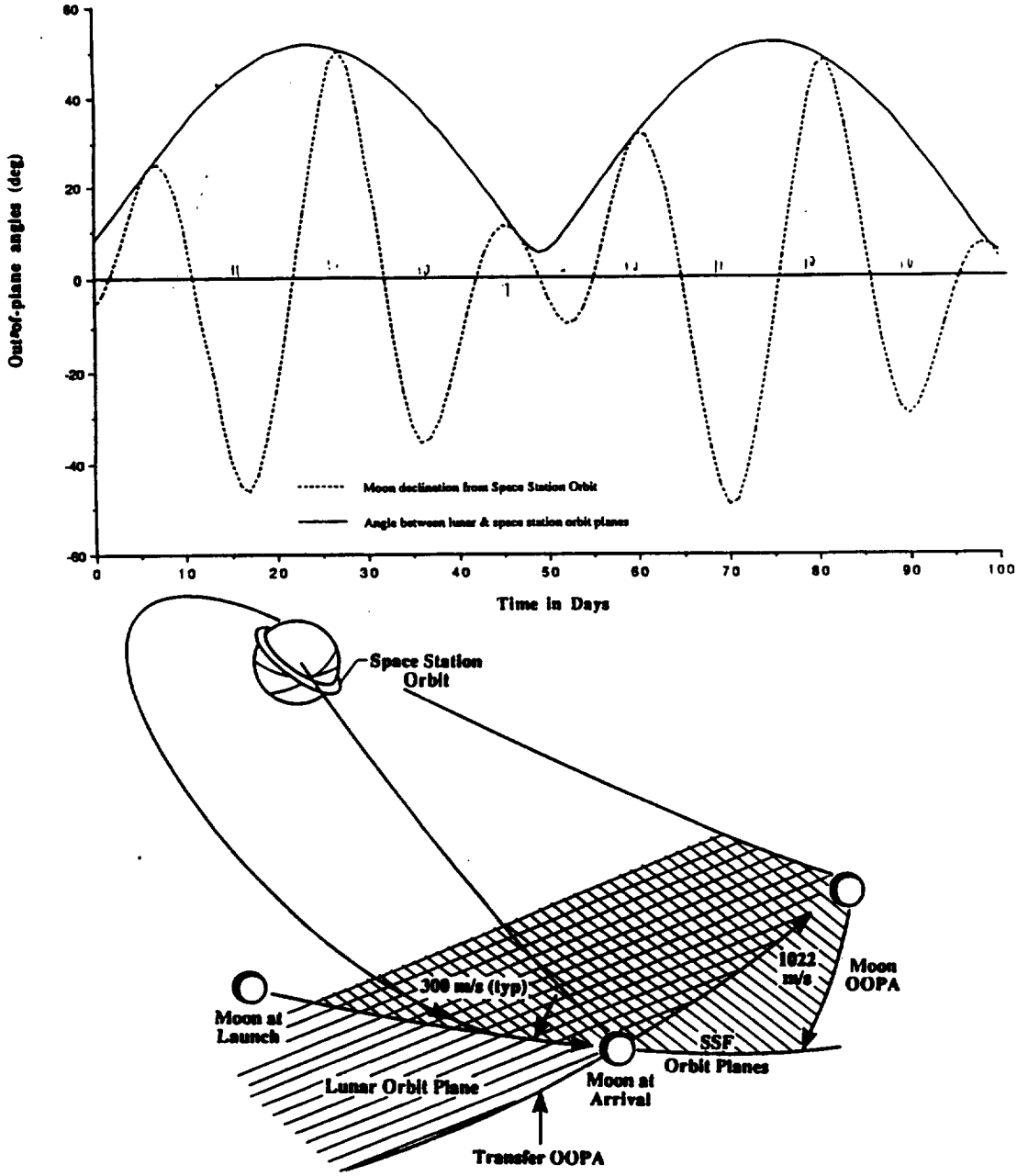
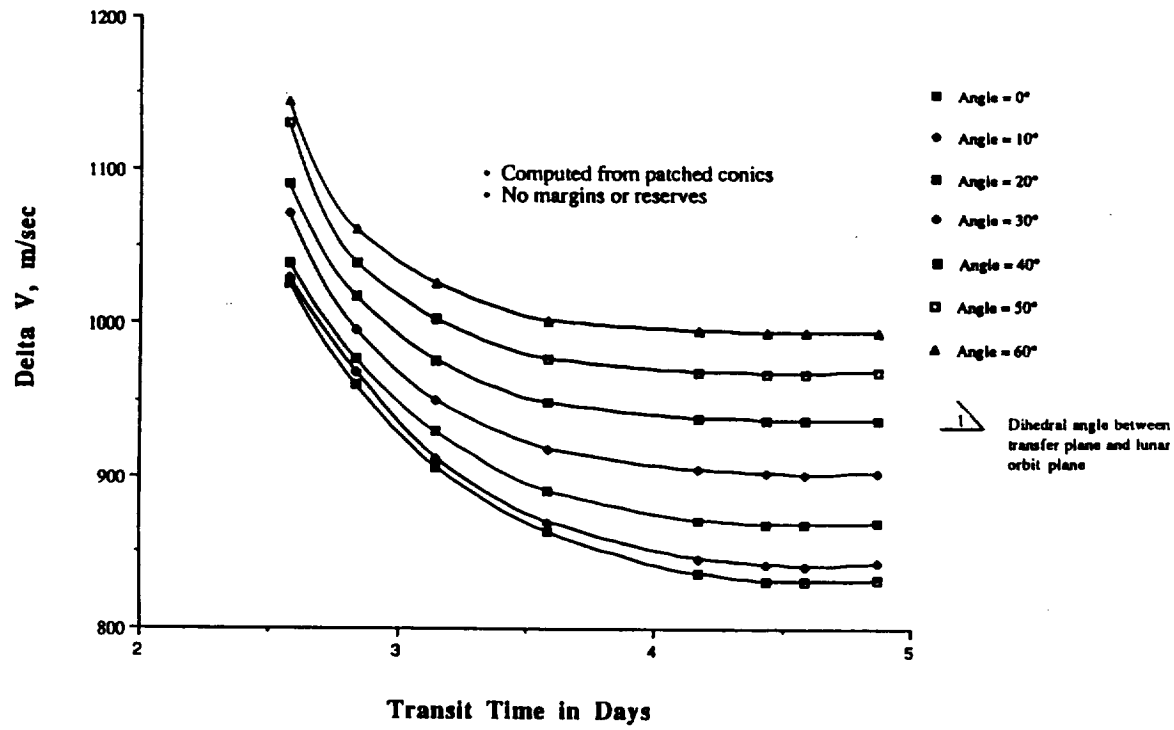


LUNAR ANALYSIS, (TOP) LUNAR ORBIT IDEAL DELTA V, (BOTTOM) EARTH-SSF MOON TRANSFER PLANES

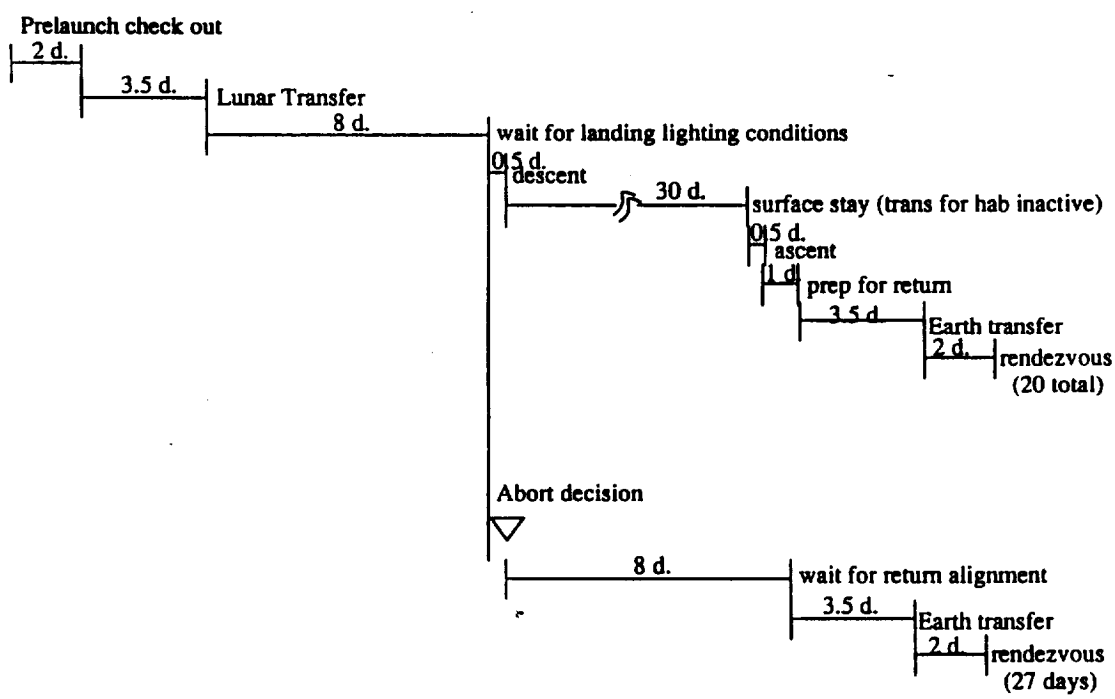
FIGURE 3-1

For a normal mission the entire transit time, outbound and return discounting staytime, is 20 days. In the case of an abort an equally long time is spent in lunar orbit awaiting the alignment of the transfer plane for return, making the total transit time of an abort 27 days. Out of plane phasing for return for either condition will be done in LLO. Phasing into the Space Station orbit will be done in a midcourse correction and completed on aerocapture, Figures 3-4.



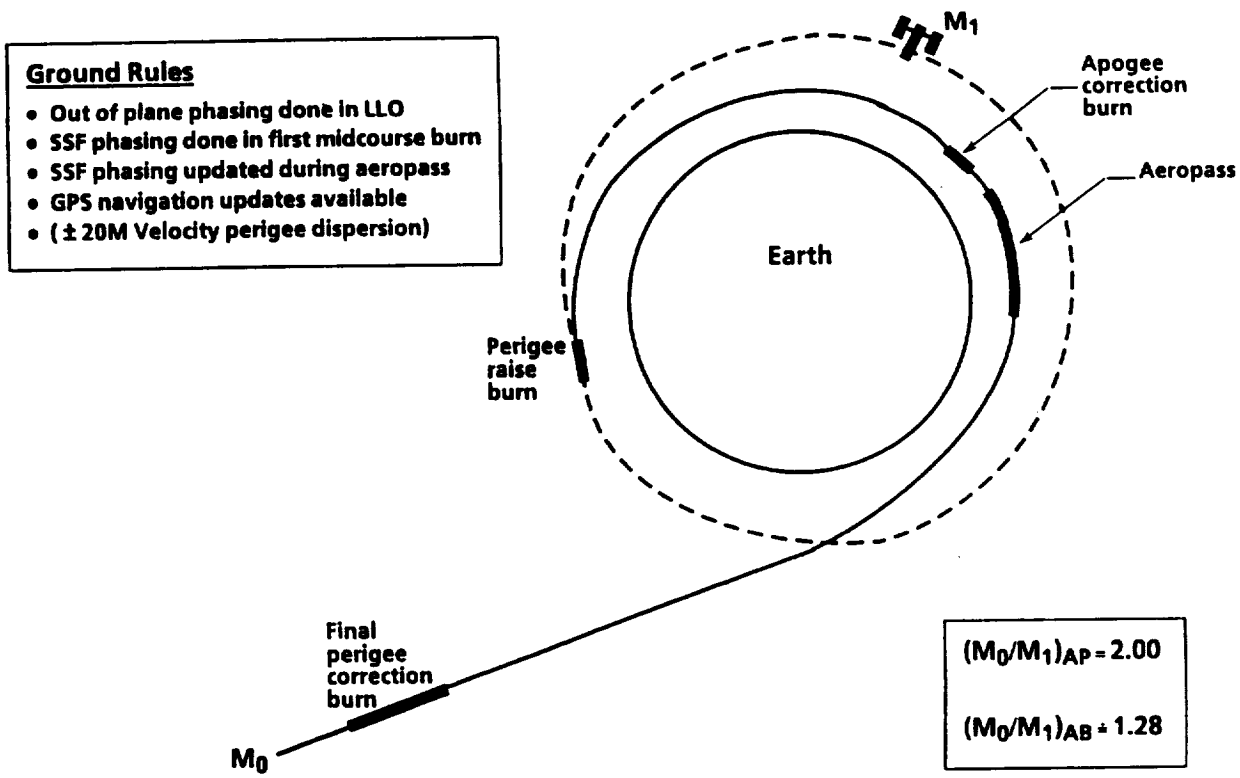
LUNAR ORBIT INSERTION/DEPARTURE DELTA VS

FIGURE 3-2



LUNAR TRANSFER HAB TIMELINES (ESTIMATE)

FIGURE 3-3



AEROBRAKING MANEUVER CHARACTERISTICS
FIGURE 3-4

Returning to Earth, the lunar vehicle will do an aerocapture maneuver consisting of an aeropass through the atmosphere, where final Space Station orbit phasing will occur, an apogee correction burn, and a perigee rise burn to bring the vehicle into co-orbit with the Space Station (Reference 11). The calculations of aeroheating on an L/D 0.5 aerobrake capturing at Earth from the Moon at 10.3 km/sec and diving to a minimum altitude of 78.75 km indicate that the TPS material required will be within current or near term technological capabilities. In addition to the Lunar cryogenic/aerobraked vehicle, NTR, SEP and NEP advanced propulsion alternatives were examined, and parametrics run (Reference 11).

A cursory analysis was made of delivering lunar oxygen to LEO from the Moon. This was done to support the top level architectures analyses. A detailed description of the lunar operations and its interplay with Mars operations are given in Section 1. Lunar operations are near term considerations that must be further evaluated in relation to Mars support.

3.2 Mars

The Mars baseline vehicle from the 90-Day Study was a cryogenic propellant (liquid hydrogen/liquid oxygen) with aerocapture for the main and excursion vehicle sections and aeroentry capability for the excursion vehicle at Mars, returning in an "Apollo-style" reentry capsule at Earth. In addition, other vehicle types were developed and explored for the main Mars transportation system. Several feasible alternate systems in addition to the baseline were selected for further study: cryogenic all propulsive (no aerocapture at Mars), nuclear thermal rocket (NTR), nuclear electric propulsion (NEP), solar electric propulsion (SEP) and gas core reactor (GCR, nuclear).

Even the best estimation of development time for the GCR places its readiness at the far end of the 2010 to 2025 time frame for a first use involving a man-rated system. Therefore, this was considered a follow-on type vehicle that might be used in long range planing and was not used in the scenarios and architectures we developed for the early part of the 21st century.

The remaining main vehicles fell into two categories, low or high thrust, based on the type of propulsion system used and their transfer characteristics. Because of the gravity losses associated with low thrust (electric propulsion) devices, the two are analyzed differently.

3.2.1 High Thrust Missions

High thrust missions entail the use of propulsive systems that work on short impulse, high momentum transfers from the propellant. This class includes the cryogenic/aerobrake, cryogenic all-propulsive and nuclear thermal rocket vehicles chosen for alternate systems study. A series of mission opportunities, both conjunction and opposition class, were generated between the 2010 and 2025 dates, concentrating around the dates 2015-2017 close to the NASA baseline. A table of these opportunities containing the Earth departure-Mars arrival, Mars departure-Earth arrival dates, orbit departure and arrival conditions and Venus swingby information is shown in Figure 3-5. Additional details may be found in References 7-11.

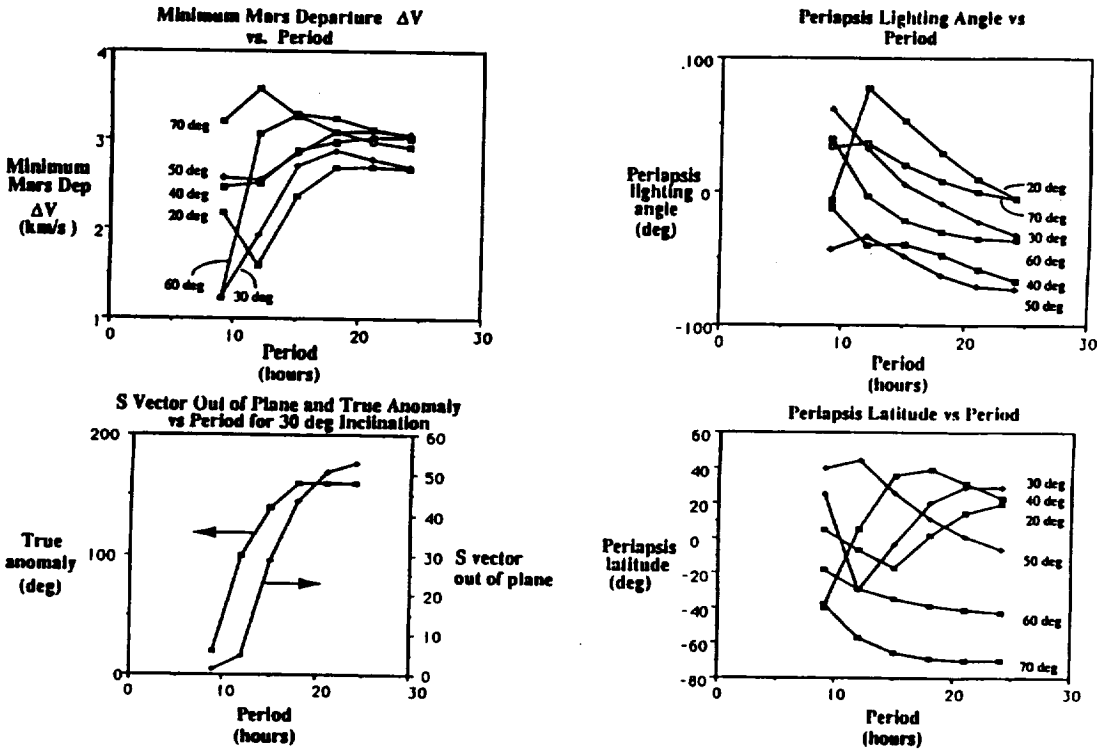
Opportunity (optimized oppositions)		Earth Dep C3	Mars Arr. Vhp	Mars Departure C3	Mars Departure Δ V	Earth Arrival Vhp	Earth Arrival C3	Mars Orbit Inc. (Deg)	Earth launch DLA	Mars Arr. LVI
2010 Opposition	12/01/10 -10/26/11, 11/25/11- 8/31/12	28.69	4.93	16.1	2.32	6.99	48.9	30	4.19	-28.63
2010 Conjunction	10/26/9 -10/31/10, 8/27/11 - 7/15/12	11.14	3.26	7.0	2.73	3.80	14.4	30	32.82	9.48
2013 Opposition	11/22/13 - 8/6/14, 10/5/14 - 8/14/15	13.08	4.10	37.7	3.54	4.53	20.5	30	20.42	25.32
2013 Conjunction	12/3/13 - 9/23/14, 9/28/15 - 9/6/16	9.58	3.15	6.9	2.61	4.47	20.0	45	23.73	31.13
2015 Level II Reference	5/23/15 - 4/22/16, 5/22/16 - 12/8/16	20.19	7.02	30.0	4.70	8.77	77.91	30	55.92	14.51
2015 Level II Alternate	10/15/15 - 7/16/16, 8/15/16 - 5/17/17	48.38	4.79	33.3	3.58	3.94	15.5	30	0.18	15.3
2015 Conjunction	12/24/13 - 11/17/14, 12/14/15 -10/8/16	8.89	4.22	5.4	2.37	5.52	30.5	35	18.45	32.93
2015 L II Ref. + 50 day	5/23/15 - 4/22/16, 5/22/16 - 1/27/17	14.19	6.93	18.0	2.37	9.47	89.7	30	55.92	15.22
2016 Boeing Nominal	2/25/16 - 7/31/16, 8/31/16 - 5/5/17	10.34	6.82	39.7	4.37	7.14	51.0	30	-35.94	-1.69
2018 Opposition	3/27/17 - 3/10/18, 4/24/18 - 12/18/18	19.71	5.96	10.9	2.58	5.04	25.4	30	17.97	22.3
2018 Conjunction	5/12/18 -11/28/18, 5/31/20 - 11/27/20	7.86	2.97	7.9	3.41	4.02	16.2	45	-37.94	-7.81
2020 Opposition	6/4/20 - 12/11/20, 1/10/21 - 1/28/22	24.40	3.89	27.8	3.77	4.28	18.3	20	15.1	-9.69
2020 Conjunction	7/20/20 - 1/16/21, 8/09/22 - 1/16/23	13.40	3.13	18.8	3.92	6.67	44.5	30	18.65	-3.40
2022 Opposition	11/11/21 - 9/17/22, 10/17/22 - 6/5/23	16.31	5.31	43.2	4.46	6.00	36.0	20.	-60.02	-4.04
2023 Conjunction	9/8/22 - 4/16/23, 7/9/24 - 5/5/25	19.03	3.18	12.3	2.66	2.86	8.81	30	50.73	23.71
2024 Opposition	9/20/23 - 7/4/24, 8/4/24 - 5/30/25	27.91	6.46	9.0	1.61	3.08	9.49	30	-19.58	-6.53
2025 Conjunction	10/17/24 - 6/24/25, 8/11/26 - 5/5/27	19.68	3.00	8.3	2.60	2.60	6.76	35	55.88	34.75

Level II Reference : ΔV Earth Departure = 4281 m/sec ΔV Earth Arrival = 6278 m/sec (at LEO)
Data From MASE ΔV Mars Arrival = 3949 m/sec ΔV Mars Departure = 3400 m/sec

MARS TRAJECTORIES FOR MINIMUM MARS DEPARTURE DELTA V
FIGURE 3-5

In performing the mission analysis, an all inclusive front-to-back mission analysis viewpoint must be considered. The general "patched conic" analysis technique used for each mission category included: (1) analyses performed over the mission opportunity from 2010 through 2025; (2) energy/velocity contours generated by the Boeing PLANET code to find the global minimum delta velocity launch date and transfer time (Reference 20); (3) PLANET data analyzed for Mars departure transfers based on optimal departure true anomaly for minimizing departure delta velocity; this optimization yields more realistic trajectory and velocity data; and (4) alternative optimizing schemes invoked to improve upon the results.

Additional constraints were imposed by physical or practical considerations in the design of missions and included items such as the length of transit time with its resultant exposure to cosmic radiation, the hyperbolic excess velocity which indicates how "hot" the entry is, daylight landing (periapsis lighting angle) for landing hazard avoidance, parking orbit inclination and period selection for minimum delta V, planet accessibility and departure constraints, and the total delta V budget, that culminates in the "price" in propellant (therefore IMLEO) that must be paid, Figure 3-6.



2010 CONJUNCTION ORBITAL PARAMETRICS
FIGURE 3-6

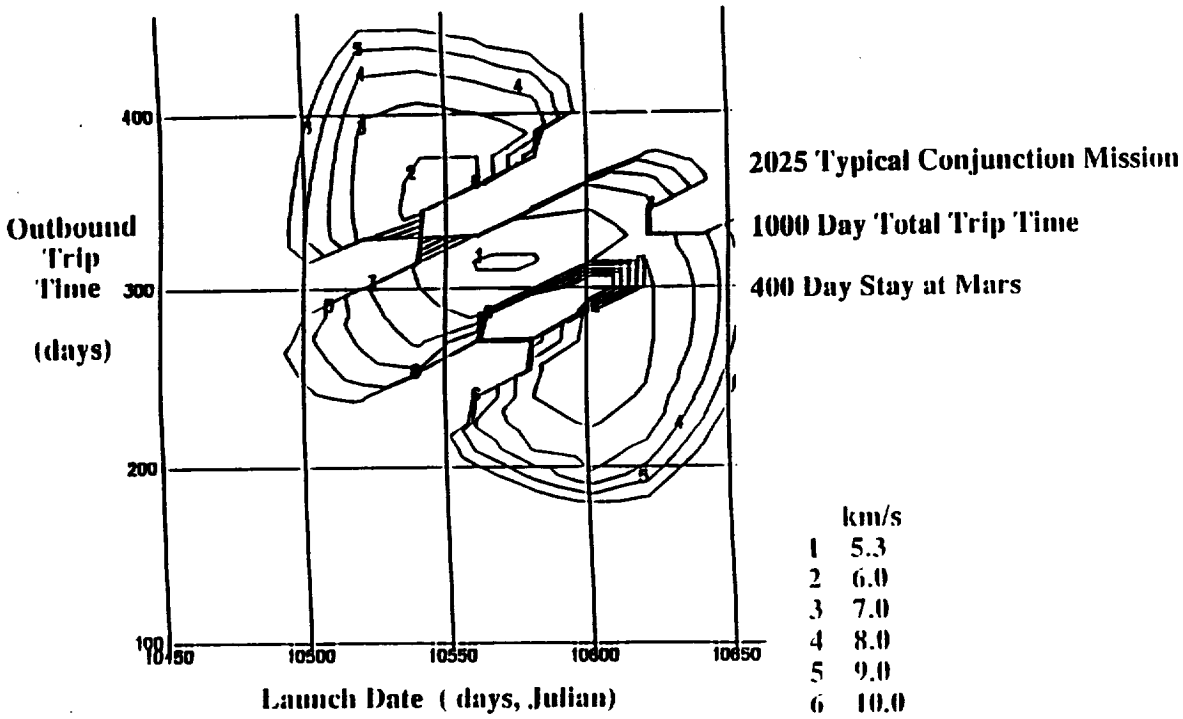
3.2.1.1 Conjunction Missions

Some advantages of conjunction missions, in general, are as follows: (1) Venus swingby complexity not needed; (2) provision of longer launch windows; (3) elliptical parking orbit can be optimized; and (4) more hardware reuse possible.

3.2.1.1.1 Low Energy Conjunction

Various types of low energy missions were analyzed. Contours and trajectory runs were found for non-optimized mission profiles and for those that were optimized for minimum total delta velocity.

The contour for a typical low energy conjunction mission in the "hard" year 2025 is shown in Figure 3-7. From this graph the total delta V may be selected for a given launch date and a given outbound trip time (stay time and total trip time are held constant). It can be seen that the launch dates for a given total trip delta V are very broad (a "forgiving" system that allows for several changes to meet the launch criteria) and overall low total mission delta Vs.



CONJUNCTION TRANSFER NON-OPTIMAL ROUND TRIP CONTOUR
FIGURE 3-7

Minimum energy launch dates and trip times were found from similar PLANET-generated contours for non-optimized Mars stay-times over the mission opportunity dates 2010 through 2025. From the non-optimal stay-time data we find that: (1) low total delta velocity for these missions translates to low propellant needs; (2) long stay time at Mars to support science missions; (3) low energy conjunction missions are insensitive to launch year; (4) mission time is approximately 1000 days; and (5) four-year personnel stay at Mars is required for continuous base occupancy.

Optimized stay-time for low energy conjunction missions were found for 1000 day total trip time, with mission years of 2018, 2020, and 2025. The results of optimizing stay-time are as follows:

(days)	(days)	(km/s)
Mission	**Staytime**	**Delta Velocity**
2018	500	5.5
2020	525	5.4
2025	300	5.0

Note that the 2025 mission is lowest in energy, with 2020 falling between 2025 and 2018. The same five results statements, as given above, apply to the optimized conjunction missions.

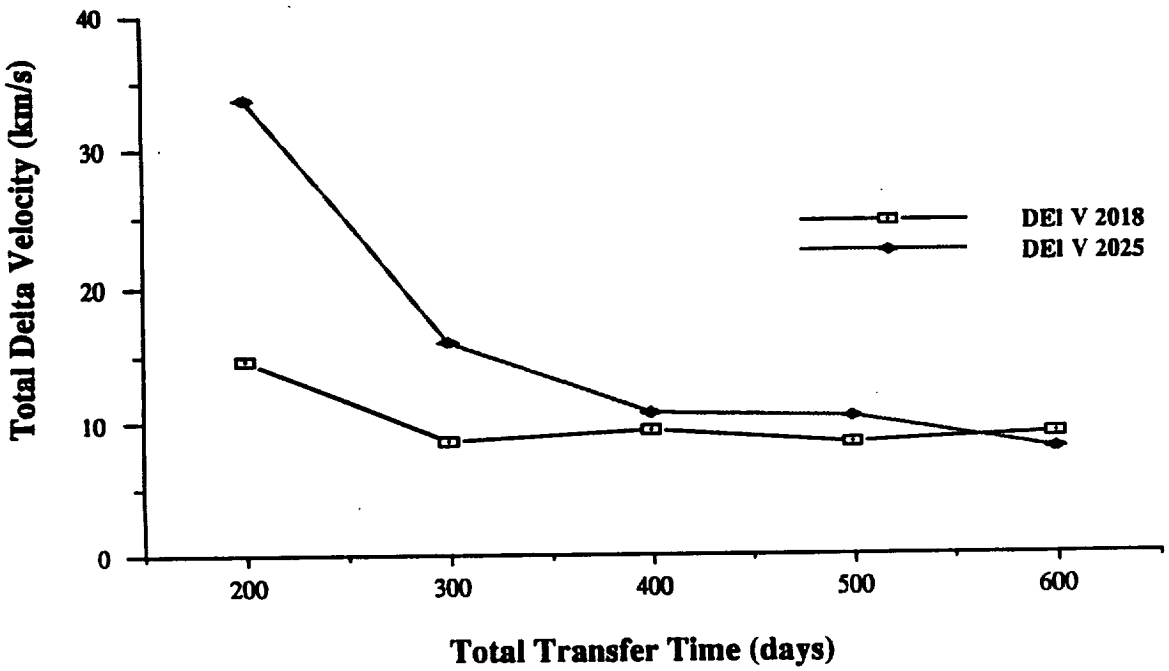
3.2.1.1.2 Fast Transfer Conjunctions

Fast-transfer conjunction studies were carried out for non-optimal and optimal missions. An optimal fast transfer conjunction mission means that for a chosen total (outbound and inbound) transfer time of 200, 300, 400, 500, and 600 days, the optimal stay and total trip times were found, which lead to minimum total energy mission profiles for those transfer times, Figure 3-8. Non-optimal means that for the transfer times given above, optimal stay times and total trip times for minimum energy missions were not examined. From the contour plot for the fast transfer in year 2025, Figure 3-9, it can be seen that the total trip delta Vs are higher and that the launch dates are more constrained than for the low energy missions.

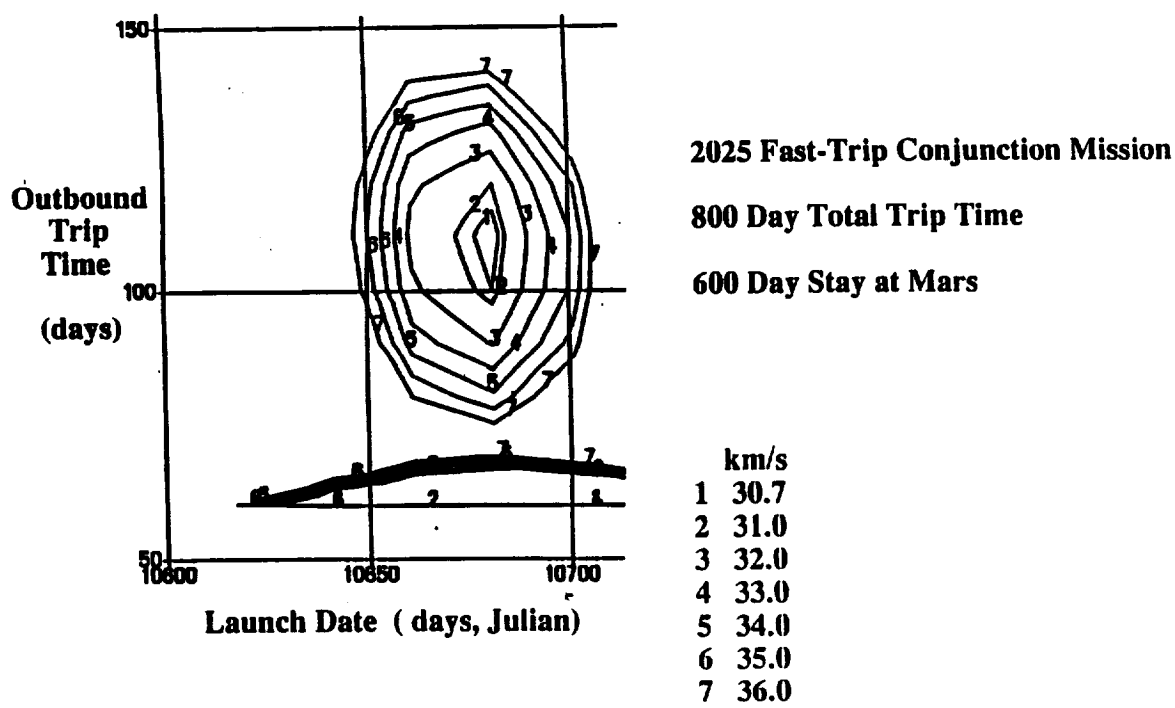
For the fast transfer conjunction missions analysis, the results are as follows: (1) reduced exposure of astronaut to hazardous space radiation (i.e. relatively short trip times); (2) lower IMLEO cost to significantly reduce trip time, Figure 3-10; and (3) IMLEO sensitivity to launch year.

3.2.1.2 Opposition Missions

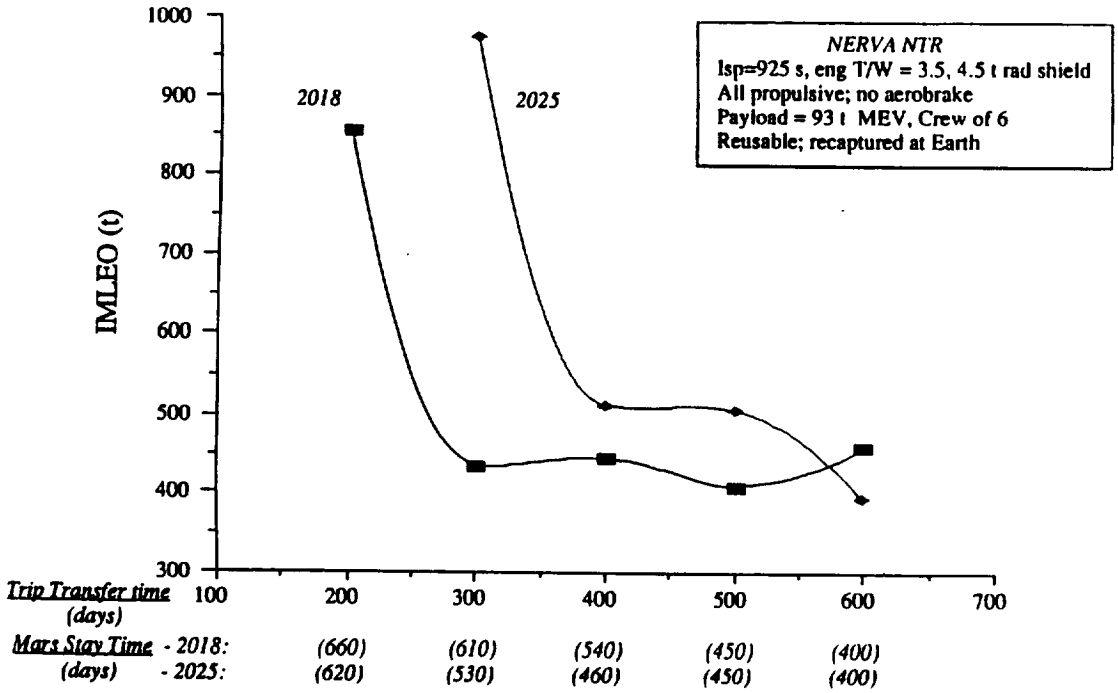
This class of high thrust mission reduces the overall trip time over low energy conjunction trips at the cost of increased delta V. The opportunities occur when Earth and Mars are positioned as shown in Figure 3-11, and the transit conditions occur across the potential velocity pattern of both planets.



CONJUNCTION FAST TRANSFER OPTIMIZATION
FIGURE 3-8



FAST TRANSFER ROUND TRIP CONTOUR TOTAL DELTA V PLOT
FIGURE 3-9



FAST TRANSFER IMLEO
FIGURE 3-10

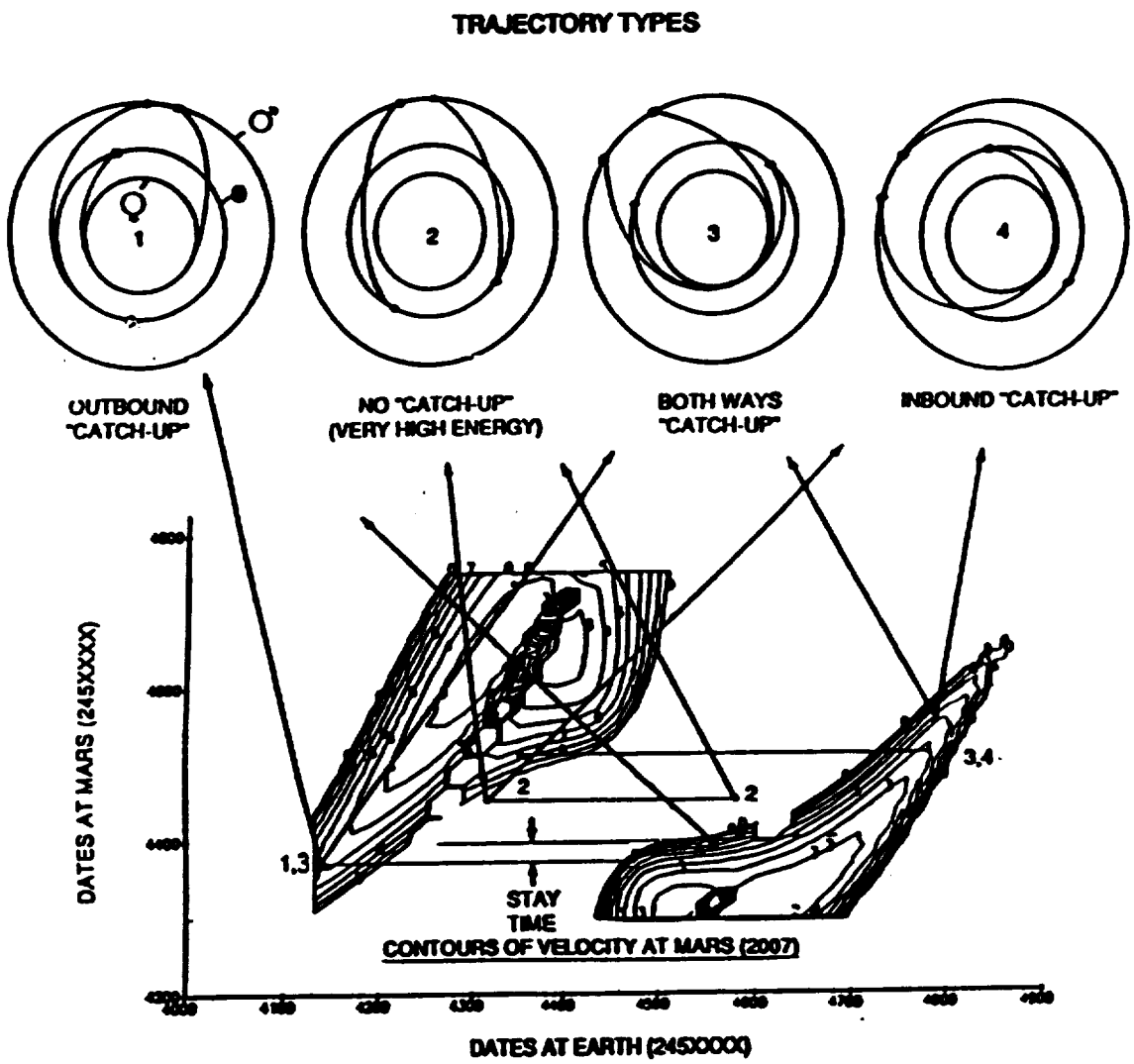
3.2.1.2.1 Venus Swingbys

Analyses were performed to find minimum energy Venus swingby opportunities for the period 2010 through 2025. The total delta V for these years is shown in Figure 3-12. The results of these analyses are as follows: (1) Reduced total mission duration (500 days); (2) allows two year personnel stay for continuous base occupancy; (3) short stay times (30 days); (4) high IMLEO; (5) short launch windows; and (6) transfer vehicle reuse on the next opportunity.

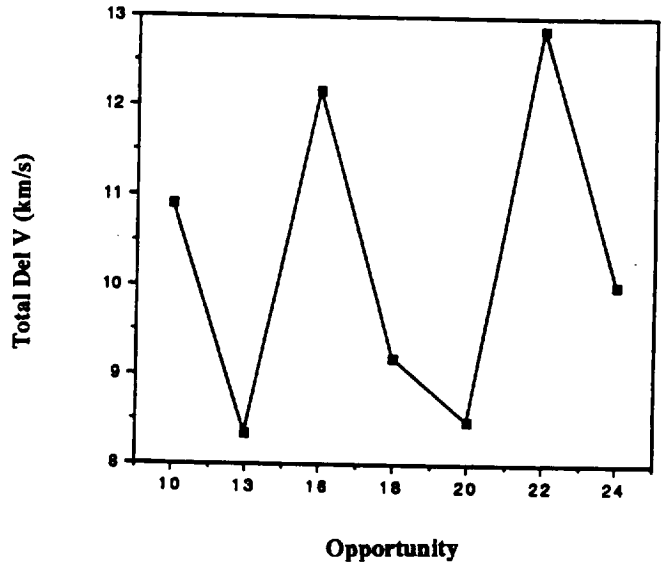
3.2.1.2.2 Fast Transfer Opposition

Fast transfer opposition analyses were carried out for optimal and non-optimal cases. In the non-optimal analysis, the fast transfers were not optimized for minimum energy transfers by deep space burn maneuvers, Figure 3-13. For the optimal fast transfer opposition, an optimal date and location for a deep space burn was found, minimizing total mission energy requirements. Fast transfer oppositions were studied for total transfer times of 350, 400, 450, and 500 days, and do not assume Venus swingby's to further

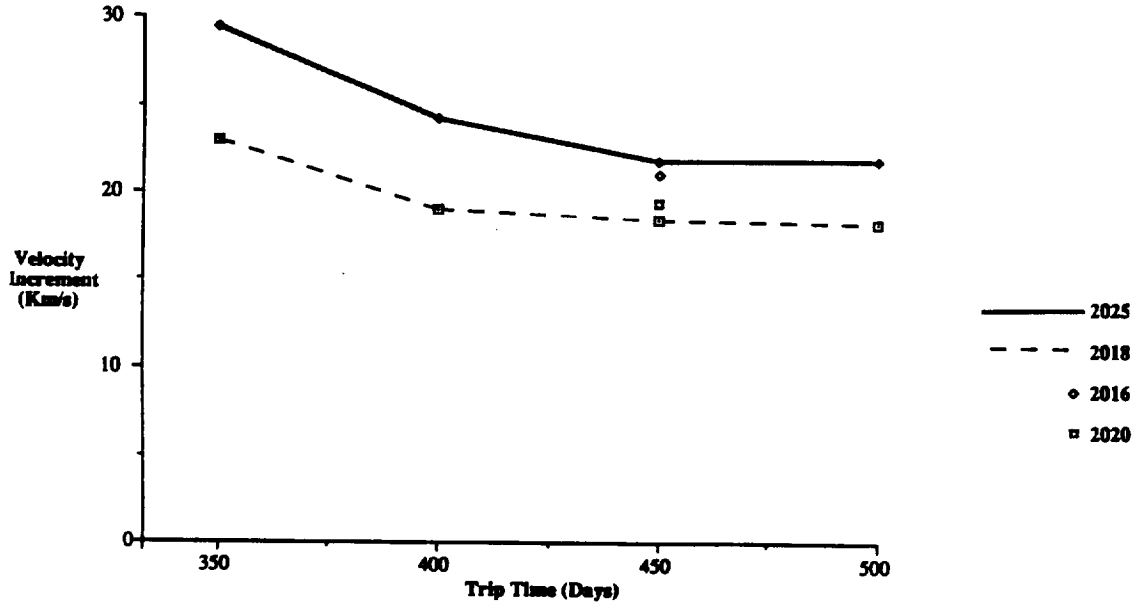
minimize energy. Planetocentric delta V portions of these transfers are shown in Figures 3-14 and 3-15 for Mars and Earth arrival and departure in the "easy" year of 2018 and the "hard" year of 2025. The total delta V cost for these years in the optimal case is shown in Figure 3-16. The results of the fast transfer analysis for optimal and a non-optimal trajectories show short mission duration (350 - 450 days), and indicate a sensitivity to launch opportunity.



GENERAL OPPOSITION CONTOUR PLOT
FIGURE 3-11



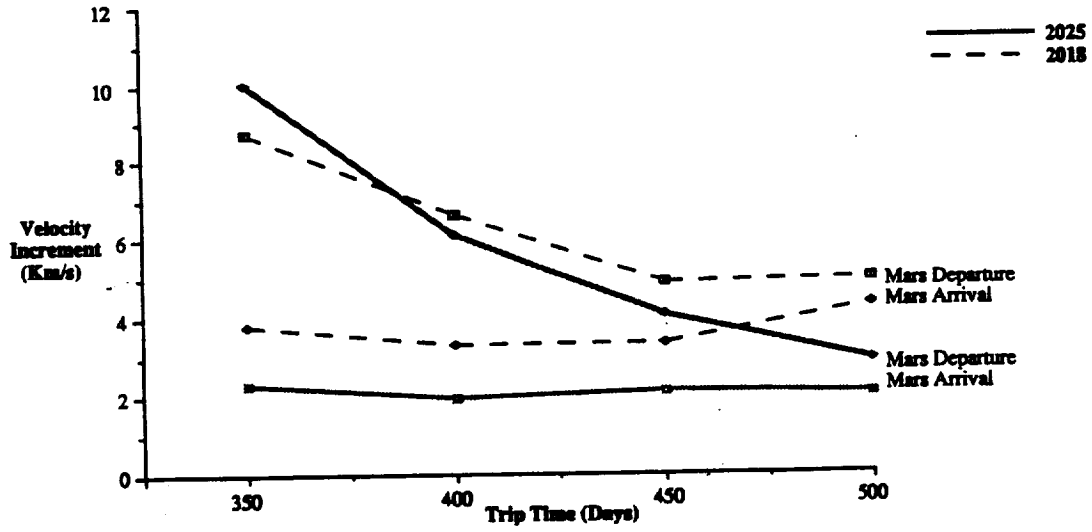
TOTAL TRIP DELTA V FOR OPPOSITION SWINGBYS
FIGURE 3-12



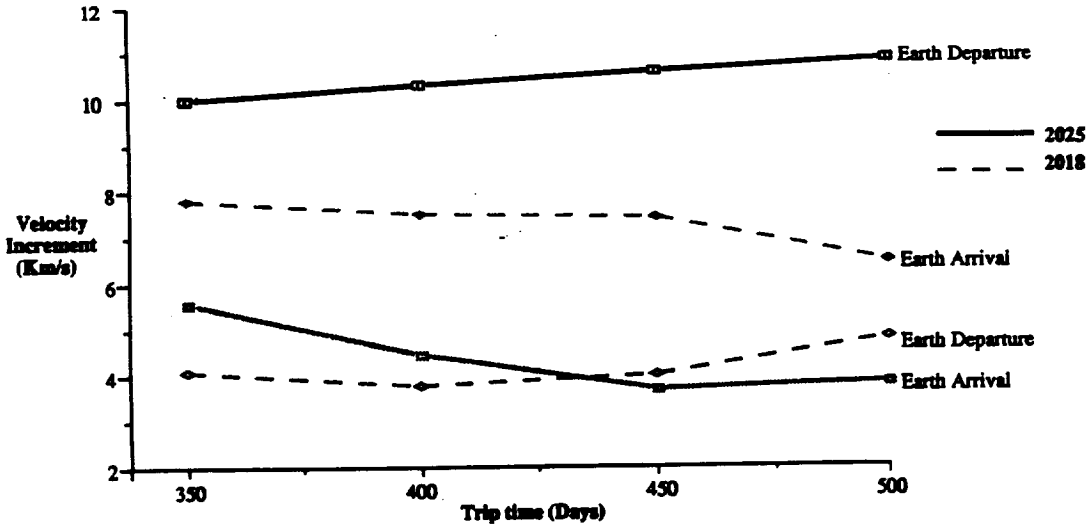
OPPOSITION TOTAL DELTA V PROFILES, ALL PROPULSIVE
FIGURE 3-13

3.2.1.2.3 Split Sprint Missions

This class of mission has been used in architectures prior to this study. In reviewing them, we determined the risk involved in sending the return supplies, food and propellant, in a separately aerocaptured vehicle that did not allow capture conformation before the manned mission left.

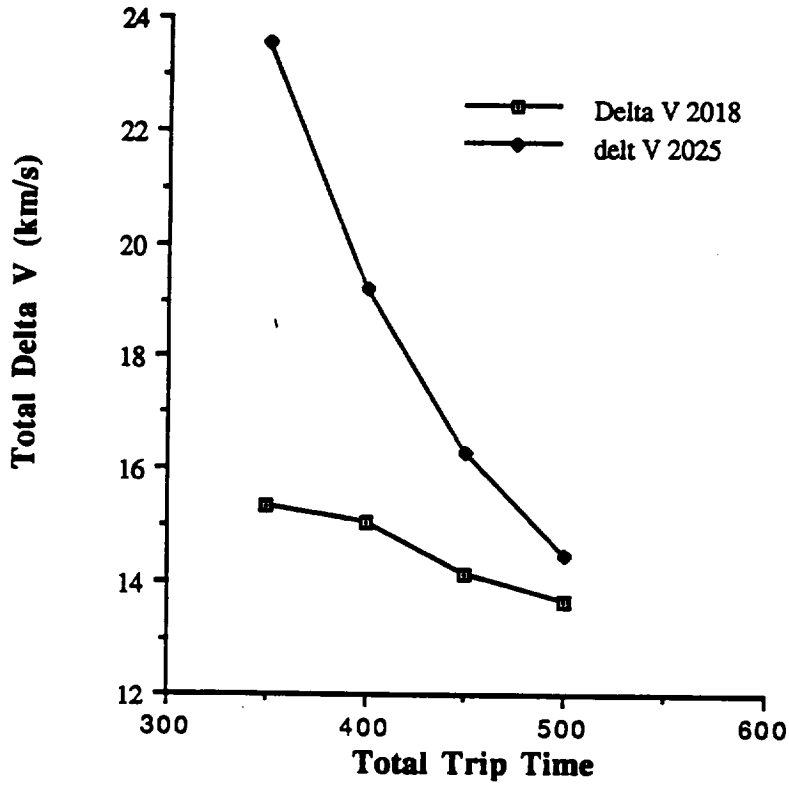


MARS DEPARTURE/MARS ARRIVAL OPPOSITION DELTA V PROFILES
FIGURE 3-14



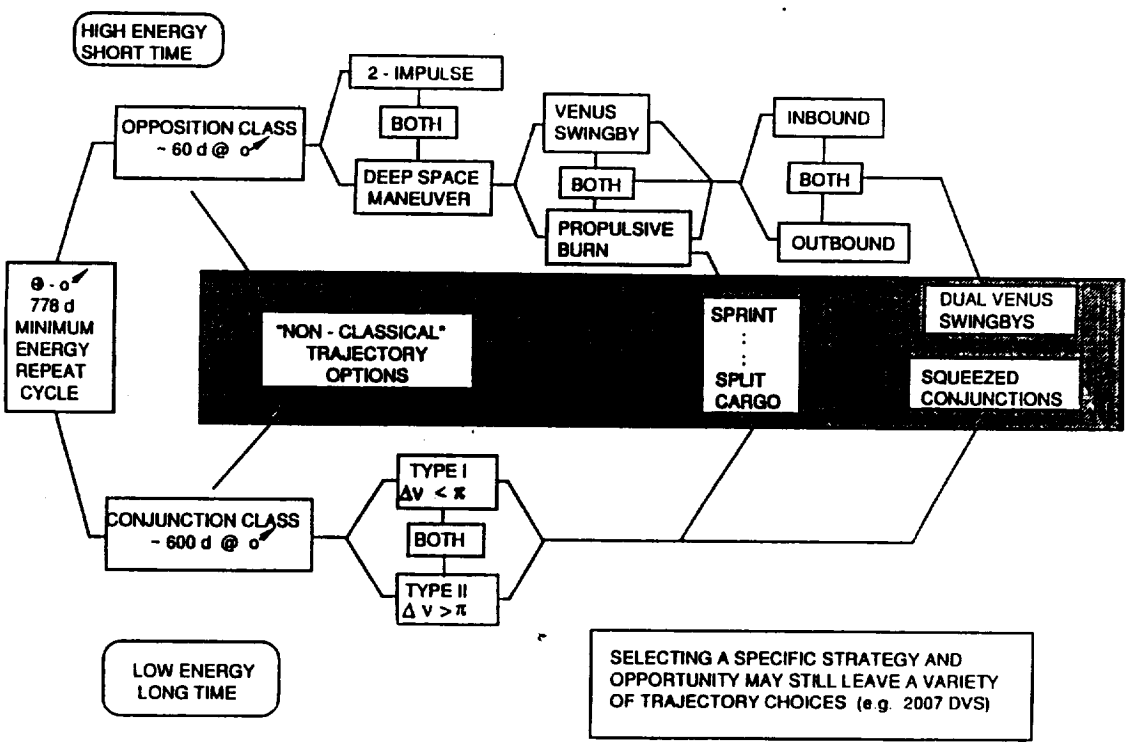
EARTH DEPARTURE/EARTH ARRIVAL OPPOSITION DELTA V PROFILES
FIGURE 3-15

We determined that the risk was too great and violated the ground rules for flyby abort capability. We did not actively pursue this type of mission in our architecture analysis. It is in a group of non-classical mission analysis solutions illustrated in Figure 3-17.



TOTAL DELTA VS FOR FAST TRANSFER OPPOSITION MISSIONS

FIGURE 3-16



MARS TRAJECTORY OPTIONS

FIGURE 3-17

3.2.1.3 Periapsis Lighting and Periapsis Latitude

Landing site accessibility studies required landing analyses to include periapsis latitude calculations. Periapsis latitude is the latitude on the planet surface projected under the orbit periapsis. It is a critical position as most descents to the surface begin at this point. Periapsis lighting (lighting conditions from the sun's position over the periapsis latitude) is important from the standpoint that landing will require daylight for hazardous-terrain avoidance maneuvers. While this may not be as critical on subsequent flights that return to a previously explored site that may have landing beacons, on a first flight to any site the crew must be able to evaluate the landing terrain and make the last minute command decision to actually set the vehicle down (perform a retargeting maneuver, like Armstrong's landing of Apollo 11).

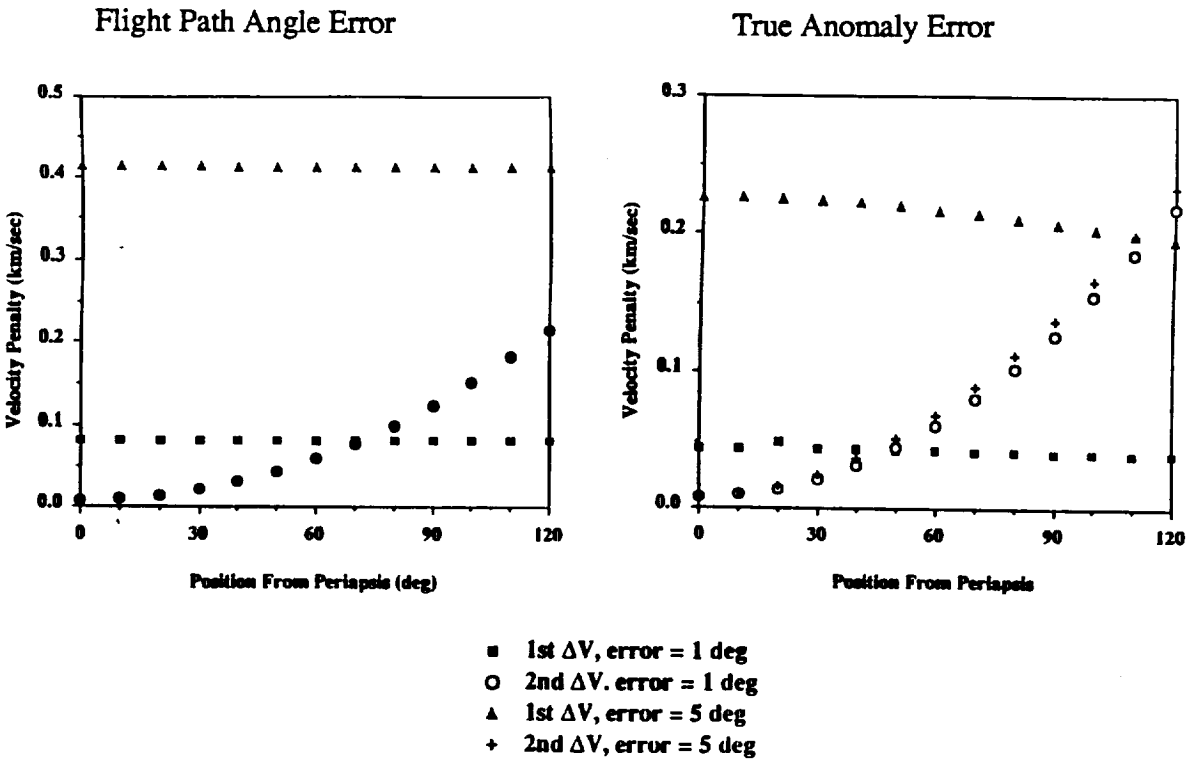
3.2.1.4 Rendezvous Error Analysis

In order to determine the change of velocity penalties associated with state vector errors after aerocapture, errors in both velocity vector (flight path angle, crosspath angle and magnitude) and position vector (true anomaly and plane) were examined.

The initial assumptions made were: (1) exit aerocapture close to periapsis, (2) target orbit conditions known, (3) error magnitudes known. It was also assumed that the errors be corrected in a two-burn sequence. The first burn (first delta V) would occur just after aerocapture exit and as close to periapsis as possible. This burn would align the apsides and would correct apoapsis height. The second burn (second delta V) would occur at apoapsis and would correct the inclination and the periapsis height. The analysis was performed using an in-house Boeing code called "MISSION" that has a subroutine that calculates the change of velocity penalties.

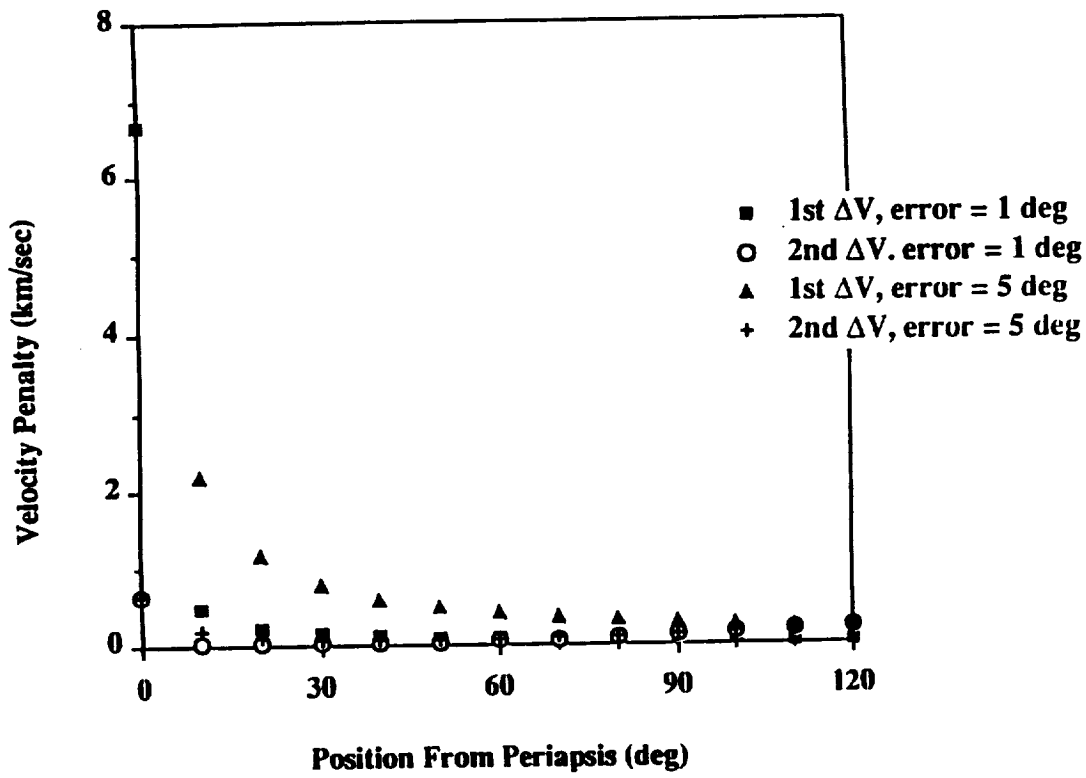
Four angular errors were tested at values of 1 degree, 5 degrees, 15 degrees, 25 degrees and 40 degrees. The velocity magnitude was evaluated for errors of 0.3, 0.75, 0.95, 2.0 and 3.0 (ratio of actual velocity to desired velocity). After all the errors were tested certain trends were observed. The magnitude of the first delta V is essentially independent of the location of the first burn. As the location of the burn moves further away from periapsis, the delta V penalty remains constant. The location of the first burn has a great effect on the magnitude of the second delta V. As the location of the first burn moved away from periapsis the magnitude of the second delta V penalty increased. The total delta V penalty

increases as the magnitude of the error increases. When the errors are associated with the velocity state vector a change in the error magnitude only affects the magnitude of the first delta V penalty; the magnitude of the second delta V penalty remains unchanged. When the error is associated with the position state vector a change in the error magnitude has an affect on both delta V penalties. In this case the error change has a greater effect on the first delta V penalty than on the second delta V penalty. The trends of these effects are shown in Figure 3-18 for 1 degree and 5 degree errors.



EFFECT OF BURN LOCATION ON ORBIT TRIM DELTA V
FIGURE 3-18

From this analysis it is observed that: (1) the first burn should be performed as close to periapsis as possible to incur the minimum delta V penalty on the second burn, and (2) the error should be as small as possible to reduce the total delta V penalty. The results and conclusions drawn from this case are not optimum. There are probably other burn sequences (these include more burns, different locations, etc.) that could produce smaller penalties.



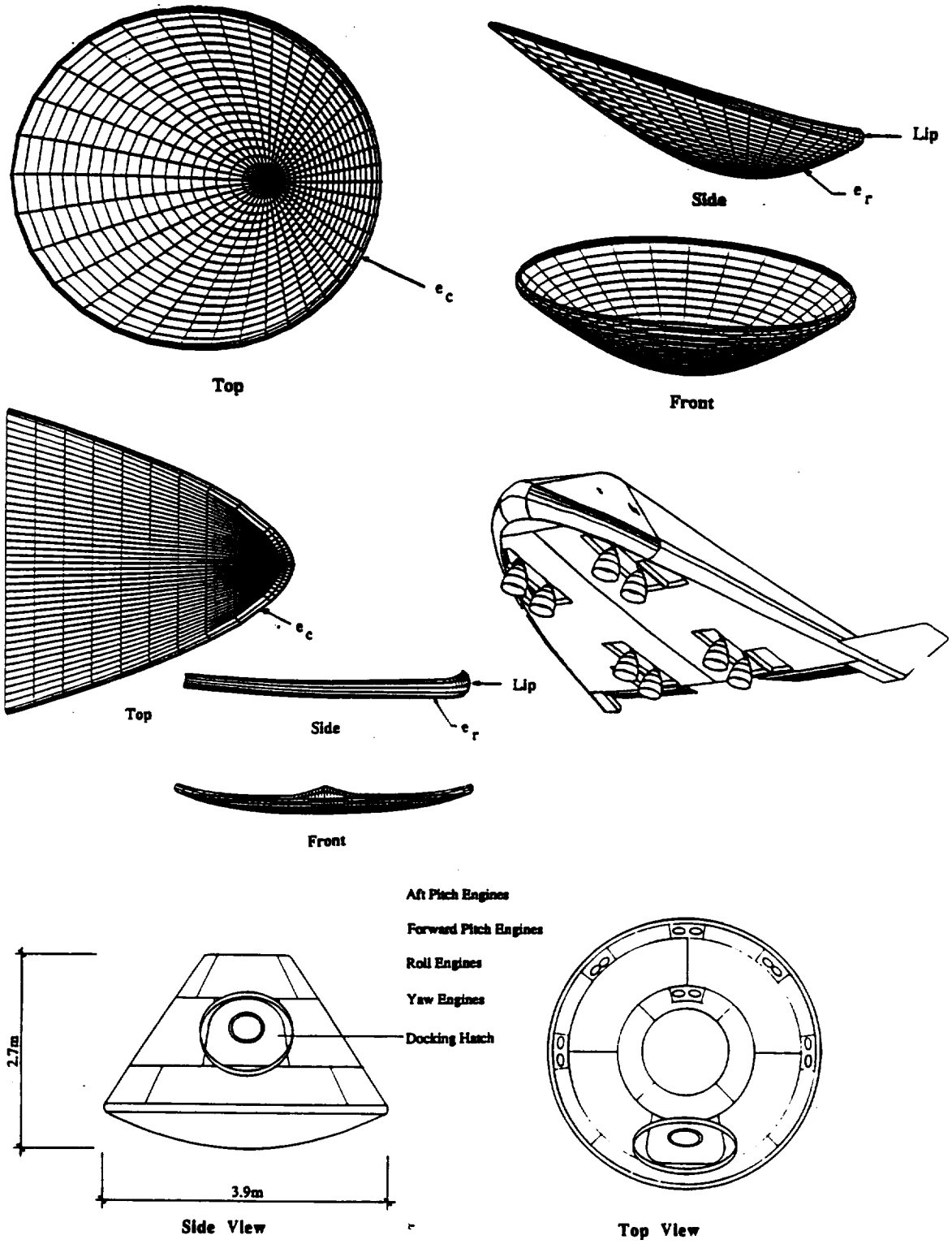
EFFECT OF BURN LOCATION ON ORBIT TRIM DELTA V (CONTINUED)
FIGURE 3-18

3.2.1.5 Aeroheating Analysis

Aerothermal analyses were performed for a variety of cryogenic/ aerobrake mission vehicles under varying entry conditions for differing mission opportunities. Three specific shapes were examined, the low L/D blunt hyperboloid (L/D ~0.5), the high L/D truncated hyperbola (L/D~ 1.0) and the Apollo type spherical shape, Figure 3-19. These vehicles included the MTV, MEV and ECCV. Examinations were made of aerocapture and entry for both Mars and Earth conditions.

3.2.1.5.1 Mars Thermal Environments

Initial studies were performed to define the stagnation point heating for Mars aerocapture. Analyses were made using the MARSIN code, initially developed by Chul Park at NASA Ames Research Center (Reference 21) and modified by Boeing. MARSIN is a 2-D trajectory and heating analysis program with both convective and radiative heating. The program iterates on initial entry angle until the aerocapture solution is reached. At the same time calculations are made for stagnation point heating. Radiation predictions in

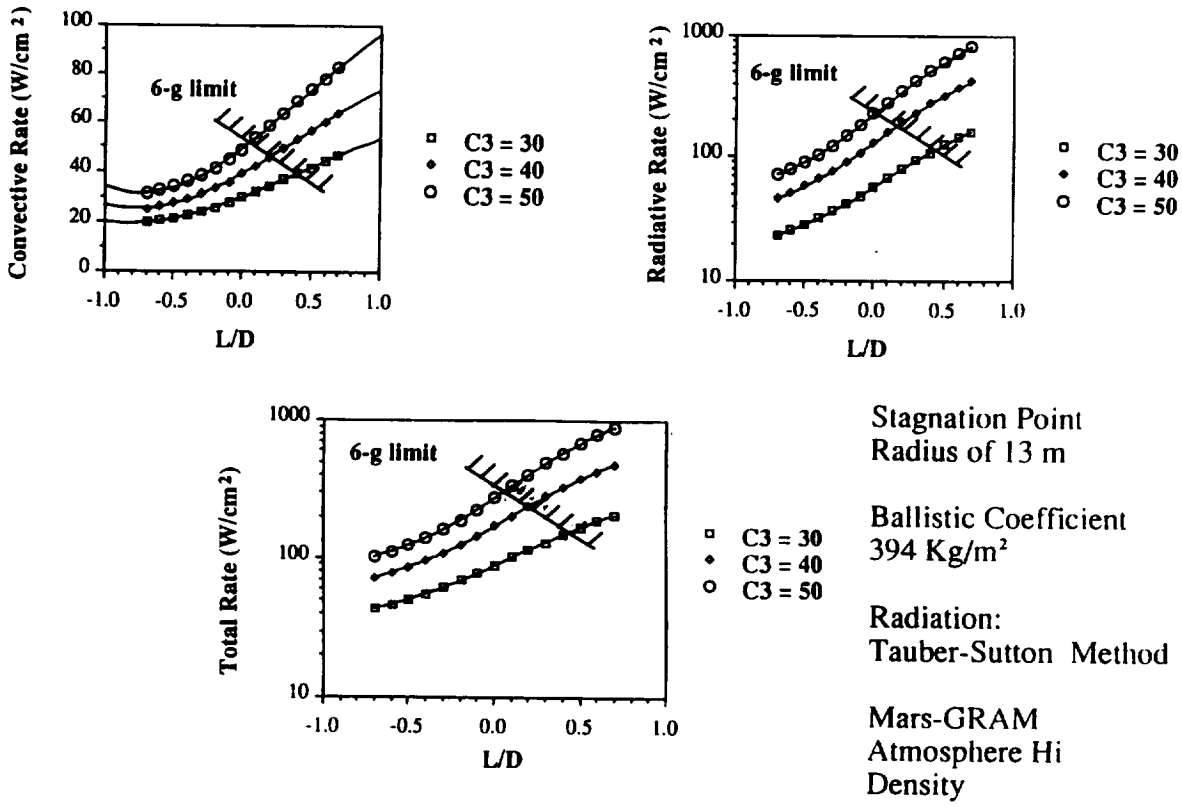


AEROBRAKES, (TOP) L/D=0.5, (MIDDLE) L/D=1.0, (BOTTOM) ECCV
FIGURE 3-19

the program were made using either the Tauber-Sutton relationships or Park's tabular NEQAIR results. A constant 50% absorption in the shock layer was assumed with the NEQAIR values which is not entirely correct. Therefore, all results presented here are based on the Tauber -Sutton method (References 22 and 23). The atmosphere models used in the program are either COSPAR high, or any chosen profile from the MarsGRAM program (Reference 24). An examination of the types of density profiles encountered reveal that the MarsGRAM density profile for the Winter Solstice in the year 2016 resulted in the most severe heating. Aerocaptures were made into 24.6 hour parking orbits with an initial entry altitude of 150 km.

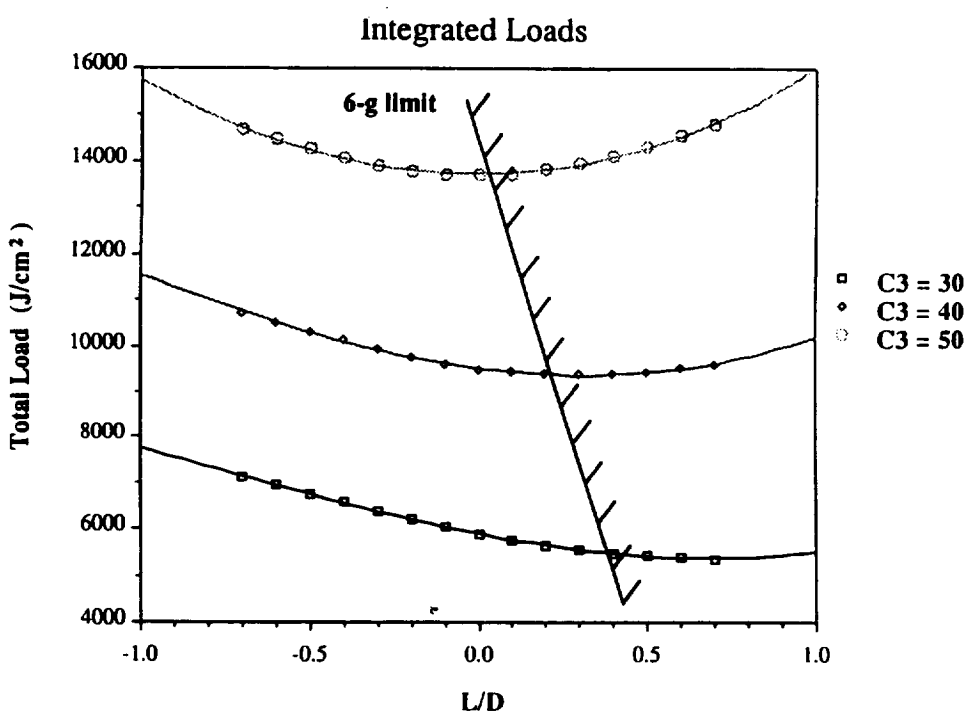
The extent of the study was to examine the effects of entry velocity and lift to drag ratio on peak stagnation point heating for the MTV. A blunt hyperboloid low L/D MTV aerobrake was examined as it results in the most severe heating conditions. The vehicle has an effective nose radius of 13 m, a 140 t mass, an L/D of 0.5, and a ballistic coefficient of 400 kg/m^2. The range of trajectories was from full lift down (overshoot boundary, L/D = -0.7) to full lift up (overshoot boundary, L/D = +0.7) with entry velocities from 6.6 km/sec to 8.6 km/sec, Figure 3-20. All data reveal that the peak stagnation point heating will be above the near term reradiative TPS capabilities for all of the up lift trajectories and that some form of down lift modulation will be required to reduce the heating environment. The corresponding integrated heat loads for these trajectories are displayed in Figure 3-21. Results for the blunt hyperboloid zero lift ballistic trajectories with an entry velocity of 7.4 km/sec show that the stagnation point heating is 65% radiative, with a total heating rate of 83 W/cm^2, Figure 3-22. The percentage of radiative heating increases with entry velocity. At 8.0 km/sec and 8.6 km/sec the radiative heating is 77% and 82% of the total heating rate respectively. At present the extent of the radiation in the shock layer is uncertain. It is possible that the shock layer will be in non-equilibrium, which could drive the radiation heating to higher values.

The effect of varying the nose radius on heating rate is shown in Figure 3-23. Minimum heating occurs at a nose radius of about 3 meters. Heating distributions were examined for both the low L/D blunt hyperboloid and the high L/D truncated hyperbola. Entry trajectories investigated were the 7.4 km/sec and the 8.6 km/sec entries, with zero lift (equal lift up and lift down) to approximate maximum heating conditions when flying modulated lift cases. Flying in the middle of the aerocapture corridor is desired to provide a margin for GN&C errors. These trajectories result in representative heating cases for actual aerocaptures. Calculations for the distributedheating were made at the maximum



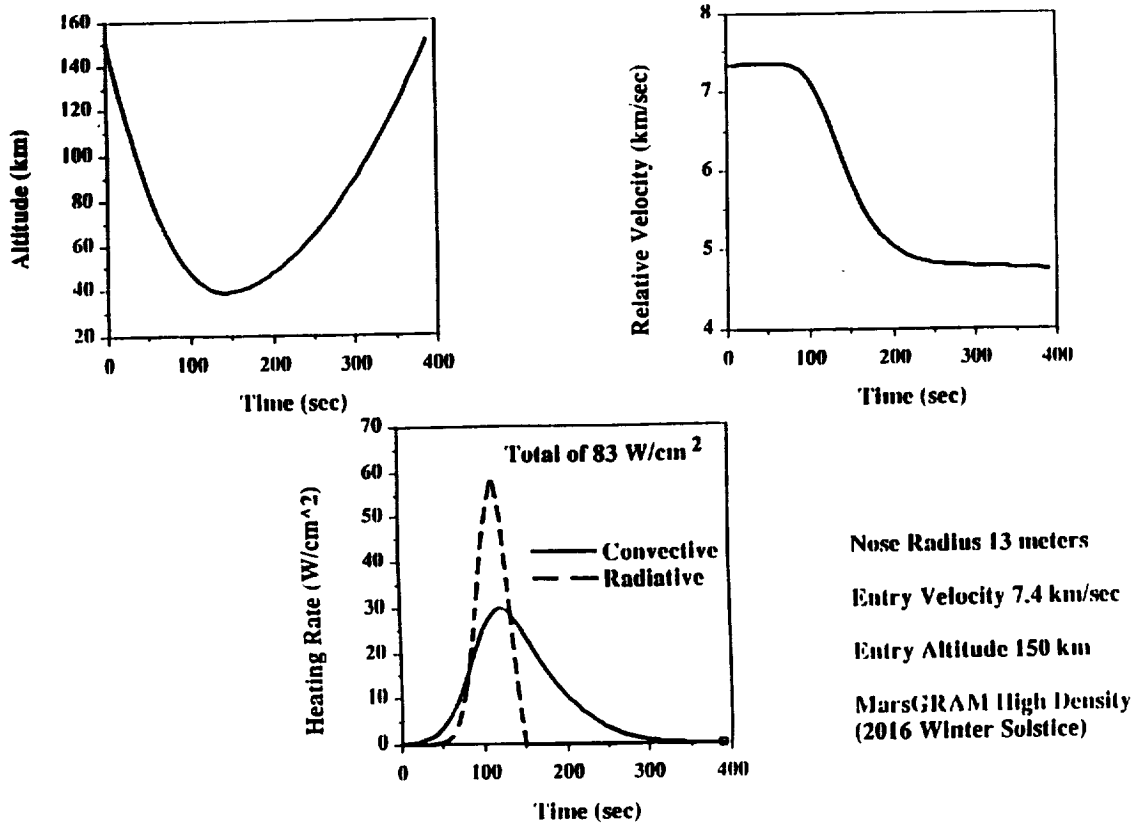
MARS AEROCAPTURE PEAK STAGNATION POINT HEATING RATES

FIGURE 3-20



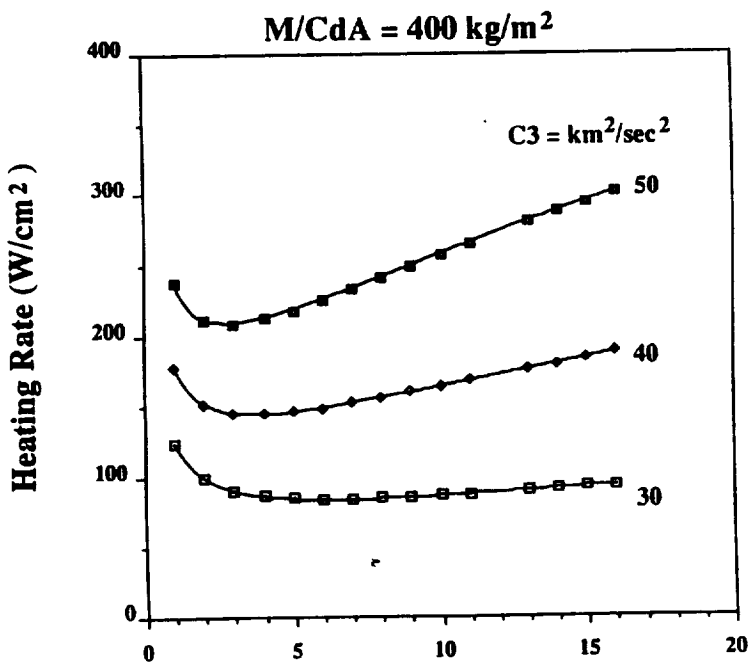
MARS AEROCAPTURE STAGNATION POINT HEATING LOADS

FIGURE 3-21



MARS AEROCAPTURE HEATING C3=30 km^2/sec^2

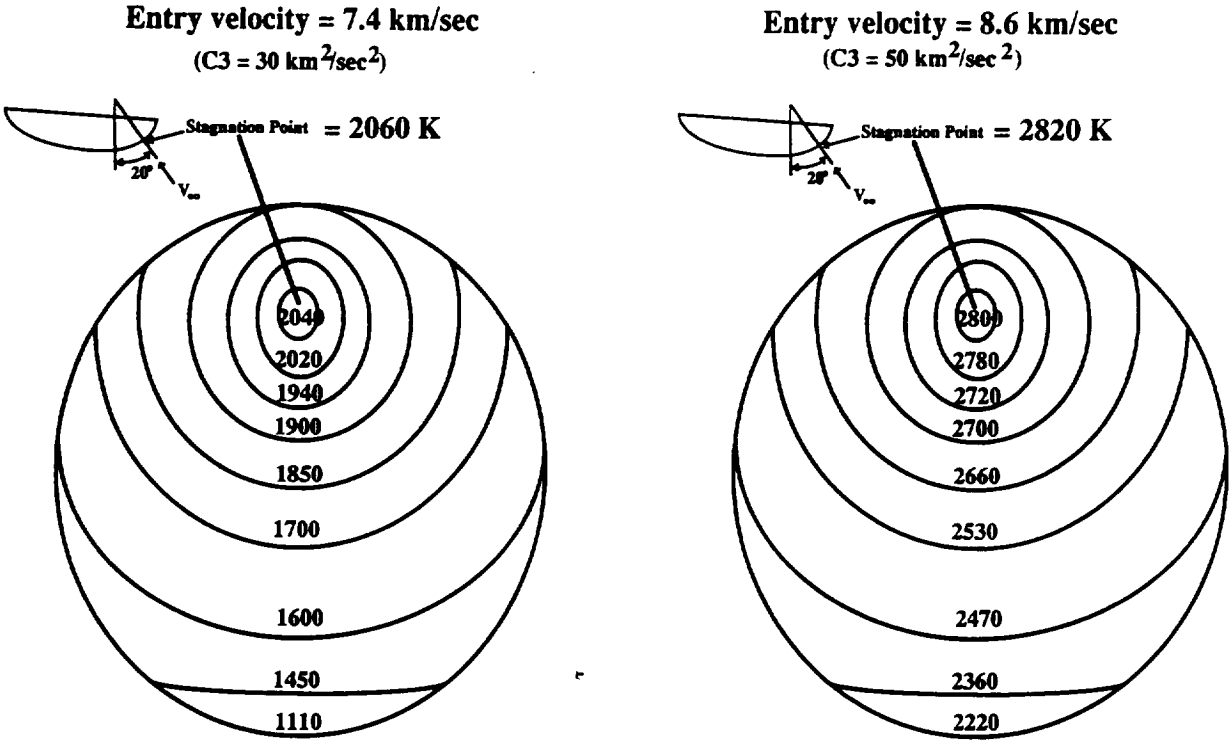
FIGURE 3-22



EFFECT OF NOSE RADIUS FOR MARS AEROCAPTURE, M/CdA=400 kg/m2

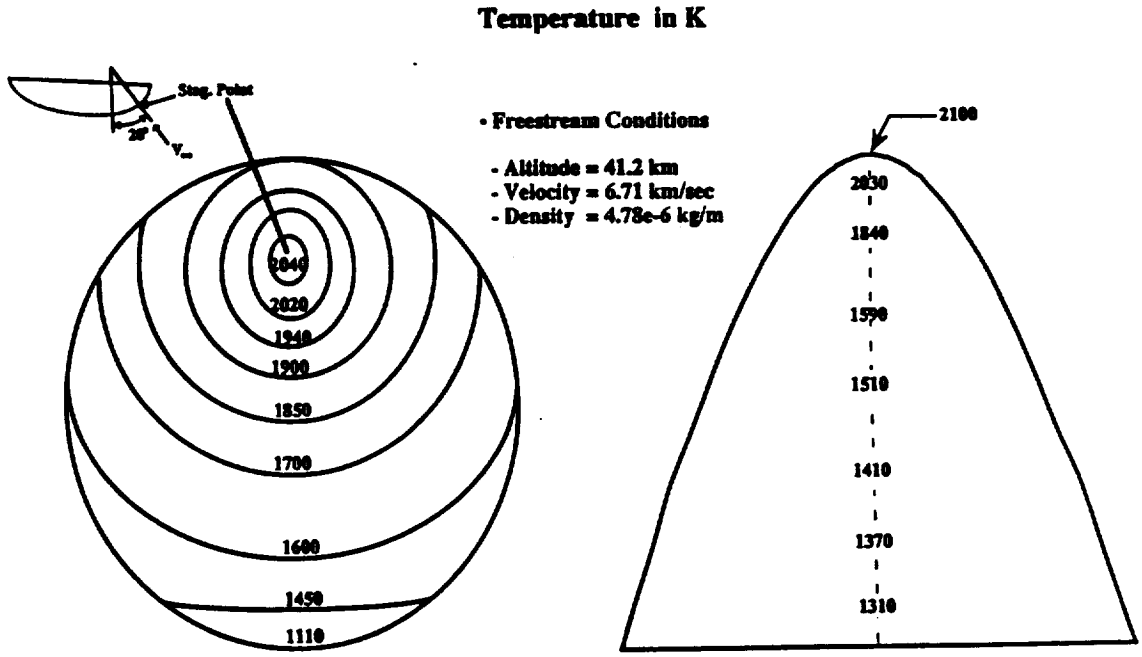
FIGURE 3-23

heating point along the aerocapture trajectories. The distributed heating for the blunt low L/D aerobrake for the two entry velocities is shown in Figure 3-24. For the 7.4 km/sec case maximum heating occurred at 120 seconds. The peak stagnation point heating values for the low and high L/D configurations at 120 seconds are 83 W/cm^2 and 90 W/cm^2 respectively, Figure 3-25. The distributed convective heating calculations were made using the Boeing Boundary Layer Analysis Program (BLAP) with thermodynamic and transport properties for CO_2 (Mars atmosphere). Radiative heating calculations were made using the Tauber-Sutton relations along with the Aeroassisted Flight Experiment Thermodynamic Handbook method for computing the radiative distribution (Reference 25). The findings of this analysis show that although the high L/D vehicle has higher stagnation point heating, the centerline heating decreases more rapidly than the centerline heating for the low L/D vehicle. As a result of this rapid decrease the high L/D vehicle has a smaller area requiring above near term reradiative TPS capabilities of approximately 1800K. A heating analysis was also performed for the low L/D configuration along a trajectory with an entry velocity of 8.6 km/sec. For this case the entire forebody was above 1800K (see Figure 3-24) and would require extensive use of advanced reradiative or ablative TPS.



MARS AEROCAPTURE TEMPERATURE DISTRIBUTION C3=30 & 50 km^2/sec^2

FIGURE 3-24



TEMPERATURE (KELVIN) DISTRIBUTION FOR HIGH AND LOW L/D AEROBRAKE

FIGURE 3-25

Descent heating calculations were made for the high L/D MEV (at present a reusable descent only design), since in the advanced propulsion missions no aerocapture maneuvers are required and transportation vehicles lend themselves to reuse. The descent trajectory was an OTIS trajectory optimized for crossrange. Radiative heating was neglected due to the fact that all descent velocities are below 6 km/sec (which is considered the cut off point on the radiation curve below which effects are considered negligible). For this trajectory, maximum radiative heating was on the order of 5×10^{-5} W/cm^2. Convective heating rates were calculated using the BLAP code and swept cylinder theory. The maximum heating rates of 14 W/cm^2 and 5 W/cm^2, for the stagnation point and leading edge at the 70 degrees point respectively, are only one sixth the values of the areocapture heating. With this vehicle a hot structure thermal protection system is adequate.

3.2.1.5.2 Earth Return Thermal Environment

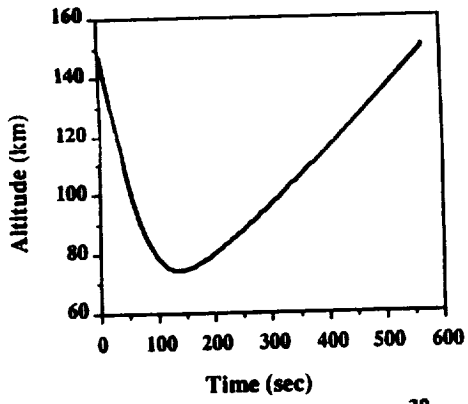
Heating analysis for the Earth conditions consisted of a parametric study of the MTV aerocapture (cryogenic /aerobrake vehicle) and an examination of the ECCV aerocapture. Return C3's of approach ranged from 20 km^2/sec^2 to 50 km^2/sec^2 or entry velocities of 11.9 to 13.1 km/sec. Mars return trajectories and heating rates were calculated using MARSRT (the same program as MARSIN with properties of air instead of of CO_2). Once

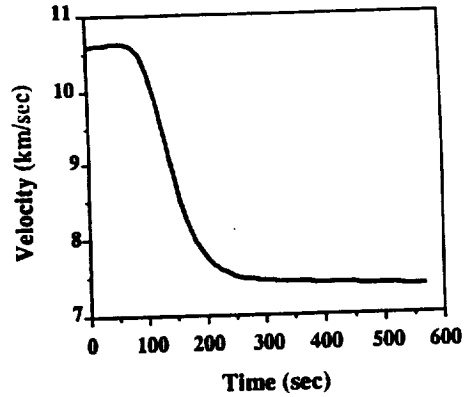
again all data revealed that the stagnation point heating values can be reduced by using down lift modulation. For the ECCV on a nominal ballistic return trajectory of 12.3 km/sec, the maximum stagnation point heating rate was 537 W/cm^2, which is twice the maximum heating rate for a Mars MTV aerocapture. Stagnation heating for the MTV return trajectory for entry velocity of 12.3 km/sec and 13.1 km/sec are 533 W/cm^2 and 950 W/cm^2 respectively. These values are two and four times higher than the maximum Mars aerocapture value. For either of these vehicles returning from Mars, an ablative heat shield will be required.

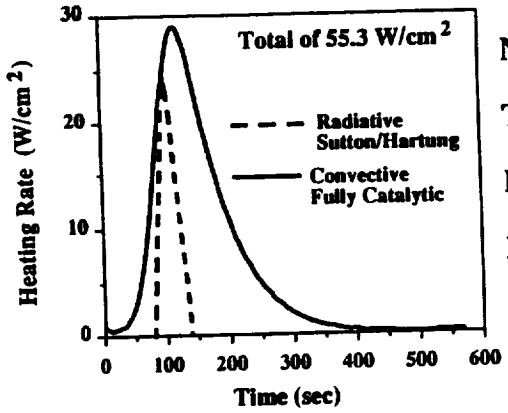
Lunar return heating analyses were also made for the LTV, Figure 3-26. Lunar return trajectories for the LTV were computed with an Earth return mass of 26.5 metric tons. The aerobrake is a symmetrical hyperbola (derived from the MTV low L/D aerobrake) and has an L/D capability of 0.25 at a 20° angle of attack, with a diameter of 22 meters and reference area of $380m^2$. The ballistic coefficient used for Earth return was 52 kg/m^2. The LTV Earth aerocapture trajectory is displayed in Figure 3-26, for an entry velocity of 11.05 km/sec and a target apogee of 407 km. The Lunar return heat value of $55.3W/cm^2$ is an order of magnitude smaller than the MTV Mars return maximum heating rate. The resulting distributed heating at this maximum heating point is displayed in Figure 3-26. All of these temperatures fall well within the capabilities of near-term advances of reradiative materials.

3.2.1.6 Guidance Navigation and Control

To perform the maneuvers of aerocapture a scheme of guidance, navigation and control was devised. For the baseline cryo/aerobrake vehicle the top portion of the atmosphere was used to slow the vehicle by drag to capture it in orbit. All missions use an aerobrake for descent and passage through the atmosphere for ascent. It is required that the Mars atmosphere be reasonably understood to use it to set vehicle performance requirements. The standard MarsGRAM atmosphere description program was modified so that we could generate density tables for in-house flight simulation and heating codes such as AEROPASS, OPTIC, OTIS, BLIMPK, and MISSION. This enabled us to use probable atmospheres as projected from known Viking and Mariner data for the chosen region of aerocapture /descent. Most of our studies focused on the equatorial region for which real data doesn't exist.







Nose Radius 13 meters

Target Apoapsis 407 km

Entry Altitude 150 km

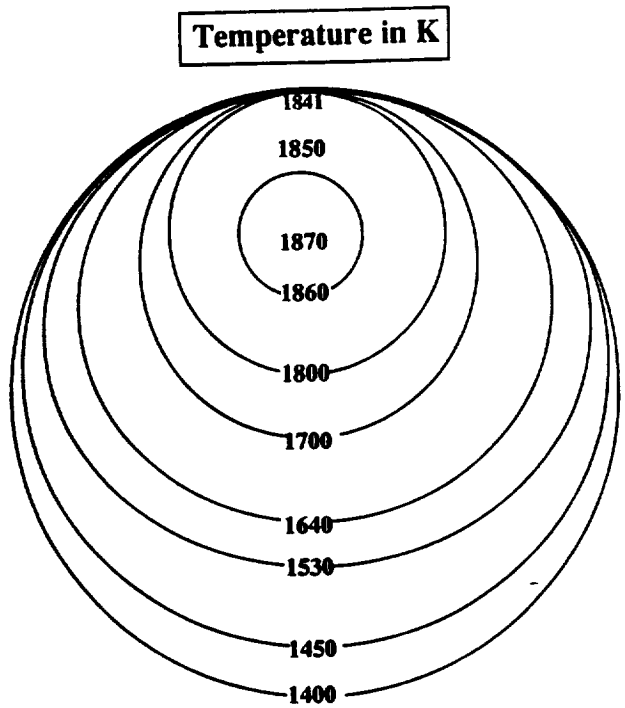
Entry Velocity 11.05 km/sec

Assumptions

- **Radiative - Sutton/Hartung**
- **Convective - BLAPP**
- **Emissivity = 0.8**

Conditions at maximum heating

- **Velocity : 10.32 km/sec**
- **Altitude : 78.75 km**
- **Radiative : 23.7 W/cm^2**
- **Convective : 31.6 W/cm^2**



LTV AEROCAPTURE, (TOP) TRAJECTORY AND HEATING RATE, (BOTTOM) TEMPERATURE K

FIGURE 3-26

Modifications were made so that for a chosen date and time one could determine the position of Mars in its orbit about the Sun. The right ascension and declination of the Sun are output so that the high density region is known for that date and time.

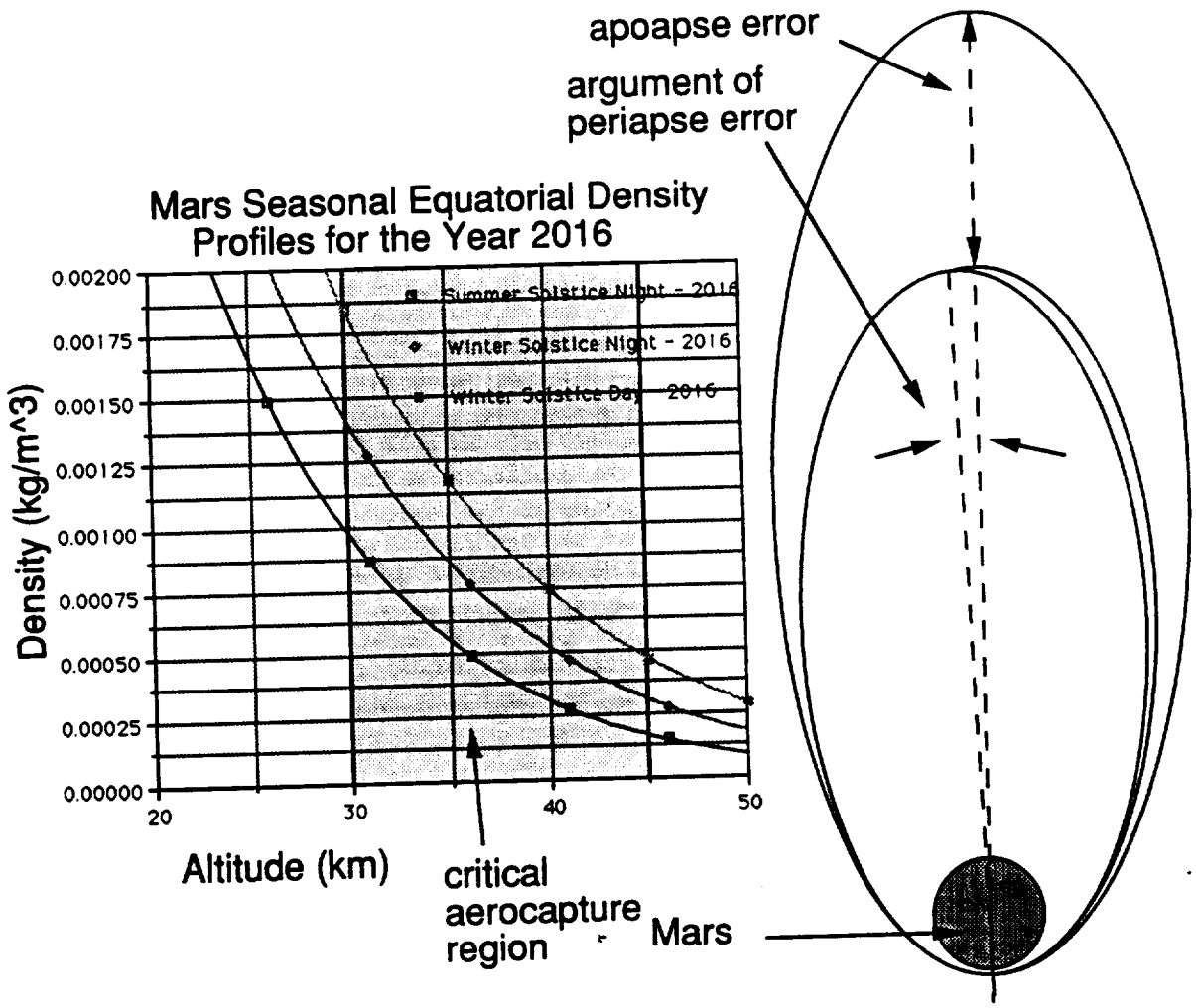
The Mars orbit for the year 2016 was examined at Winter solstice, Summer solstice and Spring equinox for the average high, average low and average mean density profiles. These density profiles were used for subsequent aerocapture GN&C and descent studies and simulations.

Traditional GN&C schemes employ the use of a predictor-corrector algorithm. An alternate scheme employed is one in which the guidance system follows a predetermined flight path angle versus velocity profile. The data are obtained from a flight profile through a normal atmosphere which meets all the normal parking orbit constraints and the heat flux and g-load limits (6 Earth gs). GN&C errors and atmospheric perturbations will create accumulated errors which will bias the guidance profile. The guidance profile of the flight through the atmosphere (shaded) is illustrated in Figure 1-32. The error on capture orbit due to density variation is shown in Figure 3-27.

Preliminary first approximations were made to look at the possible effects of local and global dust storms on aerocapture parking orbit errors. Initial results from guided runs using the alternate scheme indicate that global dust storms can create gross apoapse (radius position) and argument of periapse parking orbit errors, Figures 3-28 and 3-29. This is because at the critical aerocapture region, the unexpectedly large density profile depletes excessive energy from the spacecraft creating gross errors in apoapse radius position (altitude). The aerocapturing vehicle has to exit the atmosphere sooner, making correction of the line of apsides difficult. Localized dust storms create only gross apoapse errors because the density perturbation occurs in the critical aerocapture region only.

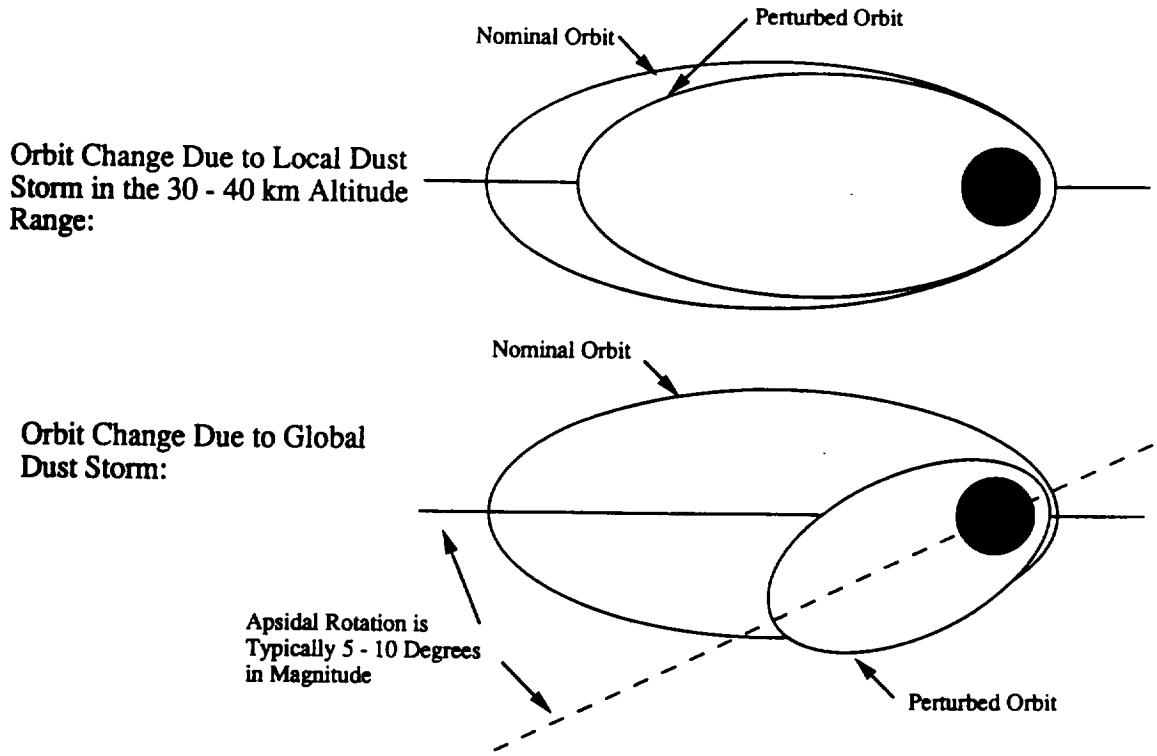
The Mars Atmospheric Working Group (in which Boeing was a participant) raised a concern about the effect of Mars atmospheric uncertainties on aerocapture GN&C. We proposed a synthetic sinusoidal density wave equation which could be implemented in guided aerocapture simulations to test GN&C "robustness." The amplitude, wavelength, phase angle and bounded altitude regions in the atmosphere were varied to study the importance of each in creating GN&C errors. Delta V errors for GN&C were found for

non-adaptive and adaptive guidance, Figure 3-30. The large amplitudes (greater than 30 percent) severely affect GN&C performance, which was measured by the required post-aerocapture apoapse delta V correction. High frequencies act as density shears (errors tend to average out) that severely affect GN&C performance. However, large wavelengths, those greater than 800 km, act as density biases. In this case the phase angle, or where the sine wave begins, is important. If the pre-periapse portion of the aerocapture experiences a less than normal density profile while the post-periapse portion experiences a greater than nominal density profile, GN&C errors can still be minimized because the vehicle has significant control authority (due to the 'thick' atmosphere). If the opposite scenario occurs, that is the atmosphere is less than nominal post-periapse, then the vehicle's control authority is decreased and GN&C errors are more difficult to correct.



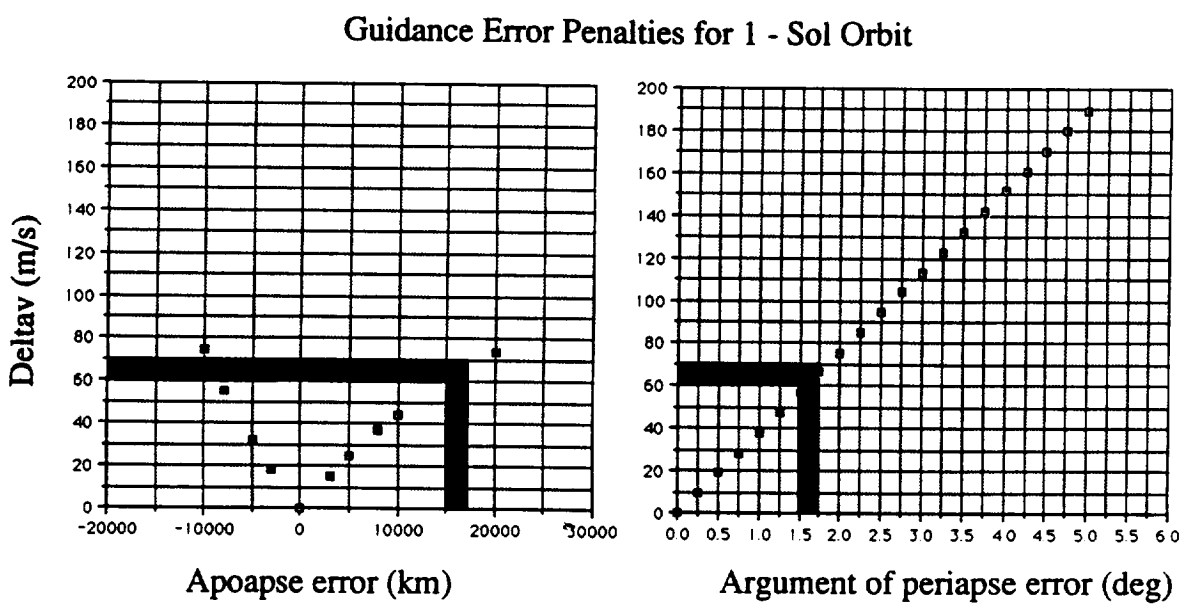
ERRORS DUE TO DENSITY VARIATION

FIGURE 3-27



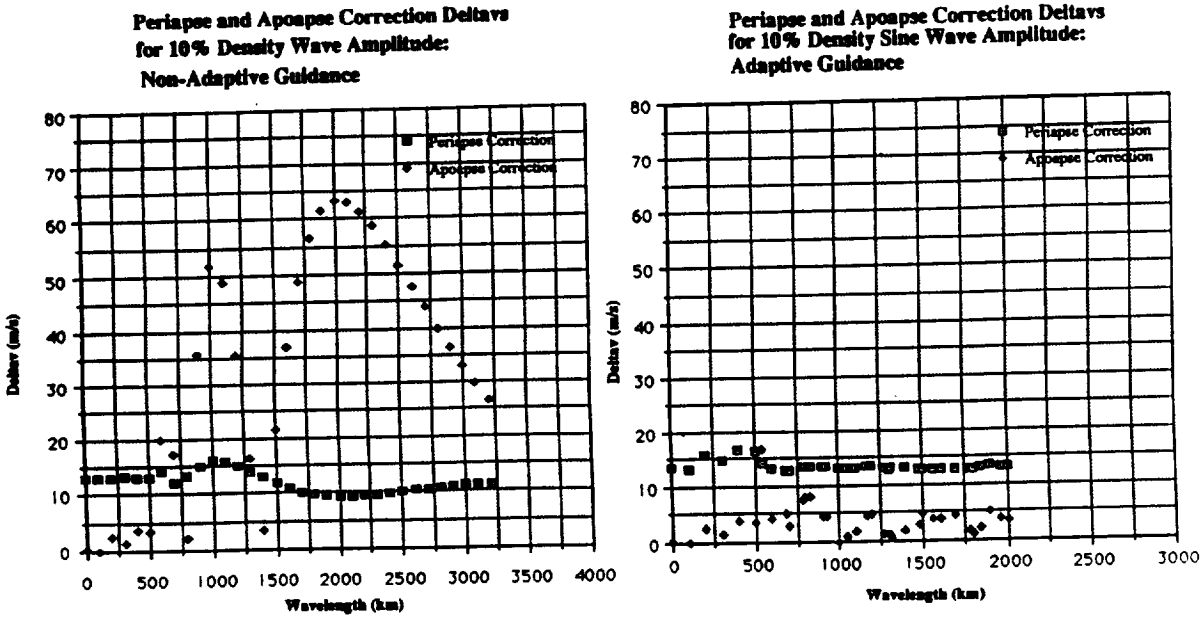
PARKING ORBIT ERRORS DUE TO DUST STORMS

FIGURE 3-28



DELTA V PENALTY DUE TO PARKING ORBIT ERRORS

FIGURE 3-29



DELTA V ERRORS FOR ADAPTIVE AND NON-ADAPTIVE GUIDANCE
FIGURE 3-30

Preliminary studies on the effectiveness of a predictor-corrector algorithm for Mars aerocapture have also been carried out (Reference 26). Results show that the MarsGRAM high density profiles with random perturbations have the greatest impact on GN&C accuracy. GN&C performance degrades with decreasing L/D with the minimal required L/D being about 0.3. The parking orbit element argument of periapse cannot be actively constrained with this algorithm.

Further studies need to be performed to determine the effects of Mars atmosphere perturbations on aerocapture GN&C. This will determine the depth of Mars atmosphere knowledge needed for manned missions. In addition the problem of argument of periapse control must be worked for missions with highly elliptic orbits.

3.2.1.7 Mars Descent Crossrange Analysis

Crossrange is a valuable feature of vehicle design for access options to different areas on the Martain surface. An analysis was carried out to determine the effect that L/D and atmosphere have on the crossrange. These analyses were performed using the OTIS program.

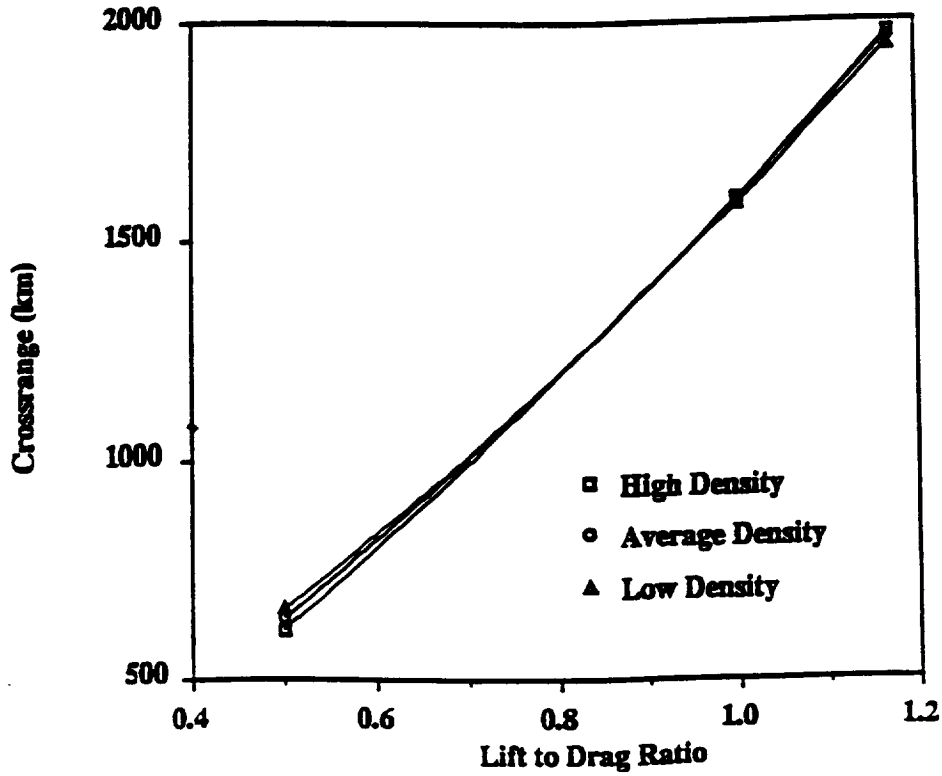
Two types of descent scenarios were studied. The first was a descent in which the crossrange was maximized. The second was a pinpoint descent. In this case the simulation was constrained to end at a certain location (defined by latitude and longitude coordinates) and the final velocity was minimized.

Both types of descent trajectories were generated using OTIS (Optimal Trajectories by Implicit Simulation). OTIS uses a non-linear solver to optimize the trajectories with respect to one or more control variables. For the descent trajectories considered, the only control variable was the bank angle which was allowed to vary within specified constraints. The actual limits on the bank angle depended on the problem. For both types of descent cases, the analysis began from a 1-sol elliptic parking orbit (24.6 hour period) with the orbit periapsis in the atmosphere. In the first descent case the simulation started at an altitude of 100 km and was constrained to end within a certain altitude and velocity range. In the pinpoint descent case the simulation began with an altitude of 112 km and was constrained to end within a certain altitude, latitude and longitude range.

The results of this study showed that the type of atmosphere has no significant effect on the amount of crossrange generated or on the vehicle's ability to reach a specific location. The results of the case in which the crossrange was maximized is shown in Figure 3-31. This figure indicates that the crossrange is primarily a function of L/D.

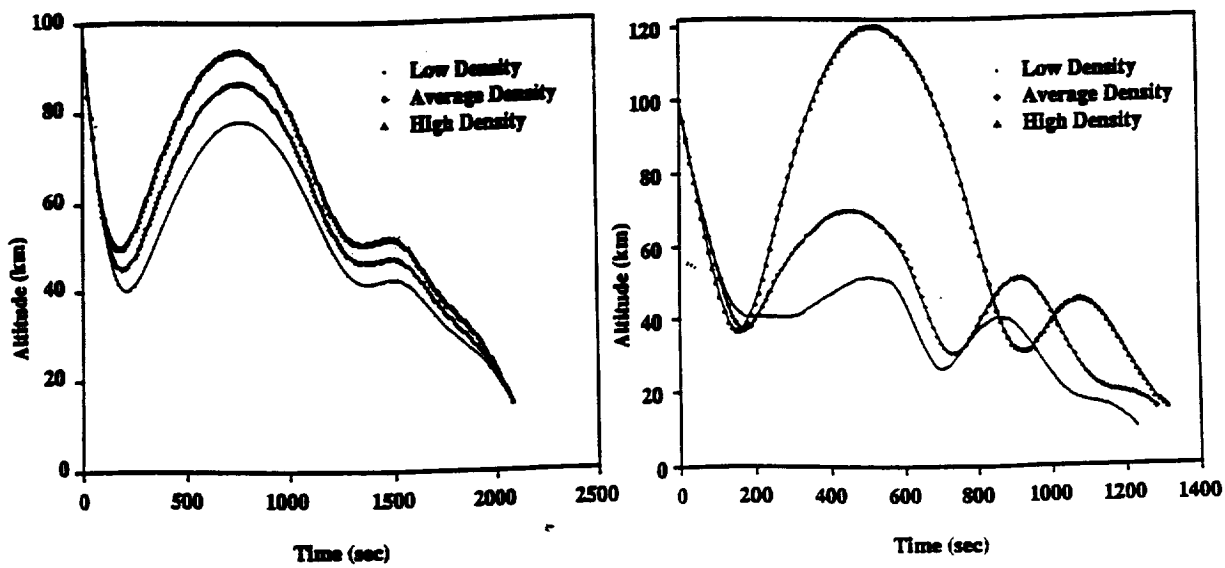
The altitude time histories for different atmospheres for the L/D of 1.0 for maximized crossrange and pinpoint descents are shown in Figure 3-32. In all the cases there is an initial dive into the atmosphere, a large skip (but not a skip out), and another dive corresponding to the final descent. As a result of the initial dive enought lift is generated to raise the flight path angle to 0 degrees (the flight path angle starts out at approximately -10 degrees) and the vehicle glides for most of the trajectory.

In both scenarios most of the trajectory is spent in the upper atmosphere, which means that local dust storms will probably have little or no impact on the trajectory. It is believed that effects of a local dust storm are felt only up to an altitude of 40 km. The graphs indicate that most of the trajectory is above this altitude for both descent types. Even though a dust storm was not simulated, it is expected that the effects would be minimal.



L/D EFFECT ON CROSSRANGE

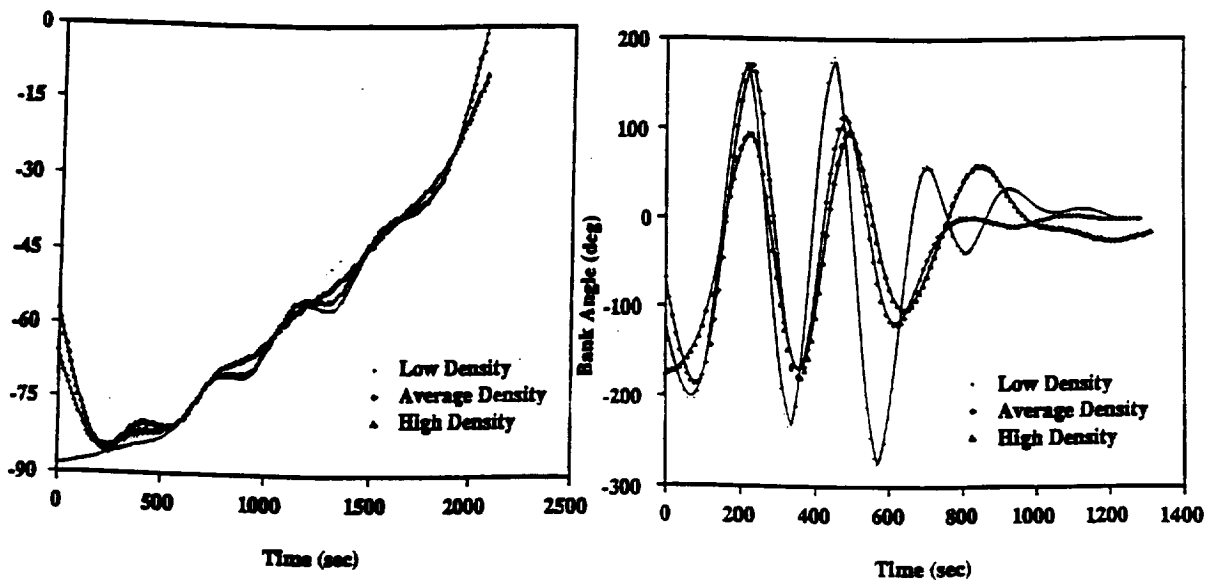
FIGURE 3-31



OPTIMIZED TRAJECTORIES FOR VARYING ATMOSPHERES, (LEFT) MAXIMIZED CROSSRANGE, (RIGHT) PINPOINT DESCENT

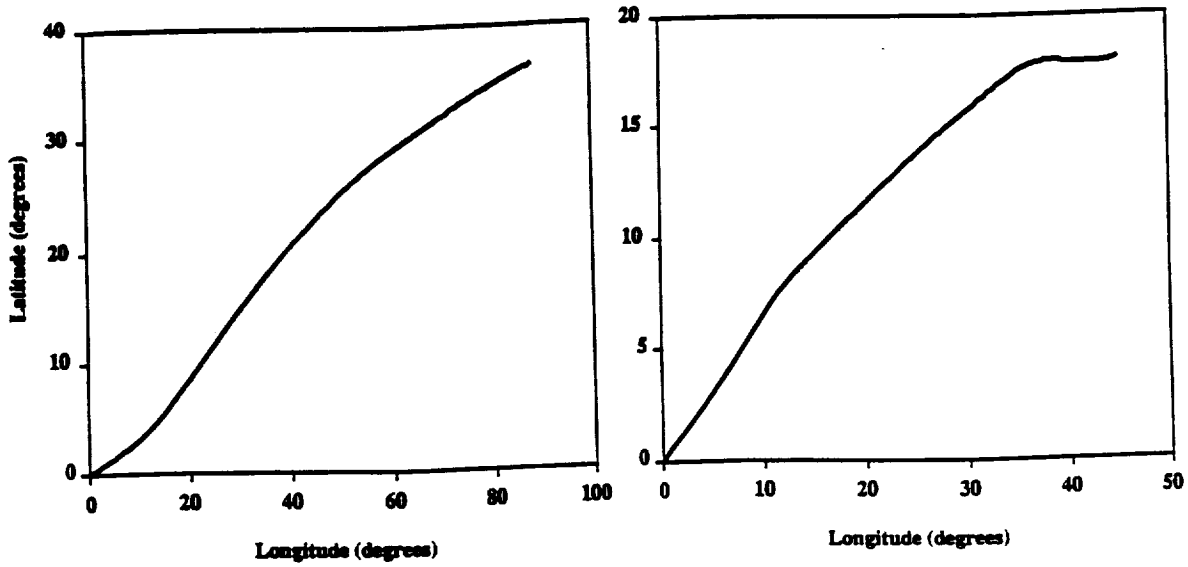
FIGURE 3-32

The bank angle time histories for both types of descent are shown in Figure 3-33. In the case of the maximum crossrange descent the bank angle is initially large (greater than -60 degrees), increases to approximately -90 degrees, and then begins to decrease. The vehicle wants to bank 90 degrees in order to turn and generate the maximum crossrange. At the same time, lift in the vertical direction must be generated in order to raise the flight path angle. The bank angle begins to decrease when the vehicle begins to climb (after the initial dive). As the altitude increases the density decreases and more vertical lift is required to keep the vehicle in the atmosphere. The bank angle continues to decrease towards zero after the start of the final dive in order to keep the vehicle in the atmosphere as long as possible.



OPTIMIZED BANK ANGLES FOR VARYING ATMOSPHERES, (LEFT) MAXIMIZED CROSSRANGE, (RIGHT) PINPOINT DESCENT
FIGURE 3-33

In the pinpoint descent case several roll reversals are performed before the bank angle oscillations dampen out. The reason for this behavior is that the vehicle is trying to deplete energy in order to meet the final altitude, latitude and longitude constraints. A sample ground trace for both descent scenarios for the high density atmosphere is shown in Figure 3-34. The variation in the ground traces is due to the different final constraints of each descent problem.



GROUND TRACE FOR HIGH DENSITY ATMOSPHERE, (LEFT) MAXIMIZED CROSSRANGE, (RIGHT) PINPOINT DESCENT
FIGURE 3-34

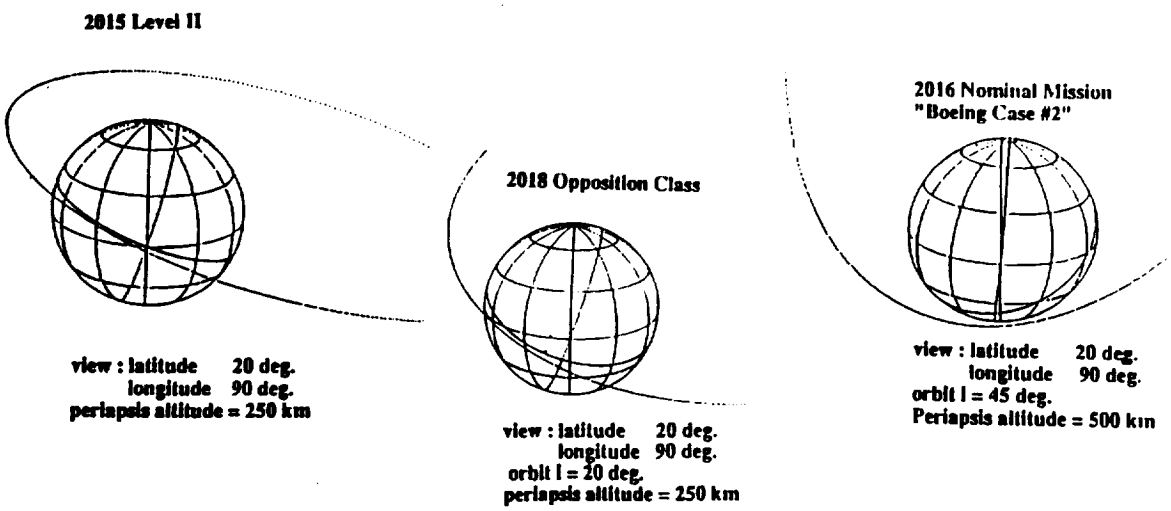
3.2.1.8 Mars Ascent Analysis

The analysis was performed to examine the drag losses associated with the reusable MEV during Martain ascent. OTIS also used to perform this analysis. The trajectory began at the Mars surface and ascended to a nearly circular orbit (the final altitude was 98 km).

The results showed that the velocity losses due to drag were minimal. When an average density atmosphere was used the velocity loss due to drag was 180 m/sec. For the high density atmosphere the velocity loss due to drag was 220 m/sec .

3.2.1.9 Standard Orbit Analysis

An analysis was conducted to determine if a module that was left in orbit around Mars at the end of one mission could be encountered during another subsequent mission without a large delta V penalty to re-acquire the orbit. The PLANET Program was used to perform the analysis by varying inclinations and periods for the standard orbit. The mission in which the module was put into orbit (baseline) was the 2016 opposition opportunity. This mission corresponds to the Boeing Nominal (Case 2) and the Level II Reference conditions. The 2018 and 2024 opposition opportunities were used as the return mission, Figure 3-35. The standard orbit was propagated according to which mission was being studied. The results of this analysis did not yield a desirable standard orbit. For all the cases studied either the insertion delta V or the departure delta V was large. Further work is needed on the analysis of the standard orbit problem.



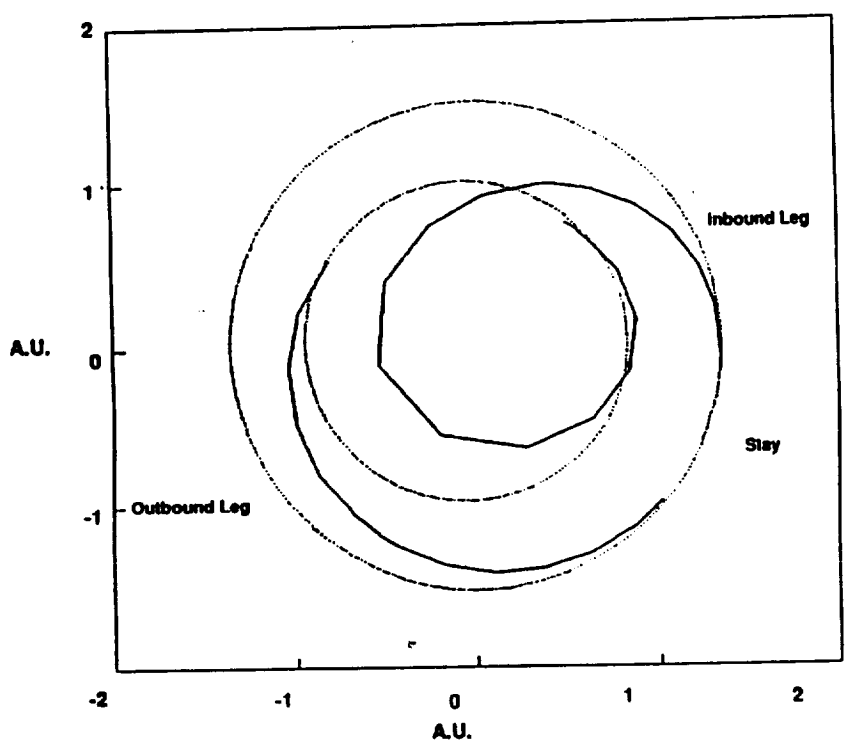
STANDARD ORBIT SCENARIO
FIGURE 3-35

3.2.2 Low Thrust Missions

The low thrust missions covered in this section involve the use of electric propulsion systems with either a solar electric (photovoltaic) or nuclear system to generate electricity for thrusters working on argon propellant electrostatic systems.

3.2.2.1 Historical Perspective

In early studies, the main focus of electric propulsion (EP) mission analysis was on planetary probe and cargo applications. Electric propulsion for piloted missions was not considered as a serious contender for manned Mars missions. Due to the inherent characteristics of electric propulsion, i.e., low thrust, the propulsion system lends itself to applications where there is a reasonably long time to accelerate and decelerate the spacecraft. The EP systems cannot perform fast orbital transfer missions in the Earth's gravity well, but can compete with high thrust propulsion options (chemical, nuclear thermal, etc.) in Earth-Mars heliocentric transfer, Figure 3-36. For EP to compete with high thrust options for manned Mars missions, power levels on the order of 10-50 MWe must be employed.



GENERAL EP INTERPLANETARY TRAJECTORY
FIGURE 3-36

In the current study, two low thrust systems were analyzed, the solar electric propulsion (SEP) and nuclear electric propulsion (NEP) options. The study revealed that EP can compete with high thrust options in terms of total manned trip times for Mars missions. This result was not as accepted in previous years when EP systems were mainly thought of as cargo carriers. One method that contributed to the shorter transfer times is the use of a Lunar and Mars gravity assist. Gravity assists are not unique in themselves, but when these specific gravity assists are coupled with the high powered EP trajectories, the trajectories are efficient.

Due to the high Isp (5,000 to 10,000 sec) associated with electric propulsion, several benefits are gained such as reduced propellant requirements, greater mission flexibility, elimination of high-energy aerobraking, and a lower sensitivity to mission opportunity variations. Electric propulsion initial masses do not vary as much throughout the Earth-Mars synodical cycle as does the high thrust options. Missions, such as the 2024 opportunity, that are not mass feasible for high thrust options are feasible from a mass standpoint for EP vehicles.

3.2.2.2 General Assumptions and Guidelines

At the onset of our analysis several ground rules were established to insure commonality in search of optimum trajectories. These ground rules included items such as Earth departure location, trip time definitions, Mars stay time, and propulsion system characteristics (References 9 and 10). Since the EP vehicles have a lengthy Earth orbit transfer time, the vehicles will spiral from an assembly point to near Earth escape before the crew will rendezvous with the transfer vehicle.

Mission analysis for both the NEP and SEP vehicles were performed by Byrd Tucker and Bill Vlases (References 27 and 28). Although CHEBYTOP was used by both analysts as the main program, different front end optimizers were used. The initial analysis was performed by Tucker using POP and CHEBYTOP to gain a parametric insight of where to initiate our point designs. Further analysis was performed by Vlases once detailed mass statements were defined. Vlases also had the capability to perform gravity assists which are discussed in the next section. The most current analysis with gravity assists included were performed by Vlases.

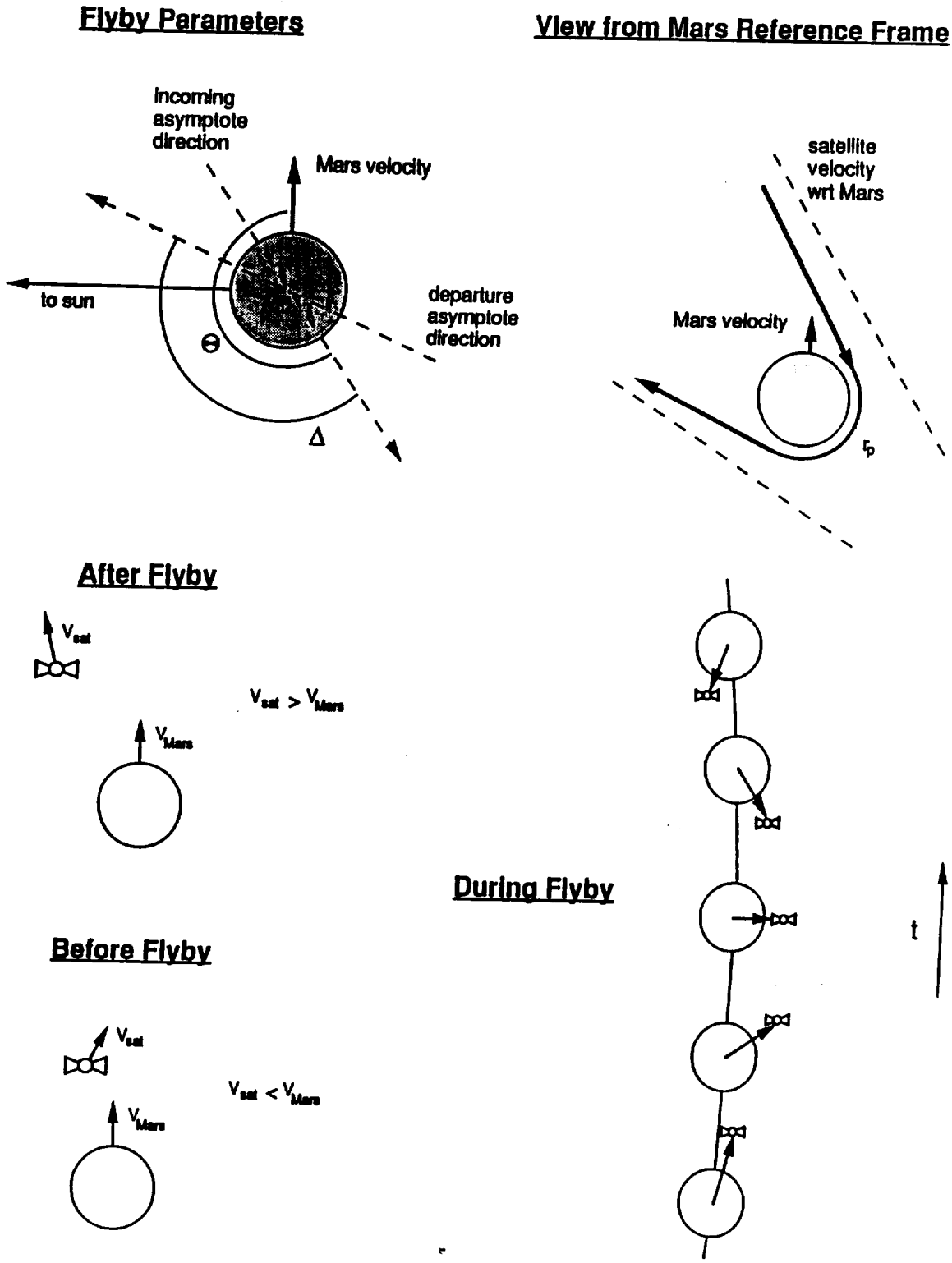
3.2.2.3 Gravity Assists

Certain gravity assists offer significant benefits for electric propulsion, without imposing launch window restrictions. The gravity assists that offer benefits are a Lunar fly-by, Mars fly-by, and an Earth fly-by. During Earth escape, the vehicle swings by the moon to gain a velocity boost on the order of 600-1000 m/s. During a Mars fly-by, the vehicle approaches Mars with excess velocity, drops the MEV off, and continues in heliocentric space in close proximity to Mars, Figure 3-37. When the vehicle decelerates enough to capture at Mars, the vehicle enters a highly elliptic orbit to allow the MEV multiple attempts to rendezvous with the transfer vehicle. The time frame for vehicle deceleration and Mars capture is calculated to be the same as the surface stay time. An Earth fly-by is similar to a Mars fly-by in the sense that the vehicle starts the deceleration phase of the mission leg later than normal. As the transfer vehicle approaches the Earth with excess velocity, the crew is dropped off and the vehicle continues in heliocentric space. When an Earth fly-by is employed, the transfer vehicle cannot rendezvous back with the Earth for a considerable length of time (~200 days). This length of time may be detrimental to thruster lifetime. Therefore, the recommended gravity assists are Lunar and Mars fly-bys. These fly-bys can offer trip time reductions on the order of 40 days total.

3.2.2.4 Solar Electric Propulsion

Mission analysis for SEP vehicles can be broken down into two distinct categories; opposition and conjunction. Virtually all mission analyses for this study were aimed at piloted missions. Mission analysis for cargo applications is more straightforward and can be interpolated from the lower power profiles that were performed.

The majority of analyses were performed for opposition class missions. There are two "legs" associated with an opposition mission, outbound and inbound. One of these two legs will have higher associated energy requirements. Since a majority of the payload mass is left at Mars, the trajectories lend themselves for a high energy inbound leg vs a high energy outbound leg. An opposition inbound leg requires the vehicle to come inside of the Earth's orbit and sometimes within 0.5 AU . This type of path allows the vehicle to "catch up" with the Earth. Stay times associated with this type of mission are on the order of 30-60 days . Longer stay times would result in the vehicle not being able to successfully return home. When on inbound leg of opposition mission is within 0.65 AU , the vehicle

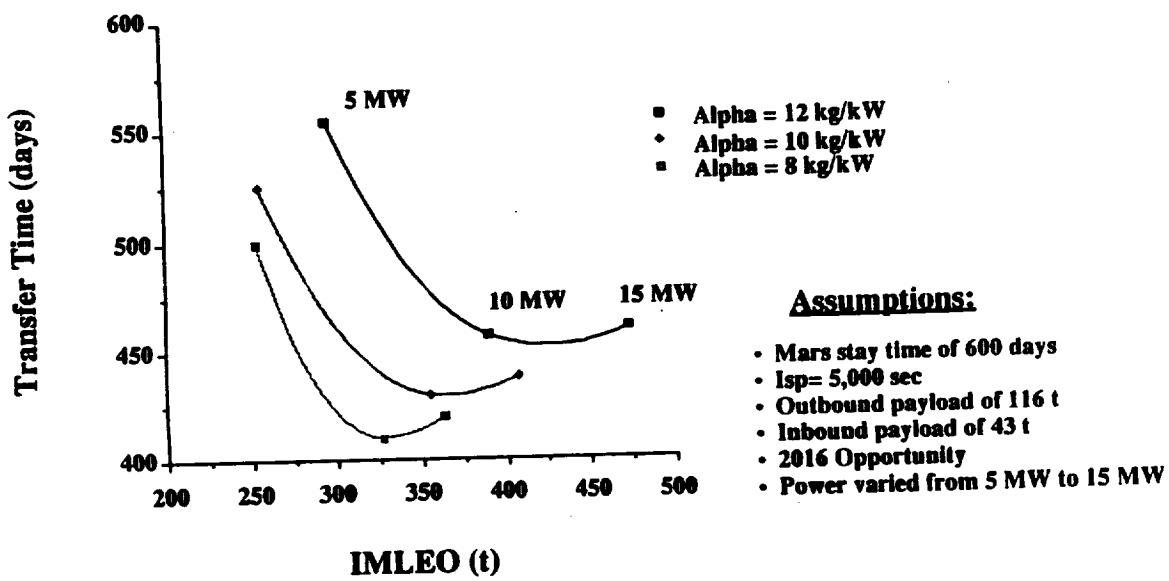


MARS FLYBY, (TOP) MARS REFERENCE FRAME, (BOTTOM) INERTIAL REFERENCE FRAME

FIGURE 3-37

will not fully face the sun and take solar flux cosine losses to keep the solar arrays from overheating.

Typical, total trip times associated with SEP opposition missions result in 530 to 630 days (including a 30 day stay time), Figure 3-38. Initial mass in low earth orbit (IMLEO) for this size of vehicle is typically in the 500 tonne to 300 tonne range, power levels are from 8 to 20 MWe. Specific impulse (Isp) for the SEP vehicles tended to optimize about 6500 sec. The solar flux decreases as the distance from the Sun increases. When the SEP vehicle is in Mars' vicinity (1.5 AU), the vehicle is at 50% of what the power level delivered to the thrusters at Earth. The specific mass of the SEP vehicle is also greater than that of the NEP which means that the thrust-to-weight ratio is less for the SEP. The reduced power at Mars and the lower thrust-to-weight ratio results in the different optimized Isp levels for the two vehicles.



SEP CONJUNCTION POWER TRADE
FIGURE 3-38

An important factor in the mission analysis is the vehicle alpha. Alpha is the ratio of total power and propulsion system dry mass to the electrical thruster input power and is referred to as the specific mass of the vehicle. Typical alpha values that have been analyzed for these multi-megawatt powered vehicles are in the range of 8-15 kg/KWe, with 10 kg/kWe

being the norm. To obtain fast trip times with the EP vehicles, alpha should be as low as possible. Technology and innovation will play the major role in reducing alpha.

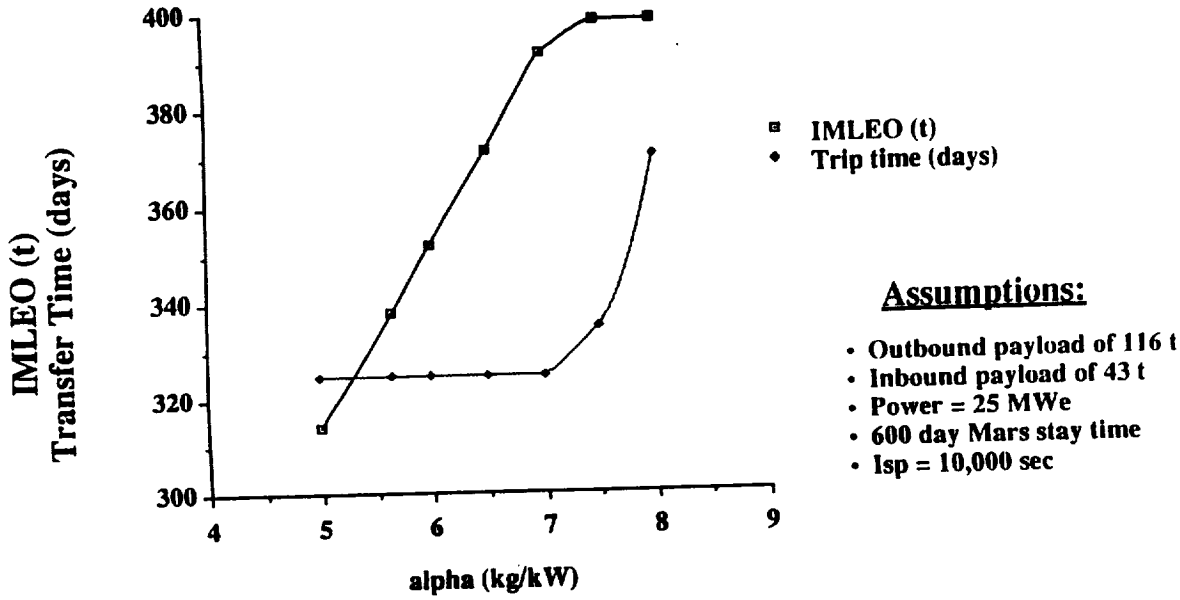
For conjunction class missions, the vehicle is basically performing the "easy" leg of two opposition class missions. The only restraint of this type of mission is that the crew must remain at Mars for approximately 600 days. Isp for conjunction class missions tends to optimize at 5,000 to 7,500 sec, as in the opposition missions. Power levels for conjunction class missions can be in the 10 MWe range for trip times in the 400 to 450 day range. Alpha for this analysis was varied from 8 to 12 kg/kWe. Power levels down to 5 MWe can be used that support 500 day trips at an Isp of 5,000 sec and alpha of 8 kg/kWe. Higher Isp's at 5 MWe result in such low thrust that the vehicle must lengthen its trip time to well beyond the Hohmann transfer trip time simply to allow the thrusters enough time to generate the required delta V to complete the transfer. These results are for a nominal 2016 mission.

Using the baseline vehicle (10 MWe, alpha=10 kg/kWe, Isp=6500 sec) and trajectory, a trade was performed in which the year of opportunity was varied through the entire Earth-Mars synodical cycle. When arrival and departure from Mars occurs near the apoapsis of the Martian orbit, Mars is further away from Earth and is traveling slower. Both of these factors require a corresponding increase in necessary total delta V for the same trajectory geometry. As a result, longer trip times and higher initial masses in LEO are required for some of the conjunction opportunities and not for others. The SEP vehicles cannot make up for higher energy requirements by increasing the delta V (the propellant mass available), so shortening the stay time is used as a way of maintaining relatively efficient paths on the "more difficult" opportunities (Reference 9).

3.2.2.5 Nuclear Electric Propulsion

The trajectories for NEP vehicles are very much the same as SEP trajectories. The difference between a NEP vehicle and SEP vehicle lies in the power profile over the trajectory. While the SEP vehicle's power generation deteriorates to roughly 50% of that generated at 1 AU, the NEP power generation remains constant throughout the entire trajectory. Another distinct difference in the two systems is that while the SEP vehicle remains fairly constant if power levels increase, the NEP vehicle alpha's can vary at certain power levels, Figure 3-39. For instance the vehicle alpha for a SP-100 powered 100 kWe vehicle would be in the neighborhood of 40 kg/kWe. A typical vehicle alpha for a 20 to 40

MWe vehicle would be in the neighborhood of 5 to 12 kg/kWe, depending on the technology assumed. There is a point where the vehicle alpha's start leveling off and becomes fairly constant. This power level is about 20 to 25 MWe. This power level will usually give trip times and initial masses that are in the "knee" of the IMLEO vs trip time curves. However, if faster trip times are required, power levels can be increased to obtain faster trips at the cost of initial mass (power system mass and propellant mass).



NEP CONJUNCTION ALPHA TRADE
FIGURE 3-39

As mentioned previously, the SEP specific impulses tended to optimize about 6500 seconds while the NEP Isp's optimized higher. For most of the cases, an Isp of 10,000 seconds was found to be optimum for reasonable trip times and initial masses. Unless otherwise specified, all Isp's for the NEP vehicles may be assumed to be 10,000 seconds. The only cases that did not optimize at 10,000 seconds were the lower power level cases (10 MWe and below) whose trajectory required more thrust than could be provided at that Isp.

The initial parametric analysis for opposition class missions was performed by Tucker to define a starting point for our vehicle point design (Reference 26). The initial power levels and associated vehicle alpha's were as follows; 10 MWe at 12 kg/kWe, 24 MWe at 6 kg/kWe, 40 MWe at 4 kg/kWe, 80 MWe at 4 kg/kWe, and 120 MWe at 3 kg/kWe. This analysis revealed several important conclusions. When the power level of 40 MWe was

doubled to 80 MWe while holding alpha constant, the IMLEO increased significantly while the trip time held virtually constant. More power is required to push more mass. Power levels of 10 MWe and below will not give trip times much faster that 600 days. Trip times in the range of 500 to 600 days are possible with power levels of 20 to 25 MWe. The power level of 40 MWe and associated alpha of 4 kg/kWe revealed the best solution in terms of IMLEO and trip times. From this analysis, 40 MWe was chosen as the power level for our vehicle concept. After a detailed vehicle concept was developed our final vehicle alpha was calculated to be 6.8 kg/kWe. This 40 MWe vehicle had a trip time of 490 days and 561 tonne IMLEO for the nominal 2016 opposition mission. Further analysis was also performed to determine the effects of varying mission opportunities (Reference 10).

Trade studies were performed for the proposed conjunction class manned Mars mission, in which Mars residence times are on the order of several hundred days. Long stay times at Mars allow the vehicle to travel on two relatively low energy legs, in contrast to the opposition class missions which generally utilize one low energy leg and one high energy leg. As a result, initial masses in Earth orbit are significantly lower than in opposition class missions. Trades were performed for optimum power levels, optimum specific impulse for a given power level, and optimum launch and encounter dates for the 2016 opportunity. Using a baseline case of 25 MWe, Isp = 10,000 seconds, the total vehicle alpha was varied to determine sensitivity of the initial mass to the design alpha. Trajectories were then generated for each of the opportunities between the years 2009 and 2026 , again using the 25 MW vehicle as a benchmark. For the best and worst of the opportunities (as far as minimum weight and minimum time), the stay time was varied between four hundred and six hundred days in order to determine and optimal Mars residence time. The study revealed that for the nominal 2016 conjunction opportunity a 25 MW vehicle can perform a 325 day trip time for 391 tonnes of IMLEO at an Isp of 10,000 seconds and vehicle alpha of 7.0 kg/kWe.

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4. TRADES AND SENSITIVITIES

4.1 Introduction

A series of trades and sensitivities were performed to examine, characterize and compare different alternatives and key parameters in order to thoroughly understand all the options available to carry out the new initiative missions. The results of these studies are provided in this section. Further details may be found in the six volumes of the Implementation Plan and Element Description Documents (IP&ED) that contain parametric data and results primarily in graphical form (References 6-11). These data present the interrelationships existing among the parameters that define the mission, the vehicle, the propulsion system, and the performance of the overall vehicles for interplanetary missions.

It is important to mention that, for a series of transportation trades covering as broad a scope as that encompassed by the Space Exploration Initiatives, the final judgements concerning winning concepts are closely tied to which individual mission goals, requirements, and technical assumptions are given the most weight in the evaluation process.

4.1.1 Mission Goals, Requirements and the Evaluation Process

A SEI mission objective, or mission model, is actually a composite entity composed of individual requirements and mission goals. Examples of individual mission goals are an "early" first flight date, or a "short" transfer trip time, or a LEV design emphasizing "ease" of cargo off-loading.

Requirements, on the other hand, differ from goals in that they are tied to specific, clearly identifiable items or time tables, such as a "2010" first flight date, a "400 day" trip time, or an acceleration limit for aerobraking. The difference between the two is a matter of preciseness. Some mission requirements were provided early in the study, requirements were periodically reviewed to ensure they did not artificially restrict trades. Many requirements applicable to space transportation mission and vehicles were derived as the study progressed.

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4.1.2 Preferred Concepts

A single overall winning vehicle concept cannot be given as an outcome of advanced propulsion trades at the present time. SEI does not presently have a definitive mission model, encompassing a specific flight trajectory, development timetable, payload manifest, etc. A more complete understanding of the capabilities and complexities of in-space transportation systems was sought by these trades. The early phase of (spacecraft) conceptual design is often an iterative process, where both mission requirements definition and vehicle design evaluations proceed together to the next level of refinement. The subjective nature of some important criteria prevent an unequivocal statement of a winner.

Identifying areas where particular concepts are differentiated as being superior is emphasized in this section. For certain categories of mission scenarios preferred concepts will be emphasized.

4.1.3 Presentation of Trade Results

The following principles were applied to trade study definitions.

a. Trades capture a sufficiently large trade space.

b. Trades address important questions related to SEI transportation.

c. Trade comparisons are as fair as possible

d. Trade evaluations are cognizant of a system's future development capability for evolution to, or potential of, improved performance in the out years.

e. Trades include adequate documentation to enable full traceability.

f. Trades are presented with a common indicator or gage as a way of "normalizing" and/or comparing the differing impacts of several unrelated systems; example: normalizing the cost of incorporating artificial g, impact of shortening trip time, and of increasing outbound payload by presenting all results in terms of vehicle Initial Mass In Earth Orbit (IMLEO).

Evaluating all of the complex interactions between propulsion system, mission and vehicle parameters is impossible; the goal of the trades and sensitivities task is to identify and understand those found to be most pertinent to the SEI missions.

The order of presentation is as follows:

- Reference vehicle designs trades
- Overall propulsion system comparison trades
- Overall vehicle system trades
- MEV trades
- Lunar vehicle trades

4.2 Reference System Trades, Baseline Evaluations

The reference system trades contain the baseline vehicle evaluations and sensitivities.

4.2.1 Introduction

A major portion of this trade area was to evaluate the performance characteristics of five candidate interplanetary propulsion systems:

Cryogenic all-propulsive (CAP),	Nuclear electric propulsion (NEP)
Cryo/aerobraking (CAB),	Solar electric propulsion (SEP)
Solid core nuclear thermal rocket (NTR)	

These reference vehicle designs were introduced in earlier sections, and represent integrated mission/vehicle/engine systems. Several trades were performed early in the study and several key assumptions were made at that time from which these designs were based. Subsequent trades and sensitivity analyses were done to evaluate system or subsystem options to these baselines. Frequently, results of these subsequent trades are presented as delta's to the reference vehicles IMLEO, performance or operational characteristics. The intent is that the reference vehicles then become a representative 'yardstick' by which other approaches are compared, without implying that these baselines represented the optimum designs.

The 2016 opposition mission detailed earlier was utilized as the reference for all the candidate propulsion system/vehicle concepts, except for the CAP, which could only accommodate conjunction-class trajectories. Mission delta V's and departure dates for this mission are given in Reference 6.

All vehicle concepts carried the same payload; identical 35 ton transfer habitat modules for the baseline crew of 4, an ECCV and a MEV. The MEV carried by the CAB did its own separate aerocapture at Mars; its total weight of 84 tons was 11 tons heavier than the MEV carried by the other vehicle concepts utilizing propulsive capture at Mars. In those cases the MEV's had a descent-only aeroshell. For the purpose of these trades, mission payloads such as the habitat were not changed to reflect habitat design changes. This permits all trade results to be directly compared.

4.2.2. Results of Reference Vehicle Design Trades

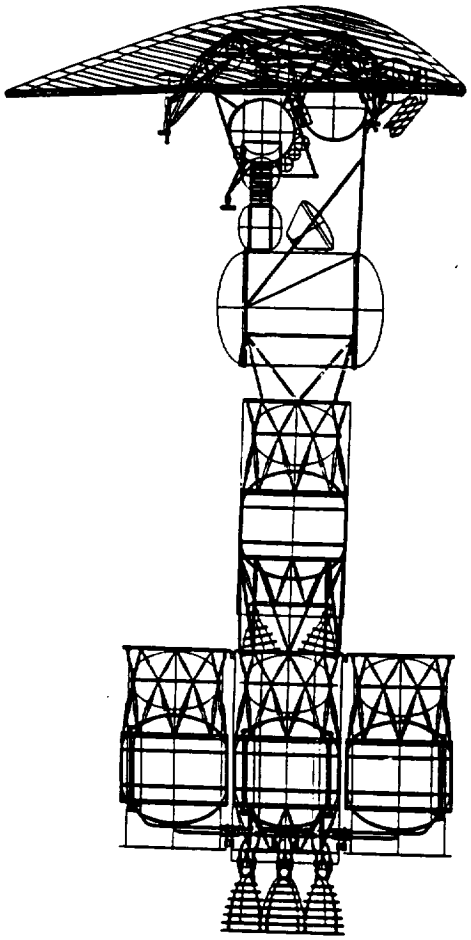
The following summaries present the significant trade results involved in formulating the reference vehicle designs. Although not exclusively propulsion related, a majority of the analysis falls into that category.

4.2.2.1. CAP Baseline Trades

Performance Limits - Exclusive use of the 475 sec Isp O2/H2 propulsion system eliminates the CAP propulsion system from consideration as a viable contender except for the low delta V conjunction class trajectories. For opposition-class trajectories, the CAP option has unreasonably high IMLEO's. CAP gains preference over the more complex CAB system, only when minimum-energy conjunction class missions are considered. The advantage of aerocapture at Mars is minimal given the much slower arrival Vhp for these trajectories.

Reference Vehicle Development Trades - The CAP baseline shown in Figure 4-1 was derived directly from the CAB baseline described below with a minimum of trade effort. A 90% propellant fraction was used for the TMI stage, as well as four 200k lbf advanced space engines, each with an Isp of 475 sec and with a nozzle expansion ratio of 400:1. A single TMI burn was utilized for Earth departure. The CAP system represents the propulsion system that could be brought to first flight readiness the earliest, and with the least developmental cost. All major propulsion hardware excepting the large 200k lbf class

TMI engines could come directly from a Lunar program. This early operational advantage is countered, however, by the fact that the 900 day plus trip times characteristic of these conjunction class missions would require more development effort in the areas of MTV habitat ECLSS systems, as well as incorporation of artificial-g systems.



Element	*mass (kg)*
MTV crew hab module 'dry'	28602
MTV hab consumables & resupply	11655
MTV hab mod science	1000
MTV crew habitat module	**41257**
TEI usable propellant (dV=860 m/s)	12959
TEI outbound boiloff	1299
TEI inorbit boiloff	2423
Total TEI propellant	**16681**
MOC usable propellant (dV=1530 m/s)	59340
MOC outbound boiloff	3310
Total MOC propellant	**62650**
EOC propellant (expendable veh)	n/a
RCS propellant (dV=30 m/s)	702
Outb midc correction prop (dV=40 m/s)	1872
Inb midcourse corr prop (dV=40 m/s)	546
MOC/TEI propul stg inert	14610
MTV propulsion stage total	**97061**
MEV descent only aerobrake	7000
MEV ascent stage	22464
MEV descent stage	18659
MEV surface cargo	25000
MEV total	**73118**
ECCV	7000
TMI inert stage wt	37800
TMI propellant load (dV=3900 m/s)	335650
TMI stage total	**373450**
IMLEO	**591886**

2016 CONJUNCTION CAP VEHICLE

FIGURE 4-1

Reusability - The motivation for reusability is primarily the desire for recovery of the expensive MTV crew habitat module. CAP reusability for conjunction class only missions is not costly in terms of a delta to total IMLEO (typically less than 10%, Reference 6). The added propellant necessary for the Earth orbit capture (EOC) burn is small, because of the low Earth arrival velocities characteristic of these conjunction class trajectories. However, this is equally true for all the high thrust propulsion systems utilized for these low delta V conjunction missions. Our baseline CAP was reusable.

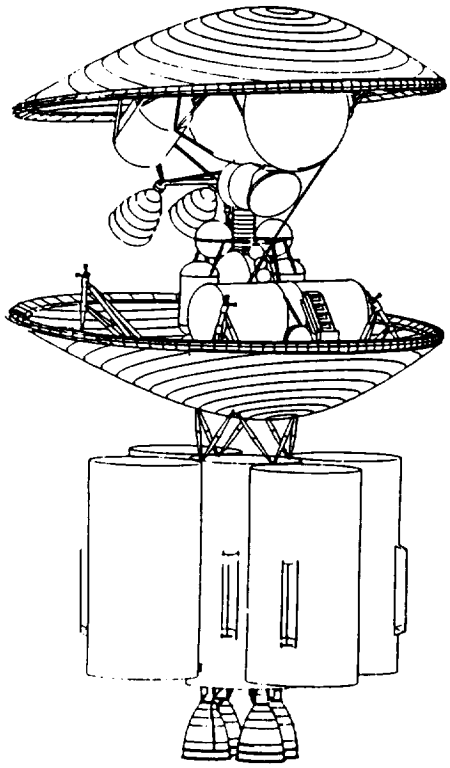
Potential for Performance Improvement - The CAP's potential for performance improvement is unlikely, because its 475 sec Isp represents the practical maximum for cryogenic propulsion systems.

Vehicle Flexibility to Accept Variations in Performance Requirements - The CAP does not adapt well to changes, other than accommodating payload variations within the conjunction class missions.

4.2.2.2. CAB Baseline Development Trades

Performance - The CAB served initially as the overall reference vehicle for this study. The 801 ton, 2016 opposition (Level II 565 day reference trajectory) vehicle required 576 tons of propellant and 73 tons of total propulsion system inert weight for the mission. The reference CAB vehicle sketch and summary weight statement is given in Figure 4-2. Additional details on this reference vehicle are given in Reference 6.

Vehicle Performance Assumptions - The CAB's Mars capture aerobrake was set to 15% of the brake's Mars capture mass. The TMI stage propellant fraction was 90%. A single TMI burn was utilized for Earth departure. Vehicle T/W at TMI was set at 0.4 to reduce g-losses. Thrust-to-weight ratio is defined as thrust in lbf divided by initial mass in lbm. The numerically equivalent SI measure is (thrust in Newtons)/(9.80665 * mass in kg.). For the 2016 reference mission the vehicle required four 200k lbf engines (Isp = 475 sec with a 400:1 nozzle).

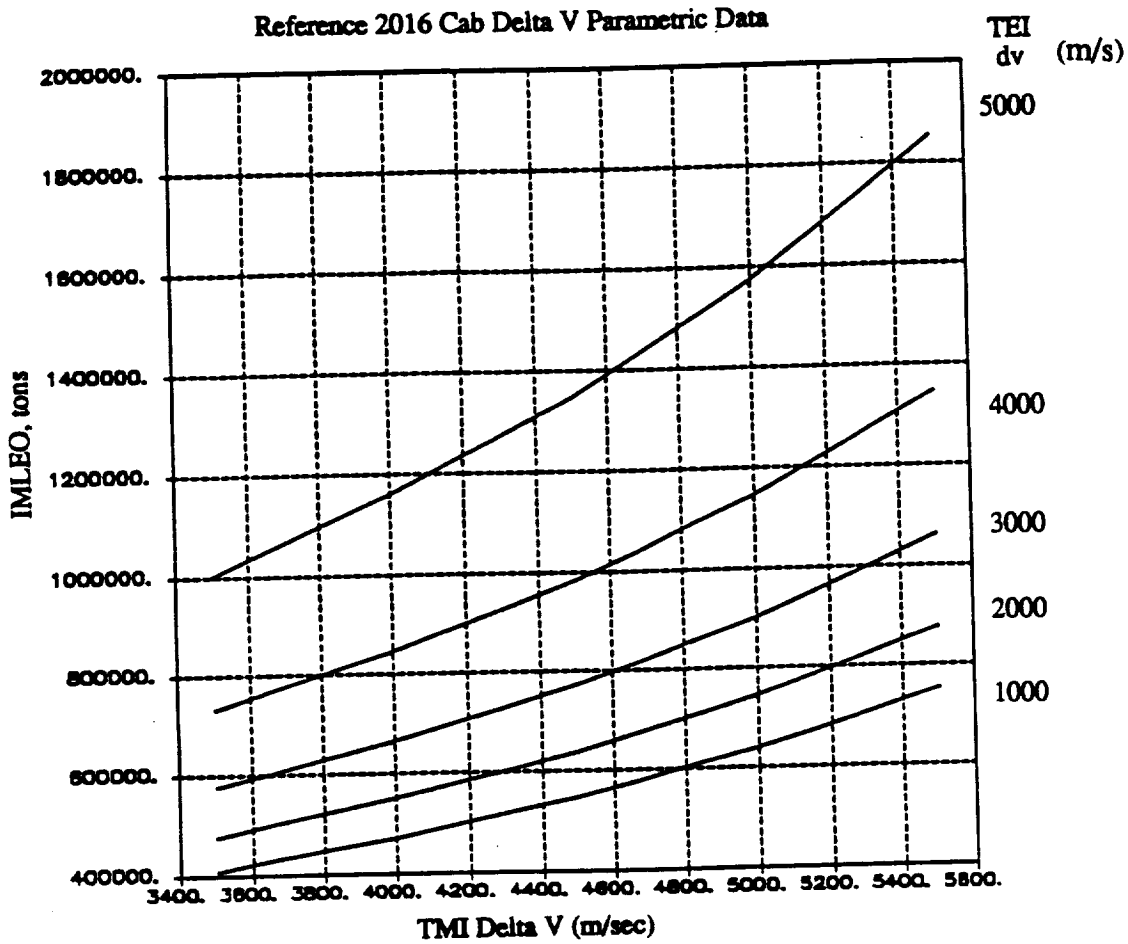


Element	*mass (kg)*
MTV Mars aerobrake	23758
MTV crew hab module 'dry'	28531
MTV consumables & resupply	7096
MTV science	1000
MTV propulsion stage	18206
MTV propellant load (dV=3400 m/s)	85141
MTV total	**163732**
MEVMars capture & desc aerobrake	15138
MEV ascent stage	22754
MEV descent stage	21457
MEV surface cargo	25000
MEV total	**84349**
ECCV	**7000**
Cargo to Mars orbit only	0
MTV-TMI interstage wt	500
TMI inert stage wt	54560
TMI propellant load (dV=4281 m/s)	490950
TMI stage total	**545510**
IMLEO	**801090**

2016 REFERENCE CRYOGENIC/AEROBRAKE VEHICLE
FIGURE 4-2

Reference Vehicle Development Trades - An important early trade of the CAB system involved the strategy for the aerocapture maneuver at Mars. Two options were evaluated in respect to the size and number of aerobrakes: (1) aerocapture of both the MTV and MEV stages together in one aeroshell; (2) two separate, but structurally similar aeroshells, for separate aerocapture of the two stages. Results favored having the MEV and MTV stages capture separately; the single aeroshell grew very large and heavy, complicating launch manifesting and in orbit assembly, and required a small inflight reconfiguration maneuver to position the MEV behind the MTV so as to be within the brake wake cone protected region and for correct CG placement for the capture maneuver. The separate capture approach allowed for the benefit of autonomous MEV capture one day ahead of the manned MTV, as a valuable 'flight test' precursor to the follow on capture of the manned MTV stage.

The 15% aerobrake reference CAB vehicle IMLEO varies with TMI delta V and TEI delta V as shown in Figure 4-3.



REFERENCE CAB TMI AND TEI DELTA V
FIGURE 4-3

Reusability - Recovering the MTV crew module required reuse of the Mars capture aeroshell at Earth. The high arrival velocity encountered at Earth on return was found to produce a maximum heating rate of 860 watts/cm2. This was shown to be beyond the near term reradiative capabilities of the aerobrake thermal protection system (TPS). For CAB reuse, a complex hybrid TPS would be required, operating reradiatively at Mars and as an ablator at Earth. The "Boeing Vehicle Synthesis Model", an inhouse computer program from which the detailed vehicle weight statements are calculated, was run to evaluate what the IMLEO cost to the 2016 reference mission would be, if the recovery of the MTV stage could be done with a 15% Mars aeroshell. The results for the analysis indicated that IMLEO increased 82 tons (10%) from 801 to 883 tons with the increase due primarily to the added TEI propellant necessary to boost the aeroshell back to Earth. The increase in

IMLEO for recovering the crew module is very sensitive to mission profile particulars, especially trans-earth injection delta V.

Issues of Concern - The principal areas of concern included verification of aerobrake integrity, successful sealing of aeroshell assembly joints and surfaces, Mars aerocapture GN&C, and tether system design and weight considerations for accommodating artifical-g.

Potential for Performance Improvement - The CAB's potential for performance improvement is unlikely; its 475 sec Isp represents the practical maximum for cryogenic chemical propulsion systems. Mars capture aeroshell weights of 15% of capture weight represent a near minimum based on current analysis.

CAB Vehicle Flexibility to Accept Variations in Performance Requirements - The 2016 reference CAB vehicle does not adapt well to changes in the design of Venus swingby mission profiles now defined, for the following reasons:

- Large variations in return propellant (TEI plus deep-space burn) result in large variations in aerobrake payload. The brake must be designed for the worst case.
- It is much easier to accommodate variations in TMI propellant as the TMI stage is modularized to match ETO lift capability.

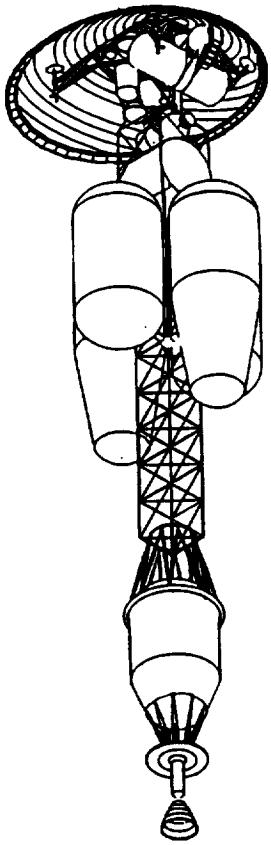
The CAB baseline adapts well to the conjunction fast transfer profile because the delta Vs and Mars arrival velocity are similar to the opposition profile.

4.2.2.3. NTR Baseline Development Trades

Initial nuclear rocket development was a joint Atomic Energy Commision-USAF project that started with exploratory research in 1953 bearing the name 'ROVER'. Soon to become an AEC-NASA program, early research was transformed into test hardware in July 1959 with the KIWI A test reactor (100 MWt, 0 lbf thrust) and peaking with the Phoebus 2A test reactor in 1967 (5000 MWt, 250k lbf thrust, 850 s Isp). In 1961, development began on the NERVA (Nuclear Engine for Rocket Vehicle Application) fight configuration series of full-up engines. Before termination of the NTR program in 1973, record performances of 62 minutes at continuous full power (NRX-A6), peak fuel temp 2750 K (PEWEE), peak

fuel power density 5200 MW/m3 (PEWEE) and 28 single engine restarts (XE) capability, among other achievements of an extensive test program, were seen at the Jackass Flats, Nevada test range. A total of 20 reactors were designed, built and tested between 1955 and 1973 at a cost of approximately $1.4 billion before support for the program ended. Post-Apollo plans for manned expeditions to the planets were abandoned due both to major cuts in NASA's budget and its transition of focus to development of a space shuttle. In the manned Mars mission plans of the 1960's NTR propulsion was the system of choice.

Performance - The NERVA NTR was involved in multiple mission applicability and payload variation trades. The 735 ton, reusable, 2016 opposition (Boeing #2 trajectory - 434 day trip time) vehicle required 525 t of propellant and 92 tons of propulsion system inert weight. A summary weight statement is given in Figure 4-4. A detailed weight statement with rationale is given in Reference 6.



Element	*NERVA*	*Advanced NTR*
MEV desc aerobrake	7000	7000
MEV ascent stage	22464	22464
MEV descent stage	18659	18659
MEV surface cargo	25000	25000
MEV total	**73118**	**73118**
MTV crew hab module 'dry'	28531	28531
MTV consumables & resupply	5408	5408
MTV science	1000	1000
MTV crew habitat system tot	**34939**	**34939**
MTV frame, struts & RCS inert wt	5593	5270
Reactor/engine weight	9684	1701
Radiation shadow shield weight	4500	4500
EOC propellant (dV=2654 m/s)	27756	19675
TEI propellant (dV=3900 m/s)	59245	40457
EOC/TEI common tank wt (1)	13845	10344
MOC propellant (dV=3870 m/s)	151680	107190
MOC tanks (2)	25572	19835
TMI propellant (dV=4201 m/s)	286146	188530
TMI tanks (2)	43092	30401
ECCV	0	0
Cargo to Mars orbit only	0	0
IMLEO	**735190**	**535960**

2016 REFERENCE NERVA NTR AND ADVANCED NTR VEHICLES

FIGURE 4-4

Performance Assumptions Propulsion Related - A NERVA derivative reactor design was selected as the baseline NTR propulsion system, and assigned a performance level of 925 sec Isp by NASA early in the study. The primary alternative to this NERVA design, was the Particle Bed Reactor concept (PBR). The advanced NTR is characterized by an Isp of 1050 sec and an engine T/W of 20 which is characteristic of the PBR concept. This performance represents the maximum potential of solid core reactor technology as applied to direct thermal propulsion, see Reference 8.

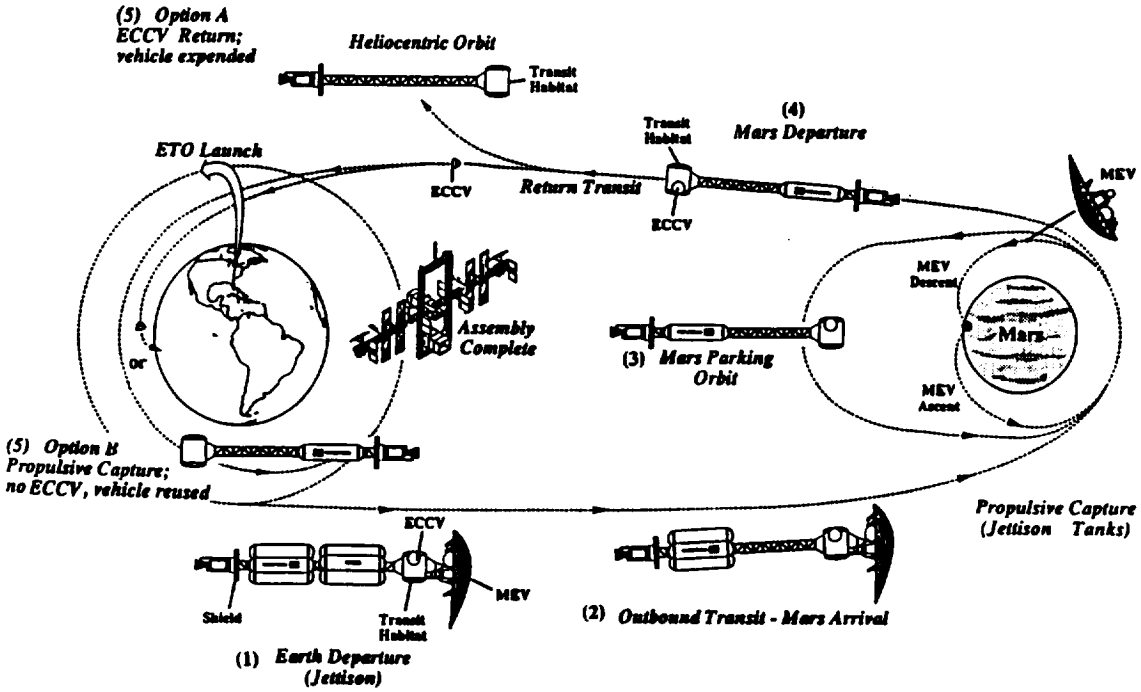
The 925 sec Isp of the baseline propulsion system is a 75 sec improvement over the maximum Isp realized in the 60's NERVA program (850 sec Isp at 250k lbf thrust, attained by Phoebus 2A in 1967. The value of 850 sec for Isp is an extrapolation based on the use of a large (500:1) expansion ratio nozzle instead of the small 40:1 nozzle actually used in the test. The Isp of 925 secs corresponds to a H2 exit gas temperature of 2700 K and would require an upgrade to composite material fuel elements. Such an improvement would not require a reactor redesign, and a ground test qualified engine development program within the near term (8-12 years), would be considered as realistic by a large majority of the nuclear propulsion community. The probable difficulties in this activity would be full-up engine testing and nuclear safety issues.

One important emphasis of the early stage of the study was the technical validation of the performance gain that could be expected for 1995-2015 technology NERVA derivatives. Recent technology advances since the early 70's in the areas of high temperature capable materials, coatings and fabrication techniques would be incorporated into any new development program, with the goal of evolving eventually to higher gas temperatures (3100K), and to the higher performance (1000+ sec Isp) associated with the advanced NTR concept.

The NERVA derivative engine thrust level was also set by NASA at 75k lbf (3.5 engine T/W, 9864 kg engine weight) early in the study. The reference vehicle utilizes a single engine. In the early portion of this study it was decided that the relatively high cost and weight of a reactor precluded the used of multiple engines. Later, that issue was reopened. Other propulsion related assumptions included setting the reactor cooldown propellant to 3% of the usable propellant for each burn, setting the propellant reserves to 2%, and assessing boiloff penalties with the inhouse Boeing "Crystore" cryogenic thermal model 2016 Reference NTR Mission Profile. Once assembled in LEO the 2016 NTR-powered Mars vehicle departs Earth utilizing a 2 or 3 burn TMI departure with its single 75k lbf

thrust NERVA derivative engine and begins its 434 day journey which includes an inbound unpowered Venus swingby (Figure 4-5). The vehicle payload consists of a 73 t MEV and a 4 man, 35 t MTV crew hab module. A single Earth departure burn would incur sizable g-losses due to the low overall T/W ratio (vehicle T/W=0.04). Splitting the departure burn into 2 or 3 phases and firing each time near the orbit periapsis point is to be used to decrease these g-losses losses to a more acceptable level. After TMI, the empty Earth departure propellant tanks are jettisoned as a means of lightening the vehicle for subsequent burns. Orbit capture at Mars is done all propulsively, and as before, propellant tanks are jettisoned. The crew enters the MEV and descends to the surface. After a 30-day surface stay the crew returns to the transfer vehicle via the MEV ascent stage. Once onboard, the transfer vehicle does a single TEI burn and begins the inbound journey with only the MTV crew module as payload. After the Venus swingby and an inbound midcourse correction burn the vehicle will return to Earth in one of two ways, as return option 5a or 5b. For the vehicle expendable mode, the crew enters a small Apollo type ECCV (7 tons) which enters the atmosphere via heat shield aerobraking to achieve eventual splashdown in the Pacific, (option 5a). In the vehicle reuse mode, an all propulsive Earth capture burn is done to capture into an elliptical orbit (option 5b). No ECCV was assumed for this reuse mode. Subsequent flight mechanics analyses showed that an ECCV is needed since the elliptic Earth capture orbit is normally not immediately accessible from the LEO node.

Recovering the MTV stage back at Earth requires a significant delta V EOC burn for capture, reduced somewhat by the selection of a high energy elliptical return orbit (500 km by 24 hr period) compared to a return down to SSF LEO circular. The NTR is refueled by an LTV in the 24-hour orbit for later return to the LEO Node orbit. The amount of hydrogen required is typically about 30 tons. This is more efficient than carrying this hydrogen on the mission profile. The 24-hour phasing orbit permits alignment of the captive orbit and LEO-node orbit lines of nodes without delta V penalty. The subject of end of life reactor disposal was not evaluated for the baseline definition.



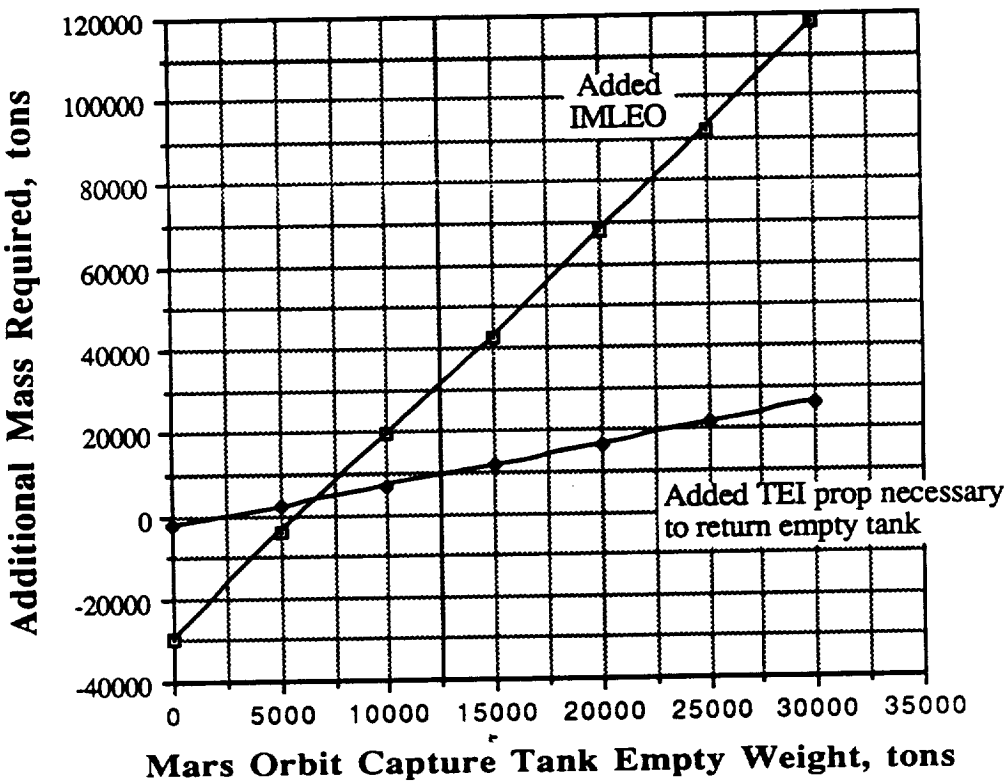
2016 REFERENCE NTR MISSION PROFILE
FIGURE 4-5

Reference Vehicle Development Trades - Operational and design assumptions used in this analysis included 1) the jettison of the TMI and MOC tanks after burns, 2) the use of as large a tank size as the launch vehicle(s) can deliver (in this study 10 m diameter tanks were used, driven by a reference launch vehicle shroud size of 10 m by 30 m), and 3) the use of advanced materials such as metal matrix composites for the H2 tanks in order to keep the tank fraction as low as possible.

Truss/Tank Staging Trade - An early NTR design trade was the truss/tank staging trade, done to determine the impact of keeping or dropping the large Mars Orbit Capture (MOC) H2 tank(s). Two options were evaluated, with and without a truss system. The truss system was used in the baseline configuration with a 4.5 ton reactor radiator shadow shield positioned in-between the reactor and the aft tank. The 2 MOC tanks were jettisoned immediately after the burn. The truss served to connect the engine/aft tank to the habitat module. The estimate of 2.4 tons for the truss reflected the use of standard SSF truss bays at 160 kg per complete 5 m by 5 m bay. The two MOC tanks together weighted 26 tons empty (14% tank fraction).

For the vehicle without a truss, the truss system that structurally links the engine to the habitat module has been replaced by using a single MOC tank as a connecting, load carrying structure, much like designers did for NERVA era designs (Reference 8). This tank is not dropped but must be carried back to Earth since its serves as a structural element. This single MOC tank (25 tons) must also be kept pressurized for the inbound leg.

The results indicated that NTR configurations that utilize either the TEI or MOC propellant tanks as structural members suffer a significant weight penalty.The large H2 tanks typical for the opposition class vehicles, covered with MLI, vapor cooled shields and meteor shields, have tank fractions estimated at 14% or more (Shuttle ET tank is 6%). For the 2016 reference mission MOC burn, a 25 ton tank is required. The 2.4 ton SSF type truss is approximately one tenth the mass of this empty MOC tank. Carrying the 25 ton MOC tank back on the inbound leg increased IMLEO by 95 tons over the 735 ton reference; 22 tons of this additional 95 tons is the added TEI propellant required to boost the empty tank back to Earth, Figure 4-6.



NTR TRUSS VS TANK TRADE DATA
FIGURE 4-6

Vehicle Tank Placement and Configuration Trade - The reactor/engine/shield combination greatly influences the overall physical configuration of the NTR vehicle. The necessity of radiation attenuation between the radiation source and the crew, as well as the placement and staging of very large H2 tanks are two major considerations that are unique to NTR vehicles. The reference design was configured to minimize exposure to the crew from reactor neutron and gamma radiation. The two factors of separation distance and axial alignment of tanks are applicable in this regard. (Note: natural radiation environment issues are not included in this section.)

The separation distance between the crew and reactor is a key factor in reducing the amount of radiation that actually reaches the crew module. Since the received dosage is equal to the inverse of the separation distance squared, grouping the lengthy H2 tanks into an axial alignment rather than a radial cluster near the engine improves radiation attenuation by increasing the separation distance provided by the tankage/structure without unduly penalizing the vehicle with the weight from structure dedicated solely to providing separation distance. Doubling the separation distance to reduce the received dosage by a factor of four is an example of how sensitive the inverse square law is to separation distance .

The forward positioning of drop tanks rather than radial clustering also allows the reactor radiation shadow shield protected cone half angle to be smaller. Radial clustering would increase the immediate projected tank area around the reactor that would scatter direct radiation and thus aggravate the problem by introducing secondary sources. Forward positioning also provides more H2 to be utilized (as a secondary thermal neutron shield) in the direct line between the crew cab and reactor. It is beneficial from a shielding viewpoint to keep the earth orbit capture (EOC) and MOC propellant in an 'inline' tank just behind the reactor shield.

Isp and Engine T/W Trades - Vehicle IMLEO variation with Isp and engine T/W is given in Reference 8 .

Reusability - The IMLEO cost to the 2016 reference mission for reuse is 170 tons above that of the expendable mode (735 vs 565 tons). The 170 tons consists of both the propellant to capture at Earth and the propellant to push that earth orbit capture H2 load through the TMI, MOC and TEI delta V's.

The selection of the reusable mode for reference was based primarily on LCC considerations. Recovering the expensive reactor/shield/truss structure and crew module system was preferable to the alternative up front benefit of IMLEO reduction gained by expending all but the ECCV for crew return. The 1960's NERVA program demonstrated reactor reusability, so NTR propulsion systems are considered completely reusable within the lifetime of the engine(s). NERVA derivatives for Mars transportation are estimated as capable of at least 10 hrs total burn time. Ten hours is roughly equivalent to the cumulative burn time of three opposition class missions, or 8 - 10 Lunar NTR missions.

Nuclear Propulsion Unique Issues and Other Issues of Concern - The following issues are of particular concern to the NTR vehicle.

- Reactor radiation shielding.
- Allowable radiation energy flux which can be incident on the H2 in the propellant tanks.
- Residual H2 capability as secondary radiation shield.
- Multi-Earth departure burns to reduce g-loss for moderate vehicle T/W ratios.
- Post burn reactor 'cooldown' requirements
- Engine testing, control and cleanup of residual radiation in the engine effluent.
- Very large H2 propellant tanks/launch vehicle integration.

Potential for Performance Improvement - Realistic growth potential to 1000+ secs Isp systems may be possible, but might be considered overly optimistic by some experts. The NERVA derivative 925 sec Isp represents an advancement over the 1960 NERVA performance level due primarily to higher temperature capable reactor fuel element materials. The Isp value of 925 sec corresponding to the 2700K exhaust gas temperature obtained from composite material fuel elements is 200K higher than demonstrated NERVA temperatures (2500K) utilizing graphite material fuel elements. An Isp of 1000+ sec may be attained utilizing carbide material elements capable of 3100K. An engine T/W increase to 20 for the PBR concept is mainly a factor of reactor internal geometry.

NTR Vehicle Flexibility to accept variations in performance requirements - The NTR vehicle readily adapts to changes in its mission for the following reasons.

- Identical engine, core (aft) tank and truss are used for all mission types
- Changes to IMLEO are implemented simply as changes in propellant tank capacities.
- Propellant capacity variations implemented in expendable tanks
- Truss must be long enough to accommodate worst case, this is a minor impact.
- Variations in thrust-to-weight handled by multi-burn Earth departures as necessary

4.2.2.4. NEP Baseline Development Trades

The NEP vehicle uses a nuclear reactor to produce thermal energy. The thermal energy is converted to suitable electrical power for propulsion means. This electrical power is then used to ionize a propellant and accelerate the charged particles at extremely high velocities. The expulsion of the ionized propellant at extremely high velocities produces a low level thrust that accelerates the vehicle. Due to the low level of thrust, the NEP vehicle is accelerating for almost one-half of the trip and decelerating for most of the last-half of journey. This propulsion philosophy is unlike the high-thrust systems that operate for only a tiny fraction of the trajectory. Although the NEP thrusts for most of the trajectory, the vehicle consumes much less propellant than a high-thrust system due to the efficiency attained by the very high Isp. The major portion of the NEP vehicle is composed of the powerplant necessary to produce the required electrical power for propulsion.

Propulsion performance and assumptions: Power Level - A parametric mission analysis study was performed to establish a baseline power level for a NEP point design. Further trades were also performed to determine the effects of vehicle alpha, Isp, trip time, and opportunity. Results of this analysis and the relevant assumptions can be found in Section 3 and Reference 10. The propulsion system is composed of roughly 1 engine per megawatt electric (MWe) of required power. The reference vehicle is designed at 40 MWe with several spare engines.

Operational Considerations - The major operational consideration for NEP is the operation scenarios for departure and refurbishment. From a technical standpoint a reactor can operate safely within a LEO infrastructure. Currently it has not been decided if reactors will be allowed to operate in LEO or what the minimum altitude will be. Once these requirements have been established the NEP near Earth orbit operations scenarios can be finalized. Details of the operational mode are found in Reference 10.

Power System Trade - The heart of the power system is a fast spectrum lithium cooled reactor. A potassium Rankine power conversion system was chosen early due to its lower system mass when compared to the other conversion alternatives. The Rankine system is less massive than other dynamic systems due to the smaller radiator required. Dynamic system efficiencies are also higher than static conversion systems, resulting in less thermal generation required per unit of electrical energy produced.

Propulsion System: Ion vs MPD Thrusters - The propulsion system is composed of many ion thrusters that use argon as the propellant. There are several types of electric thrusters such as arcjets, resistojets, MPD trusters, and ion thrusters. Each of these thrusters are suited for a particular application. For a Mars mission application, MPD and ion thrusters are the only logical choices. MPD thrusters have a lower efficiency and a lower expected lifetime when compared to ion thrusters. Due to these reasons the ion thruster was chosen for the baseline propulsion system. Future developments in MPD thrusters may improve their efficiency to the competitive range.

There are many propellants that can be used with an ion thruster. A good propellant is one that ionizes easily and has a high molecular weight. Mercury is a good propellant from a performance standpoint and has been used in previous applications. However, mercury is toxic and poses serious implications when tested on Earth. Previous experiences with clean-up of test facilities has discouraged the use of mercury in future applications. Other possible propellants include argon, cesium, krypton, and xenon. The performance of xenon is similar to mercury but there is a limited supply of xenon and it is very expensive. Argon performance is acceptable and there are no issues of abundance or cost. Therefore, argon was chosen as the propulsion system propellant.

4.2.2.5 SEP Baseline Development Trades

The SEP vehicle uses a large solar array to produce power from the sun's energy. The large solar array is composed of small solar cells (4-16 cm^2) that turn the sunlight into electrical power. The electrical power is then used to ionize a propellant and accelerate the charged particles at extremely high velocities. The expulsion of the ionized propellant at extremely high velocities produces a low level thrust that accelerates the vehicle. Due to the low level of thrust, the SEP vehicle is accelerating for almost one-half of the trip and decelerating for most of the last-half of the journey. This propulsion philosophy is unlike the high-thrust systems that operate for only a tiny fraction of the trajectory. Although the

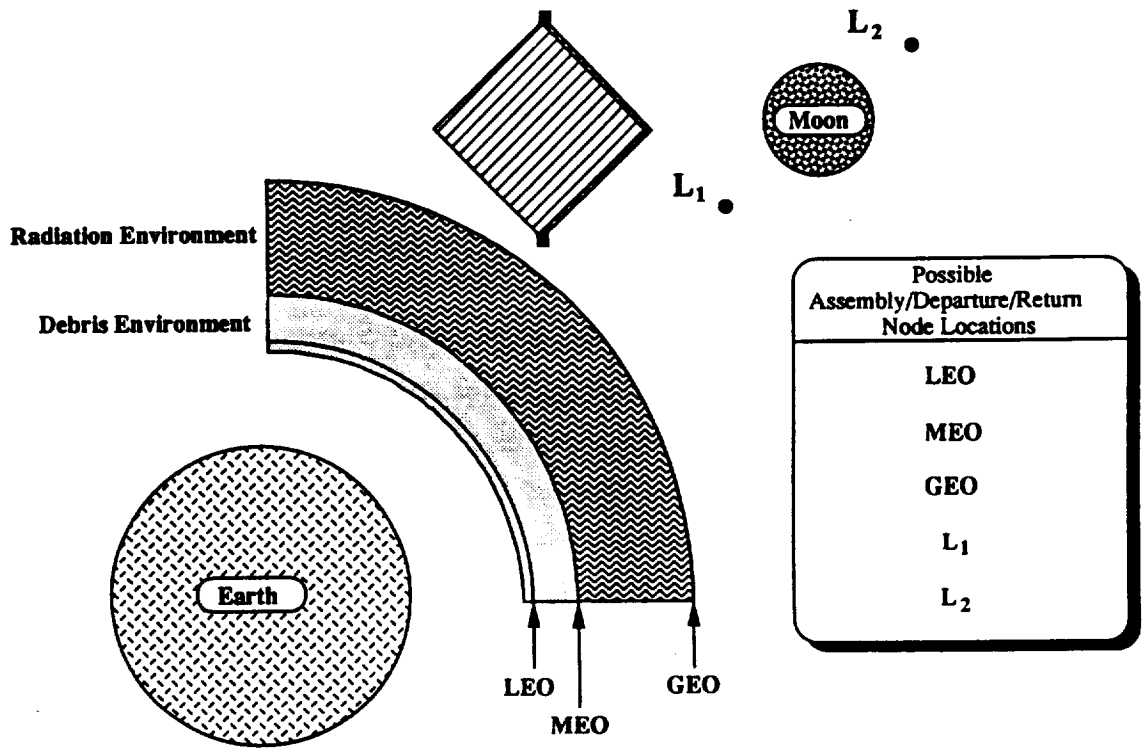
SEP thrusts for most of the trajectory, the vehicle consumes much less propellant than a high-thrust system due to the efficiency attained by the very high Isp. The major portion of the SEP vehicle is composed of the solar array and associated structure necessary to produce the required electrical power for propulsion. Solar electric propulsion is a flight proven technology with ten years of operation from the SERT II spacecraft.

Power system - The solar array is composed of GaAs/CIS multi-junction solar cells. At the onset, several types of solar cells were investigated such as silicon, indium phosphide, multi-junction, gallium arsenide, thin film, and concentrator cells. For interplanetary transfer, a solar cell that is efficient and lightweight is preferred. The multi-junction solar cell array was found to be the most efficient and lightweight array. Another consideration for the solar cell selection was its performance in the van Allen belt radiation environment. The indium phosphide cell has been shown to be the most radiation resistant cell. If an operation mode is chosen that includes the vehicle spiraling through the van Allen belts, an indium phosphide array would be a prime candidate. The difference in cell efficiency between the indium phosphide array and the multi-junction array would be made up for by the amount of radiation degradation the multi-junction cells would experience.

Propulsion System - The propulsion system is similar to that used by the NEP vehicle (see Section 4.2.2.4).

Propulsion performance and assumptions - A parametric mission analysis study was performed to establish a baseline power level for a SEP point design. Further trades were also performed to determine the effects of vehicle alpha, Isp, trip time, and opportunity. Results of this analysis and the relevant assumptions can be found in Section 3 and Reference 9. The SEP differs from the NEP in the sense that as the vehicle gets farther away from the sun, the power generated decreases. When the power decreases the power to weight ratio also decreases which results in a lower Isp optimization. The propulsion system is composed of roughly 1 engine per MWe of required power. The reference vehicle is designed at 10 MWe with spare engines.

Operational Considerations - The major operational consideration for SEP is the assembly/departure and refurbishment node locations and operation modes. Details on the operations mode may be found in Reference 6. A trade was performed to determine the optimum assembly/departure node locations as well as operation modes. The node



SEP EARTH ORBITAL OPERATIONS
FIGURE 4-7

locations and operation modes considered are shown in Figure 4-7. The criteria used to evaluate the options and results are shown in Figure 4-8.

The six options are:

a. The vehicle is assembled in LEO with the transfer array deployed and the main array in a stowed configuration. The vehicle spirals out of the Van Allen Belts with its own propulsion system where the transfer array is left for another possible application. At this point, the vehicle deploys the main array and initiates the earth escape sequence. Prior to Earth escape, the crew will rendezvous with SEP vehicle via LTV a few days prior to Earth escape.

b. The vehicle is fully assembled in LEO with the main array. The main array can be oversized to account for the radiation degradation or the array can be composed of a radiation resistant solar cell (I_nP) will tolerate more passes through the Van Allen Belts than would an oversized array, therefore the radiation resistant solar cell should be favored. After assembly, the vehicle initiates the Earth escape sequence with crew rendezvous via LTV a few days prior to Earth escape.

Criteria	SEP Mission Options												
	Option 1		Option 2		Option 3		Option 4		Option 5		Option 6		Weights
1. Total IMLEO	5	4.15	5	4.15	1	.83	4	3.32	3	2.49	3	2.49	.83
2. Orbital Debris Shield Mass	1	.56	1	.56	5	2.80	5	2.80	1	.56	2	1.12	.56
3. Mass @ Spiral Initiation	2	.66	2	.66	3	.99	3	.99	3	.99	2	.66	.33
4. Mass @ Departure	3	.84	1	.28	3	.84	3	.84	3	.84	3	.84	.28
5. Isp (transfer stage)	5	3.60	5	3.60	1	.72	5	3.60	5	3.60	3	2.16	.72
6. Spiral Time	1	.50	2	1.00	5	2.50	5	2.50	5	2.50	4	2.00	.50
7. DeltaV	5	3.90	5	3.90	2	1.56	1	.78	5	3.90	4	3.12	.78
8. Days Exposure to Radiation	1	.39	2	.78	5	1.95	5	1.95	5	1.95	4	1.56	.39
9. Days Exposure to Orbital Debris	1	.44	1	.44	5	2.20	5	2.20	1	.44	2	.88	.44
10. Total Mission Time	3	1.83	1	.61	5	3.05	2	1.22	1	.61	4	2.44	.61
11. Resiliency Time	5	4.10	5	4.10	1	.82	1	.82	5	4.10	4	3.28	.82
12. Infrastructure Cost	4	3.80	5	4.75	3	2.85	2	1.90	3	2.85	4	3.80	.95
13. Infrastructure Complexity	4	.68	5	.85	2	.34	1	.17	2	.34	3	.51	.17
14. % Degradation of Solar Array	5	3.35	4	2.68	5	3.35	5	3.35	5	3.35	2	1.34	.67
15. Total HLLV Flights	5	4.45	5	4.45	1	.89	4	3.56	3	2.67	3	2.67	.89
16. Reusability of Used Hardware	3	.03	1	.01	1	.01	5	.05	5	.05	1	.01	.01
17. GCR Exposure to Crew	3	.33	5	.55	1	.11	1	.11	3	.33	4	.44	.11
18. Flight Proven Technology	1	.06	1	.06	1	.06	5	.30	5	.30	1	.06	.06
Total Scores	33.67		33.43		25.87		30.46		31.87		29.38		

1 unfavorable 2 3 average 4 5 favorable

SEP MISSION OPTIONS MATRIX
FIGURE 4-8

c. The vehicle components are boosted with a chemical stage to a high Earth orbit (out past the Van Allen Belts) for assembly. Once the vehicle is assembled at this node, the vehicle initiates the Earth escape sequence. Prior to Earth escape, the crew will rendezvous with the SEP vehicle via LTV a few days prior to Earth escape.

d. The vehicle components are boosted with an electrical orbital transfer vehicle (EOTV) to a high Earth orbit (out past the Van Allen Belts) for assembly. The EOTV should be a beamed power EOTV to avoid occultation and drag associated with a solar powered EOTV and radiation impacts associated with a nuclear powered vehicle. Once the vehicle is assembled at this node, the vehicle initiates the Earth escape sequence. Prior to Earth escape, the crew will rendezvous with the SEP vehicle via a few days prior to Earth escape.

e. The vehicle is assembled in LEO with the main array in stowed configuration. The vehicle is then "tugged" to a higher Earth orbit by an EOTV (powered as an Option #4). The EOTV will be much larger than in Option #4 due to the increased payload. Once the vehicle is clear of the radiation belts, the main array will be deployed and the vehicle will initiate the Earth escape sequence. Prior to Earth escape, the crew will rendezvous with the SEP vehicle via LTV a few days prior to earth escape.

f. The vehicle is fully assembled in LEO without the payload. The vehicle then spirals out past the radiation belts to wait for payload rendezvous. The transfer without the payload allows for a fast spiral time, which reduces the amount of debris and radiation degradation to the SEP vehicle. The payload is then sent to the SEP vehicle by means of a chemical boost stage. Once the payload has been attached, the SEP vehicle begins the Earth escape sequence. Prior to Earth escape, the crew will rendezvous with the SEP vehicle via LTV a few days prior to Earth escape.

The MEO node location was dismissed early due to unacceptable debris and radiation environments. Due to the slow spiral times associated with electric propulsion, environmental impacts become a primary driver in node selection. The trade assumed that no infrastructure was in existence so any option that used support hardware was penalized fully. The two options that were favored in this trade were a vehicle that uses a transfer array to get past the van Allen belts (the array could possibly be used for power beaming or other application) before installing the main array and a vehicle that uses a less efficient, radiation resistant indium phosphide array for the van Allen belt transfer and the main interplanetary transfer. Further details and results can be found in Reference 9.

4.2.2.6 Reference Vehicles Subsystem Summary

A concise listing of the principal characteristics of the seven major subsystems for each of the seven major vehicle stages of the reference configuration is given in Reference 6.

4.3 Advanced Propulsion Trades:Overall Vehicle Performance Comparisons

4.3.1 Introduction

A major portion of this task applied only to Mars transfer propulsion systems. After the baseline reference trades were complete, subsequent trades and sensitivity analyses were done in order to compare overall performance of the baselines and their applicability to several of the SEI mission scenarios. These analyses included evaluations of potential performance enhancements and their impact on the comparisons. Frequently, results of the trades are presented as delta's to the reference vehicles IMLEO, performance or operational characteristics.

As the advanced propulsion trade progressed it was realized that several key parameters and requirements had great bearing on differentiating the propulsion system concepts or for demonstrating a superior option among the group of candidates. The advanced propulsion trades also included much work in the methodology of mission and trajectory analysis which is not treated in this section. Seven key parameters are listed below:

- Specific transfer trajectory establishes mission delta V's required of the propulsion system as well as trip duration required of the crew habitat system
- First flight date establishes crucial technology readiness timetable
- Flight frequency influences payoff of vehicle reuse mode
- Propulsion system Isp
- Technology development considerations
- Propulsion system operations considerations

An assessment of propulsion system performance was carried out. The six mission trajectory classes evaluated are:

- Oppositions without swingbys - short to intermediate trip times; 30 day surface stays
- Oppositions with swingbys - intermediate total trip times; 30 day surface stays
- Mars flyby oppositions with surface exploration mode; also called dash/flyby mode - intermediate total trip times; 20-50 day surface stay times.
- Conjunctions - long total trip times; long surface stay times

- Fast Transfer conjunctions - transfer times as low as 200 days; stay times 450 - 660 days
- Split Sprint missions - short trip times (opposition profile) for piloted craft, unmanned cargo vehicle sent ahead on low energy conjunction trajectory

4.3.2 Non-Venus Swingby Oppositions

Oppositions without swingbys represent the class of trajectories that can be utilized to provide short total trip times, but also require high total mission delta V's. Trip times of 350-500 days were evaluated.

The non-swingby opposition class trajectory delta V's vary considerably with the Earth-Mars synodical cycle; for this comparison two representative opportunity years were chosen: 2018, a relatively 'easy' year (in terms of total delta V), and 2025, a 'hard' opportunity year. For each of these, optimized trajectories with total trip times of 350, 400, 450, and 500 days were evaluated. The delta V's and departure dates for these missions are given in Figure 4-9.

Trade Objective - The objective of the trade was two fold: first, gain an understanding of how the IMLEO for each propulsion/vehicle concept varies with trip time, and second, compare one to another to determine which concept could distinguish itself as more capable for missions with trip times too short to benefit from gravity assists at Venus (<500 days), or could best cover the range of non-swingby opportunities between 2010 and 2025.

Trade Results - The results of the analysis indicate that propulsion options with higher Isp are not as sensitive to variations in mission opportunities. Vehicle weight can increase by factors of two or three from a mission performed in the most favorable year to the least favorable. IMLEO is plotted vs trip time for the CAB, advanced NTR, NEP and SEP concepts in Figure 4-10. Mission opportunities are represented by a 'band' that forms the boundary for all optimum trajectories between 2010-2025. For the high-thrust options without swingbys, a 450 day mission usually produces lowest IMLEO. The following observations can also be made:

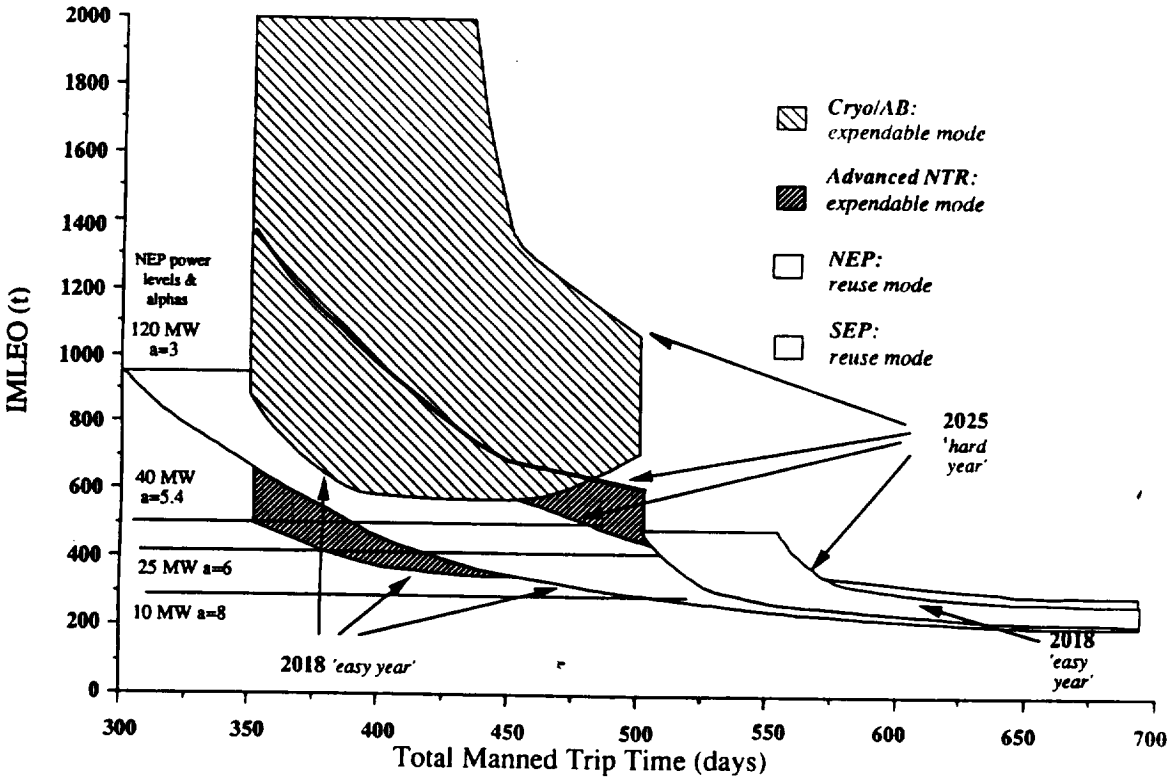
- For the CAB vehicle
  - The CAB is unable to do the 2025 hard missions with reasonable IMLEO.

	Total trip time	Julian dates TMI	MOC	TEI	EOC	(1) TMI dV	Outb Deep Space burn	MOC Vhp	(2) Δ	MOC dV	TEI dV	EOC Vhp	(3) Δ	EOC dV
2018 'easy year'	350	8280	385	415	630	4.101	0	6.857	0	3.772	5.460	9.524	0	4.027
	400	8270	400	430	670	3.741	0	5.260	0	2.528	3.763	10.217	0.517	4.506
	450	8230	400	430	680	3.610	0	4.521	0	1.857	3.646	11.000	1.300	3.747
	500	8140	375	405	640	4.489	0	4.590	0	2.098	3.603	9.523	0	4.046
2025 'hard year'	350	0390	595	625	740	10.228	0	7.020	0.020	3.984	6.223	8.050	0	3.119
	400	0365	579	609	765	7.805	1.613	7.156	0.156	4.018	4.193	4.744	0	1.428
	450	0300	540	570	750	5.187	2.761	7.229	0.229	4.079	2.030	6.020	0	2.000
	500	0275	391	421	775	4.374	2.925	6.761	0	3.693	1.946	3.749	0	1.541

(1) *g-losses not accounted for*
(2) all aerocapture veh's arriving Mars with Vhp>7 (km/s) use cryo propulsion to slow veh down to Vhp=7 for aerocapture
(3) ECCV's arriving Earth with Vhp>9.7 (km/s) use cryo propulsion to slow ECCV capsule down to Vhp=9.7 for entry

MISSION DELTA V AND DEPARTURE DATES, NON-SWINGBY

FIGURE 4-9



2010-2025 NON-SWINGBY OPPOSITION MISSION MARS VEHICLES

FIGURE 4-10

- CAB mission opportunities that can be considered as having resonable IMLEO occur in the 500 to 630 day
- CAB is most sensitive to variable opportunities, due to its inherently lower Isp.

- For the NTR vehicle

  - NTR offers significant mass savings over the CAB for any opportunity
  - Advanced NTR offers reasonable IMLEO figures for quick trips (<400 days) for all but "hard" opportunities
  - The NERVA NTR is affected more than the advanced NTR in the harder years and faster trips due to the lower Isp and T/W ratio associated with the NERVA technology.
  - The mass penalty for designing a reusable advanced NTR vehicle is small compared to the dollars benefit gained in returned hardware. For a reasonably small mass penalty (~100 t average) the NTR vehicle can be returned to Earth for future missions.

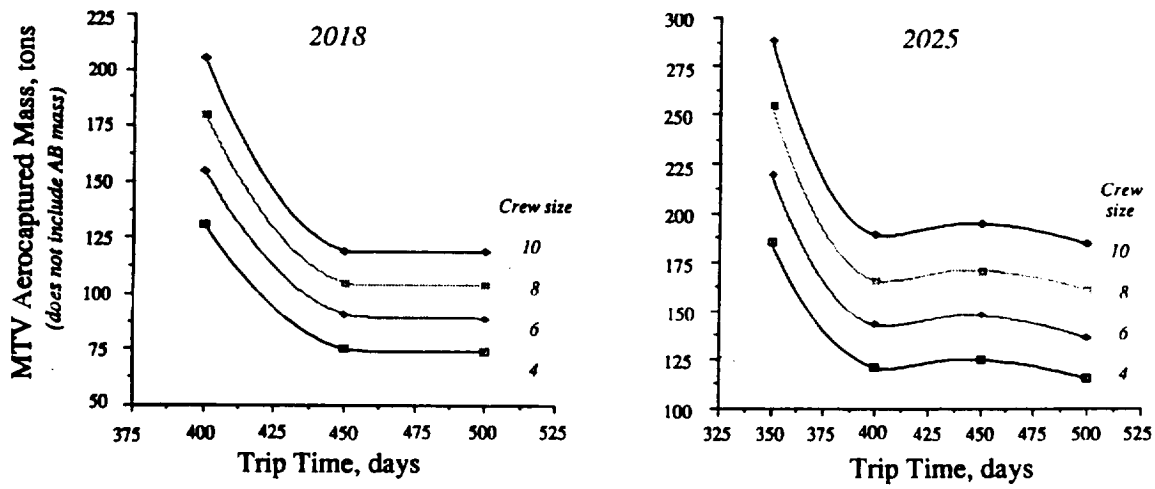
- For the NEP vehicle

  - Electric propulsion offers significant mass savings over the CAB for any opportunity
  - NEP offers reasonable IMLEO figures for quick trips (<400 days)
  - NEP offers low mass for all opportunities and is flexible for the entire time frame of interest (350-700 days)
  - Slow spiral out time for NEP from LEO to Earth escape creates the necessity for some type of chemical OTV or LTV to deliver crew to NEP vehicle prior to Earth departure

- For the SEP vehicle

  - SEP offers low mass for all opportunities in the 550-700 day range
  - SEP is incapable of short trip time missions (< 450 days)
  - SEP would require a significant spiral up time (100 days) from LEO to Earth escape; creates the necessity for some type of chemical OTV or LTV to deliver crew to SEP vehicle prior to Earth departure

CAB aeroshell capture trade - The CAB vehicles aeroshells capture mass was evaluated for 350, 400, 450, and 500 day trip time missions for both the 2018 and 2025 opportunity years. The required weight that the MTV aerobrake must capture into Mars orbit for these missions is shown in Figure 4-11. This weight does not include the MEV system, since for the CAB vehicles, the MEV has its own aeroshell and aerocaptures independently of the MTV. Captured weight in tons is plotted vs total mission trip time in days. The assumption was made for this analysis that the aerobrake weighed 15% of the captured weight, and that the vehicle was not reusable. This captured weight actually included the weight of the brake itself. To restate the brake weight percentage in terms of captured weight not including the brake weight, the percentage would be 17.7%. The aeroshell was dropped after Mars capture.



MTV AEROCAPTURED MASS AT MARS VS TRIP TIME
FIGURE 4-11

4.3.3 Oppositions with Venus Swingbys

Oppositions with swingbys represent a class of trajectories that provide intermediate total trip times with reduced total mission delta V's over oppositions without swingbys. The typical trip times for this group range from 530 to 675 days. For this analysis only the high thrust systems were evaluated. A few of the opportunity years, for example 2016, offer "fast" Venus swingby trajectories of less than 450 days.

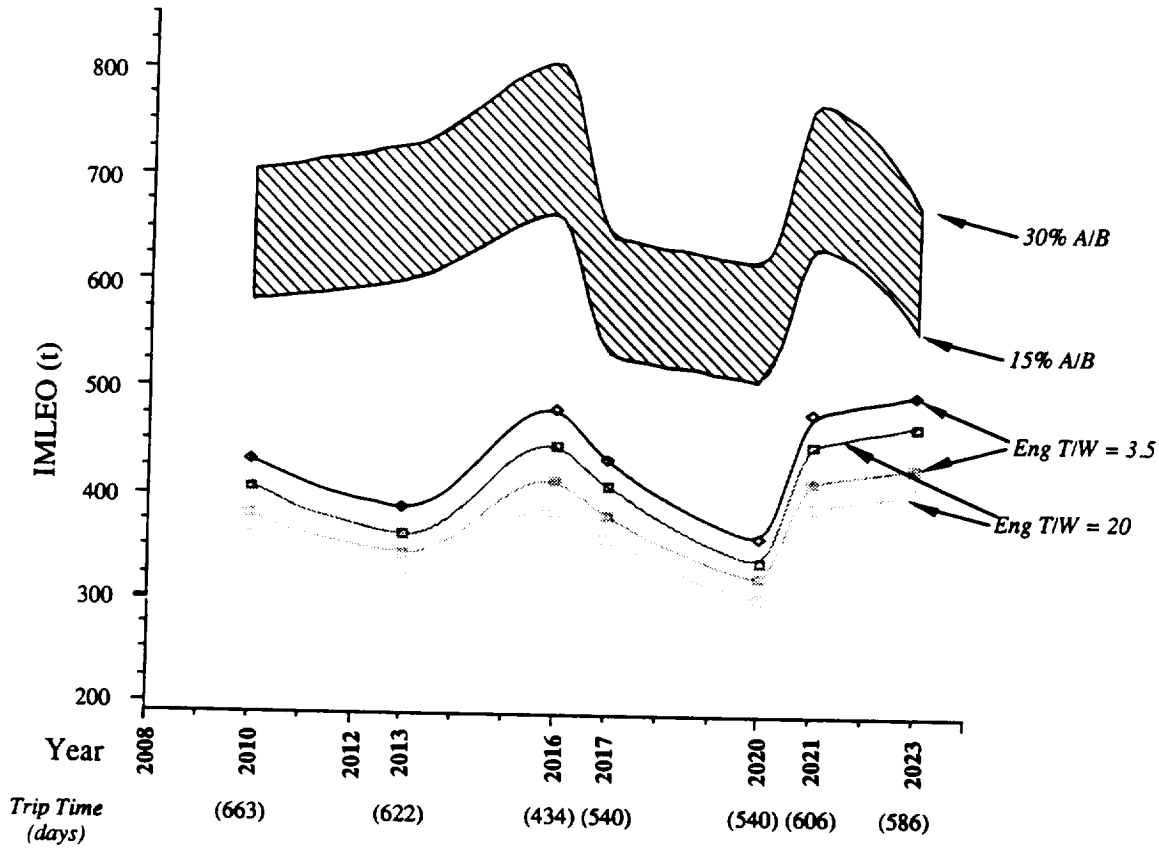
Seven optimum Mars opposition trajectories with Venus swingbys were developed in cooperation from MSFC, LeRC and JSC for the high thrust systems only. The delta V's and departure dates for these missions are given in Figure 4-12.

Year	Total trip time	TMI	*Julian dates* MOC	TEI	EOC	TMI dV	Outb Deep Space burn	MOC Vhp	MOC dV	TEI dV	EOC Vhp	EOC dV *elliptical orbit*	Total dV
2010	673	5529	5859	5889	6192	4.426	0	4.927	2.318	1.310	7.550	2.831	10.885
2013	632	6618	6899	6929	7240	3.692	0	3.374	1.280	3.235	4.406	1.134	9.341
2016	434	7463	7621	7651	7897	3.805	0	5.308	2.562	3.979	5.562	1.799	12.145
2017	540	7850	8201	8231	8390	4.249	0	5.480	2.703	1.115	3.834	1.110	9.177
2020	540	9055	9216	9246	9595	3.867	0	3.761	1.523	1.826	4.235	1.251	8.467
2021	606	9518	9820	9850	10124	4.258	0	5.795	2.944	2.520	8.172	3.138	12.860
2023	586	10194	10494	10524	10780	4.264	0	6.391	3.466	1.464	2.820	0.811	10.005

MISSION DELTA V'S AND DEPARTURE DATES, VENUS SWINGBY
FIGURE 4-12

Trade Objective - The objective of the trade was to determine which concept could distinguish itself as more capable for missions benefiting from gravity assists at Venus, or could best cover the range of swingby opportunities falling between 2010 and 2025.

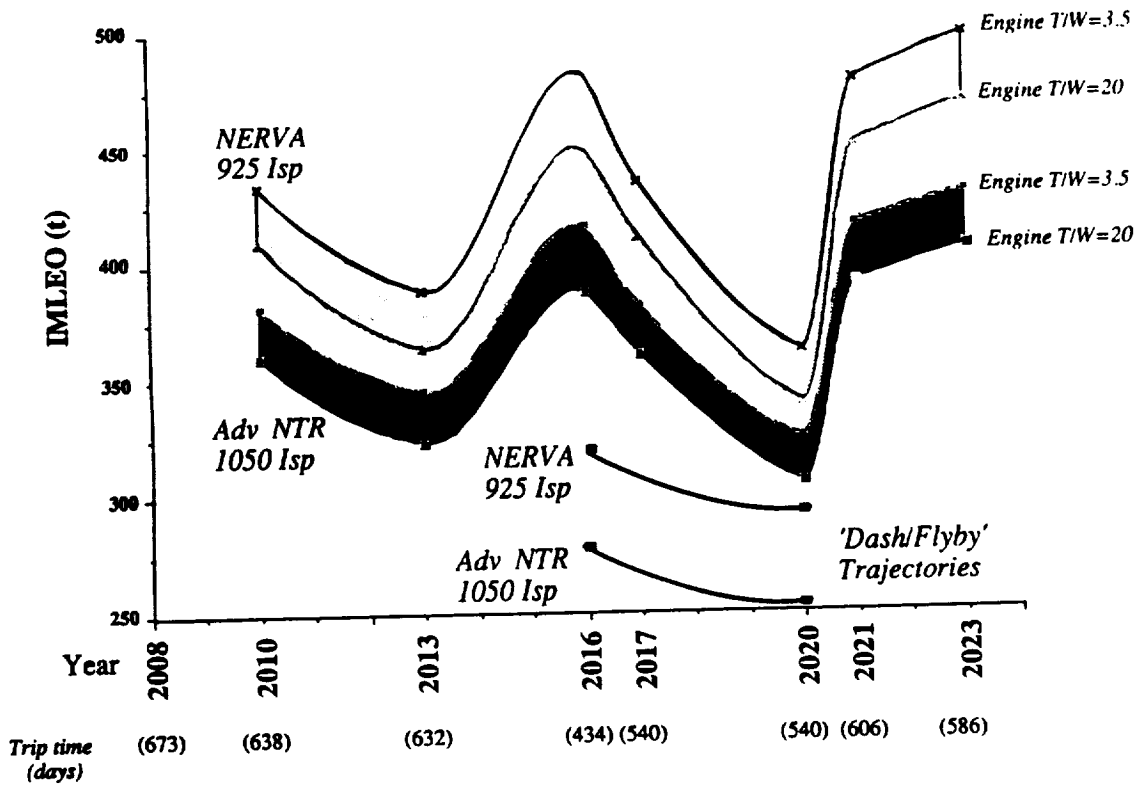
Trade Results - The CAB, NERVA NTR, and advanced NTR vehicle IMLEO estimates are presented with their corresponding swingby mission opportunities from 2010 to 2023 in Figures 4-13 to 4-16. The CAB IMLEO data is bounded by lines representing aerobrake masses equal to 15 % and 30 % of brake captured weight at Mars. NERVA class and advanced NTR vehicle data are bounded by lines representing engine T/W ratios of 3.5 and 20. The large dip in the CAB IMLEO seen occurring at the 2017 swingby opportunity is due to the relative distribution of the Mars departure and capture delta V's; in 2017, MOC delta V is large relative to TEI delta V. CAB vehicle IMLEO is insensitive to Mars capture delta V, so the marked advantage in lower IMLEO characteristic of NTR over CAB is reduced somewhat for those opportunities which have high Mars orbit capture delta V's relative to the TEI delta V. For this reason all-propulsive and aerobraking trajectories are sometimes optimized differently for the same opportunity years. A representation of this data in a bar chart format is given in Figure 4-15.



NTR AND CAB VEHICLES IMLEO COMPARISON, OPPOSITION MISSION WITH SWINGBY
FIGURE 4-13

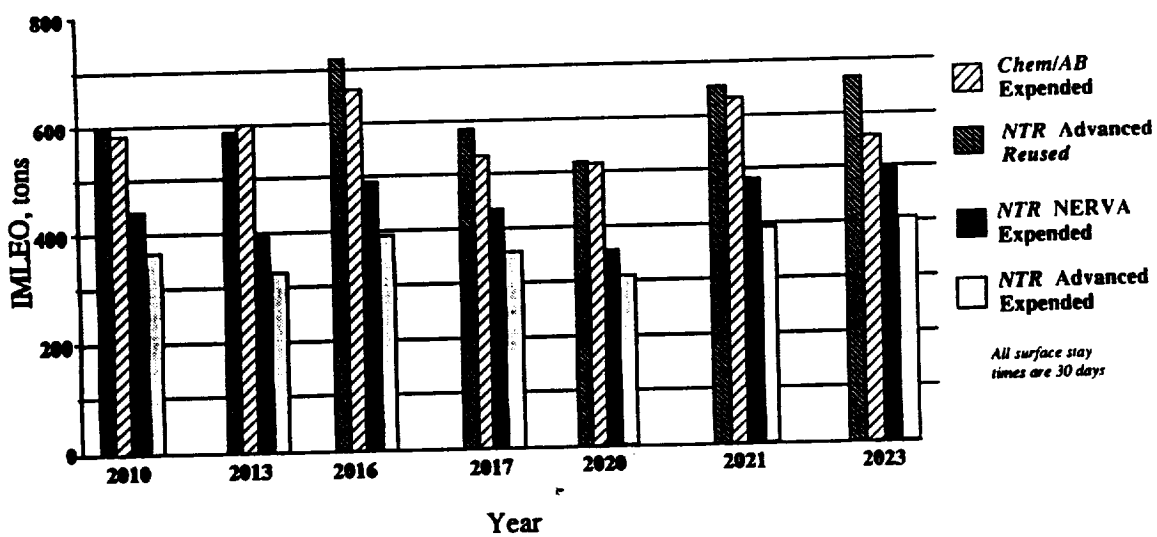
Although most swingby opportunities have a longer trip time than the non-swingby opportunities, they offer a lower IMLEO for all years due to the Venus gravity assist benefit, providing delta V reduction. The swingby IMLEOs are represented as discrete points in Figure 4-16, and not as 'bands' as in Figure 4-13 and 4-14. The preceding IMLEO vs swingby opportunity year graph presented the swingby points as being part of a band. It should be noted that there is no continuity between two swingbys or a swingby and non-swingby trajectory.

The majority of swingbys missions are in the 530 - 675 day trip time regime. An important point to note is that some years contain an outbound swingby opportunity, while some years contain an inbound swingby. This situation imposes less that an 18 month departure time between consecutive opportunities. This restricted time frame could interfere with time restraints on the launch of HLLVs and assembly for achieving the next mission.



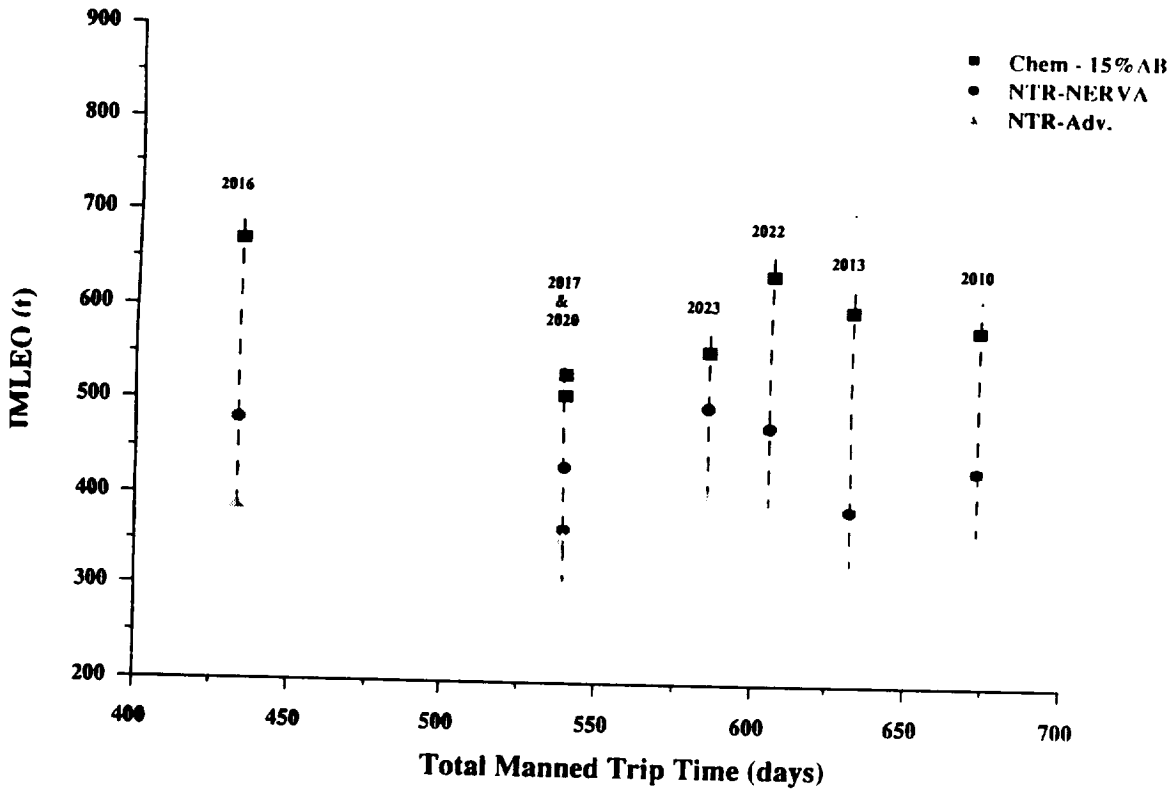
ADVANCED NTR AND NERVA NTR VEHICLES IMLEO COMPARISON, OPPOSITION MISSION WITH SWINGBY

FIGURE 4-14



CAB AND NTR VEHICLES IMLEO VS. OPPORTUNITY YEAR, VENUS SWINGBY OPPOSITION MISSIONS

FIGURE 4-15

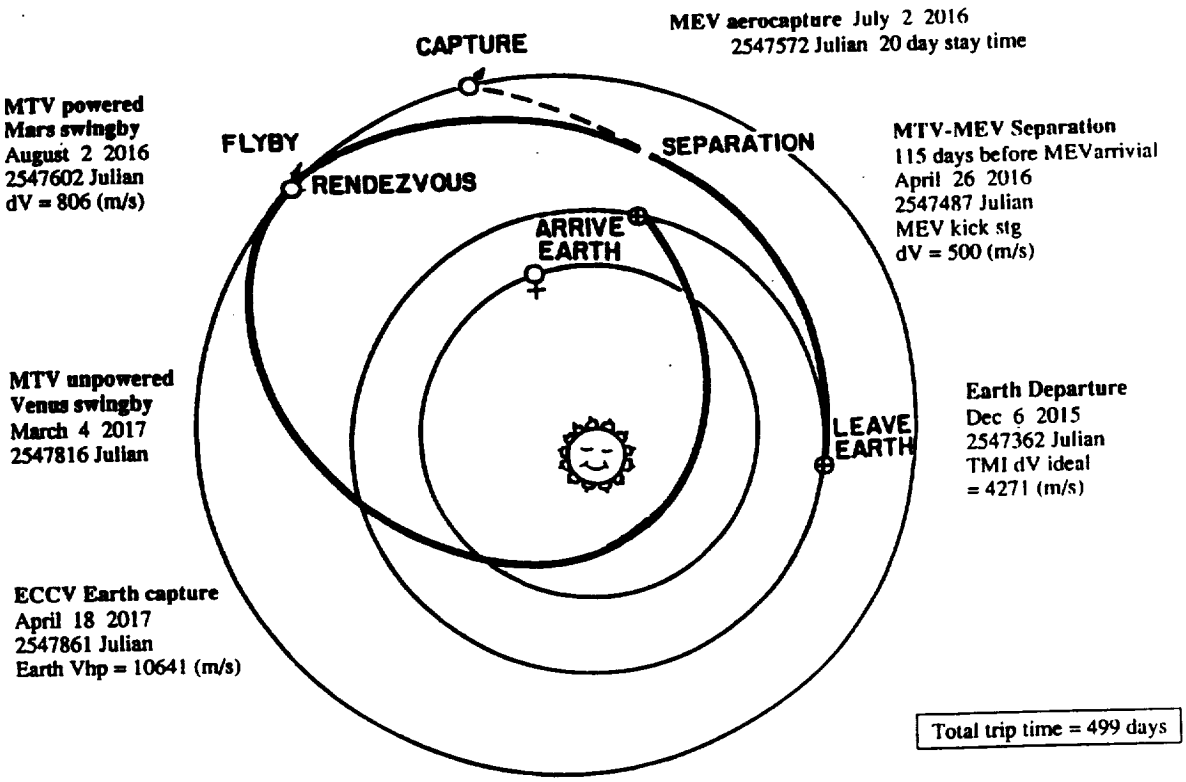


PROPULSION OPTION SWINGBY OPPORTUNITIES
FIGURE 4-16

Results - For the 2010-2023 swingby opportunities the NTR vehicle is the preferable system. The NTRs potential for evolution from the near term performance level (Isp 925 sec range), to the higher performance range (Isps 1000+ sec) in the out years is good.

4.3.4 Mars Flyby with Surface Exploration Mission

The Mars flyby with surface exploration mission, hereafter called the 'dash/flyby' mission, is a special opposition mission case that provides low total mission delta V's, with intermediate trip times (400-550). A Mars flyby trajectory is shown in Figure 4-17 with a 'dash' (acceleration) off the trajectory path occuring past the midpoint of the outbound transfer leg. The dashed line extending out to Mars ahead of the main MTV stage flyby path illustrates the accelerated MEV flight portion that is the distinctive characteristic of the dash/flyby mission.

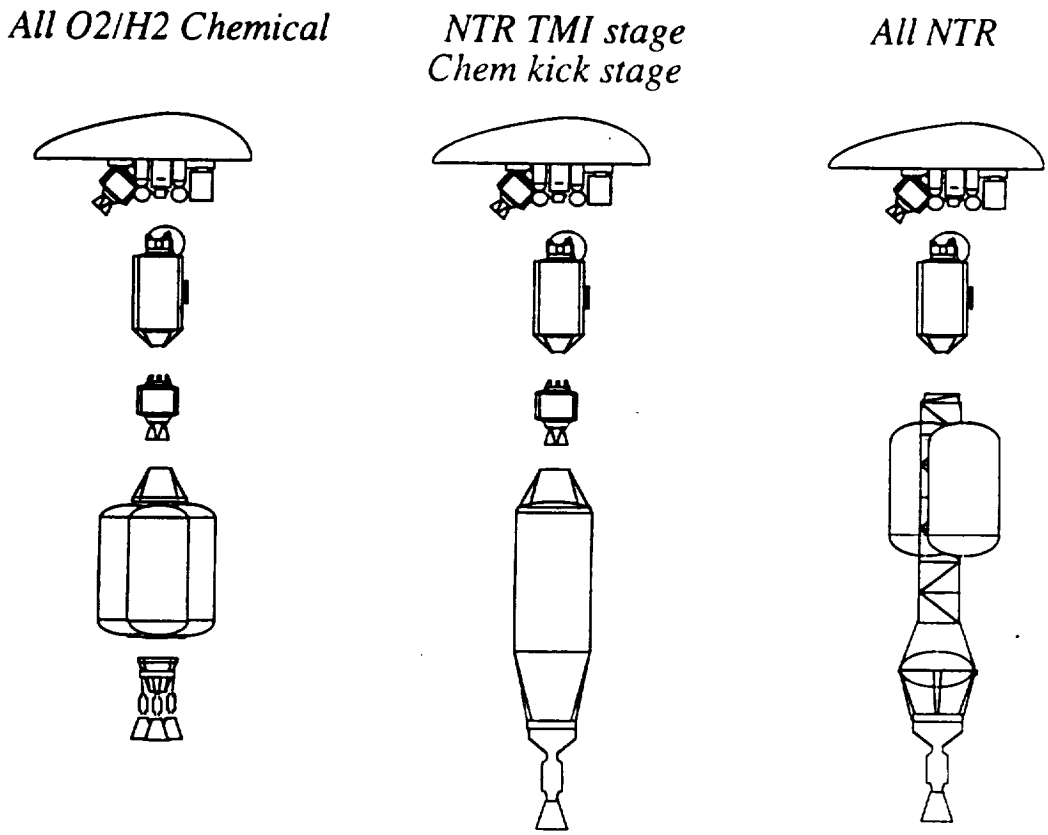


2016 MARS FLYBY WITH SURFACE MISSION TRAJECTORY
FIGURE 4-17

Trade Objectives - The trade objective was to determine the saving in IMLEO afforded by the dash/flyby trajectory versus the reference "stopover" (traditional capture opposition) missions. Identical MTV crew module mass and MEV delivered surface cargo masses were used for the comparisons. Vehicle propulsion options evaluated are illustrated in Figure 4-18.

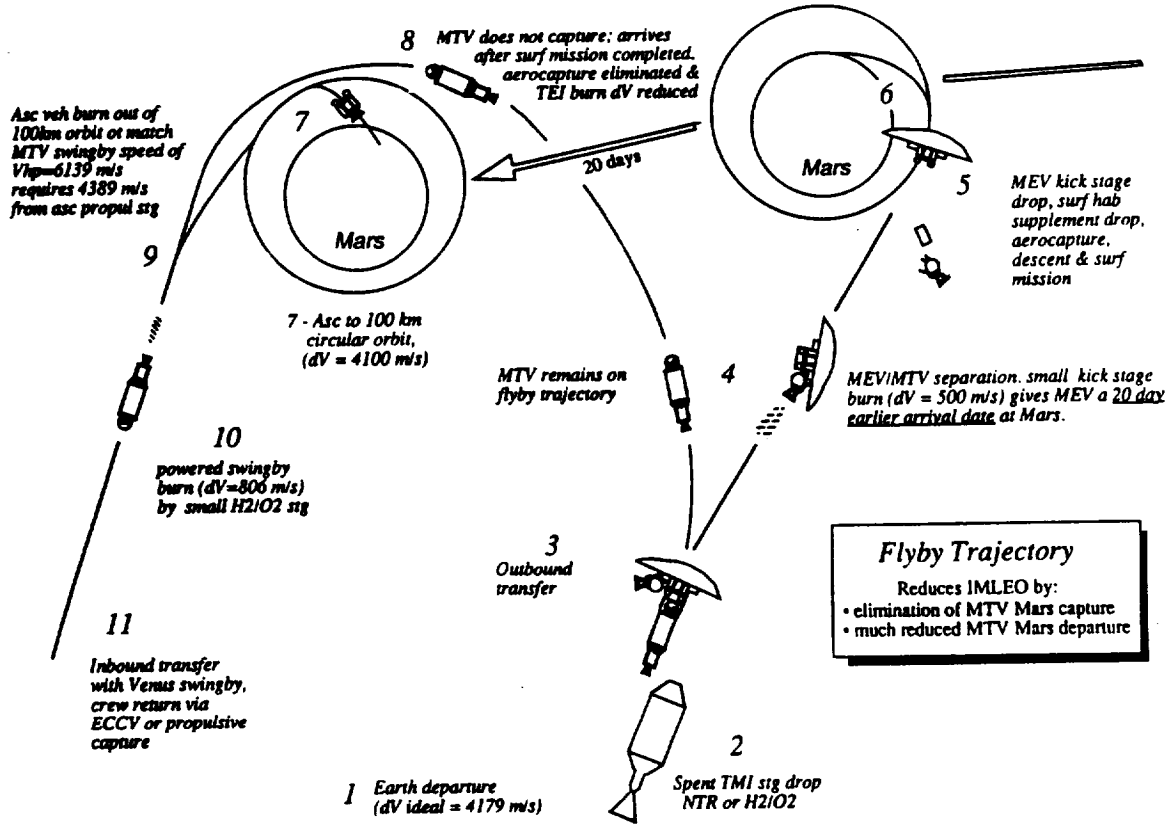
Dash/Flyby Mission Profile - The Mars spacecraft departs Earth on a standard Mars flyby trajectory and continues on until past the midpoint of the outbound leg, Figures 4-17 and 4-19. At a specified time the MEV will separate from the MTV stage and with a small chemical kick stage do a deep space burn of relatively low delta V (~500 m/s). This burn provides the MEV with enough extra velocity for it to reach Mars 20+ days ahead of the MTV stage which has continued on its original trajectory. For each MEV-MTV separation point, there is a unique separation velocity vector which will allow a given stay time at Mars. The MEVs new accelerated transfer leg (shown as the dashed line in Figure 4-17)

can vary in duration, depending on kick stage delta V and the time of separation from the MTV, Figure 4-20. A small four ton consumables module supplements the MEV surface habitat module utilized by the MEV crew for this partial transfer leg and is dropped just prior to MEV aerocapture or MEV direct entry.



DASH/FLYBY VEHICLE PROPULSION OPTIONS
FIGURE 4-18

After Mars orbit capture, the MEV does its surface mission, which culminates with crew ascent to a waiting parking orbit via the ascent vehicle. The incoming MTV stage does not capture, but rather swings by Mars in a hyperbolic turn. The ascent vehicle then does a second burn to depart orbit and establishes a hyperbolic path ahead of the the MTV path, and effects a rendezvous when the MTV catches up several Mars radii distant from the planet. The two propulsion options available to the ascent vehicle to do the Mars departure burn are illustrated in Figure 4-21. There are unique combinations of rendezvous distance and Mars escape speed which result in a tangential intercept between the flyby and rendezvous trajectories, which would ease rendezvous guidance problems.

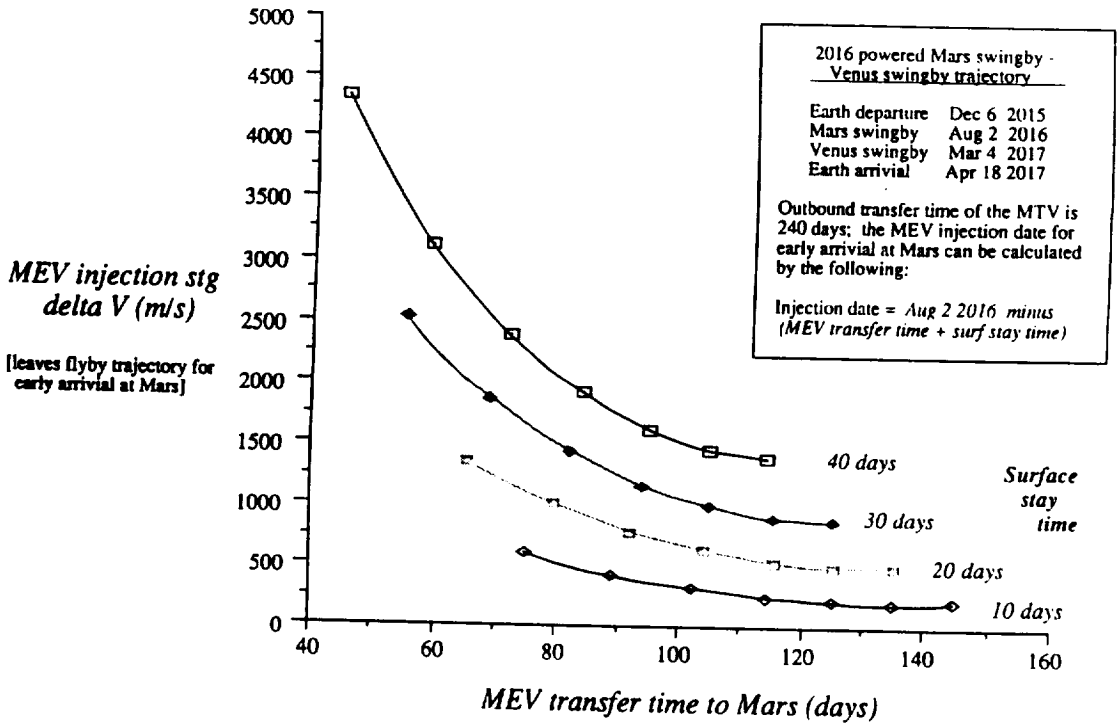


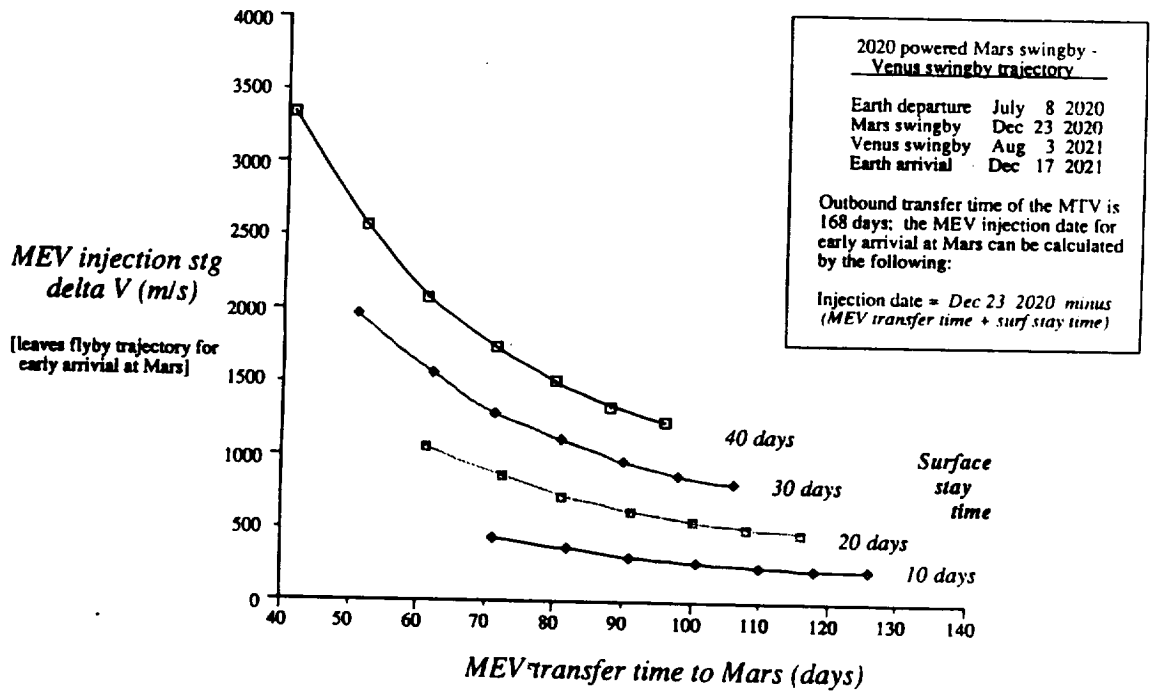
DASH/FLYBY MISSION PROFILE
FIGURE 4-19

The hyperbolic rendezvous maneuver can be made over a wide range of intercept distances from the planet with little variation in the over all propulsion requirements. Coplanar paths were assumed for the rendezvous burn. After crew transfer, the MTV stage performs a modest powered swingby burn, if necessary. An unpowered Venus swingby occurs on the inbound leg.

The two options mentioned for Mars departure of the ascent vehicle are explained as follows:

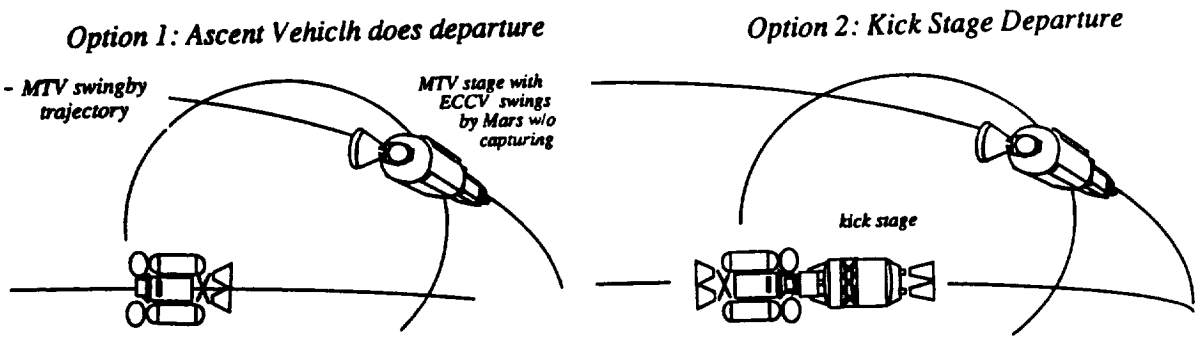
Option 1. After ascent to the Mars parking orbit, the ascent vehicle does a second burn utilizing its own propellant to depart orbit and effect a rendezvous when the MTV catches up to it several Mars radii distant from the planet. The added ascent stage delta V necessary for the hyperbolic departure was calculated based on the MTV's swingby Vhp at Mars.





MEV INJECTION DELTA V FOR SURFACE STAY AND TRANSFER TIME, (TOP) 2016 MISSION, (BOTTOM) 2020 MISSION

FIGURE 4-20



DASH/FLYBY/SURFACE RENDEZVOUS WITH MTV ON SWINGBY
FIGURE 4-21

Option 2. After ascent to the Mars parking orbit, the ascent vehicle rendezvous with the small kick stage. The kick stage may or may not have been first utilized for the deep space MEV kick injection burn for early Mars arrival. Its second task (option 2 only) is to do the orbit departure burn for rendezvous with the MTV stage. The MEV descent stage, ascent stage and this small chemical kick stage were earlier aerocaptured into orbit behind the MEV aeroshell. Before descent, the kick stage is released and stays in orbit while the MEV descends to the surface. Once the ascent vehicle has ascended from the surface and has docked with the kick stage, the kick stage does (or provides the propellant to the ascent engines to do) the departure burn. Rendezvous occurs when the MTV catches up to it several Mars radii distant from the planet. This approach is similar to the LEM/service module relationship used on the Apollo missions except in this case the small kick stage is unmanned.

The advantage of option 2 described above concerns the added propellant needed to leave Mars orbit. For option 2, this propellant does not have to be taken down to the surface and back up to orbit again, as it would if the ascent stage (option 1) did the orbit departure burn with its own propellant. This approach saves some MEV propellant weight. However, for some of the Dash/flyby missions investigated, the MTV Mars swingby Vhp was low enough (<4 km/s) that the penalty of added propellant necessary for the ascent vehicle to do

the departure burn was not enough to warrant the use of this kick stage. Opportunity year 2020 analysis did not utilize option 2 for this reason.

The IMLEO advantage of the dash/flyby trajectory lies in the way the total mission delta V is distributed among the MTV and MEV stages. In the standard "stopover" mode, all major propulsive impulses are applied to the MTV stage. For the dash/flyby, the two major velocity changes at the target planet are eliminated for the MTV stage. Mars orbit capture and Mars orbit departure burns are now unnecessary, and only a small powered Mars swingby burn (800 m/s delta V for 2016 as an example) is required for the MTV. The large Mars departure burn is now only assessed against the very light ascent stage, which must not only achieve orbital velocity, but escape velocity as well.

Eliminating the MTV stage Mars orbit capture requirement altogether, and redistributing the TEI delta V to the ascent stage is an effective means of reducing overall vehicle IMLEO while still retaining equivalent (or identical) mission goals of total trip time, surface stay time, and delivered payload.

Savings are realized because the total cumulative MTV and MEV momentum change is significantly reduced. Momentum change is a better indicator than delta V for the total impulsive energy required of the propulsion system to do the mission. For the dash/flyby type missions, the delta V required of the ascent stage is increased over that of a stopover missions, but that stages very low inert mass insures that this delta V increase effects only a modest overall system momentum increase, and hence is only a small overall contributor to overall IMLEO. In contrast to this, the large MOC and TEI delta V's characteristic of the stopover trajectories, when multiplied by the respective heavy MTV stage masses these delta V's are applied to, result in a large total momentum change. The corresponding contribution to overall vehicle IMLEO is large.

By eliminating the MOC delta V and reducing the delta V required for leaving Mars (from a full TEI orbit departure burn down to a small powered swingby burn), the necessary Mars vicinity MTV stage momentum change is reduced by an order of magnitude compared to the MTV momentum change necessary for the "stopover" mission. This is the primary advantage offered by this mission type. A momentum account profile showing five major momentum changes encountered during the 2016 mission is given in Figure 4-22.

stage	TMI stg dV (m/s)	TMI stg payload (tons)	Kick stg dV (m/s)	Kick stg p/l (t)	MOC stg dV (m/s)	MOC stg p/l (t)	Descent dV (m/s)	Descent p/l (t)	Ascent dV (m/s)	Ascent p/l (t)	TEI dV (m/s)	TEI p/l (t)	Total Vehicle Momentum change (Kg-m/s) x E6
stopover opposition:													
MTV	**3851**	311			**3870**	175					**3900**	63	2.12 kg-m/s
MEV			n/a	n/a			**4200**	53	**4100**	7			0.25 kg-m/s
													2.37 kg-m/s
													IMLEO: 565 tons
dash/flyby opposition:													
MTV	**4271**	182			**0**	64					**806**	64	0.83 kg-m/s
MEV			**500**	94 t			**4200**	58	**8474**	10			0.38 kg-m/s
													1.21 kg-m/s
													IMLEO: 324 tons

COMPARISON OF MOMENTUM CHANGES
FIGURE 4-22

Total momentum change of the dash/flyby mission is one half that of the stopover mission.

Mission delta V distribution - It can be seen from Figure 4-22 the higher ascent stage delta V (8474 m/s), is double that of the stopover mission ascent stage delta V (4100 m/s). Notice, however, that the mass that this delta V is applied to (multiplied by) is extremely small (10 ton ascent stage weight, dry) when compared to the stopover mission MOC and TEI stage payload masses of 175 ton (Mars arrival) and 63 tons (Mars departure).

Overall vehicle IMLEO's were calculated and compared to the traditional stopover opposition mission vehicle IMLEO of the same opportunity year, delivered Mars surface payload, and MTV habitat module mass.

The results of the analysis for the 2016 and 2020 mission IMLEO as a function of the Mars stay time is shown in Figure 4-23. The data for all propulsion types are summarized in Figure 4-24.

NTR NERVA vehicle comparisons:

The IMLEO savings afforded by the 2016 dash/flyby mission (499 day trajectory) expendable vehicle vs the initial reference 2016 "stopover" opposition mission (434 day trajectory) expendable vehicle is significant: IMLEO is reduced by 241 tons or 42% (324 tons dash/flyby vs 565 tons stopover IMLEO). Compared to a more optimized 2016 "stopover" mission trajectory (data obtained later in the study), the difference in IMLEO is 157 tons or 33% (324 tons vs 481 tons IMLEO).

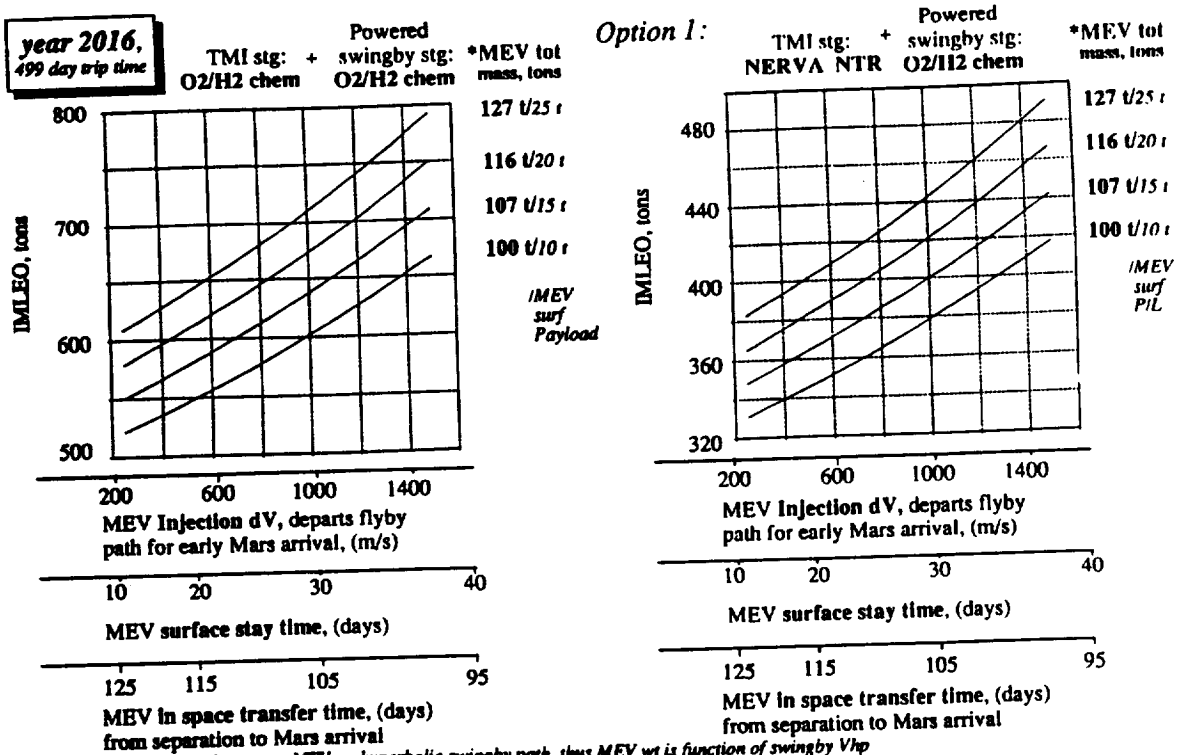
Summary of missions:

Earth return	Year	Dash/Flyby IMLEO (324 vs 480 tons)	Percentage
• Expendable	2016	157 tons savings in IMLEO (324 vs 481 tons)	33%
• Expendable	2020	65 tons savings (296 vs 361 tons)	19%
• Reusable	2016	536 tons savings (477 vs 1003 tons)	52%
• Reusable	2020	321 tons savings (335 vs 656 tons)	49%
NTR Advanced NTR vehicle comparisons:			
• Expendable	2016	110 tons savings (277 vs 387 tons)	28%
• Expendable	2020	52 tons savings (253 vs 305 tons)	17%
• Reusable	2016	307 tons savings (413 vs 720 tons)	43%
• Reusable	2020	217 tons savings (301 vs 518 tons)	42%
CAB vehicle comparisons (aerobrake 15% of captured mass):			
• Expendable	2016	96 tons savings (573 vs 669 tons)	14%
• Expendable	2020	17 tons savings (493 vs 510 tons)	3%

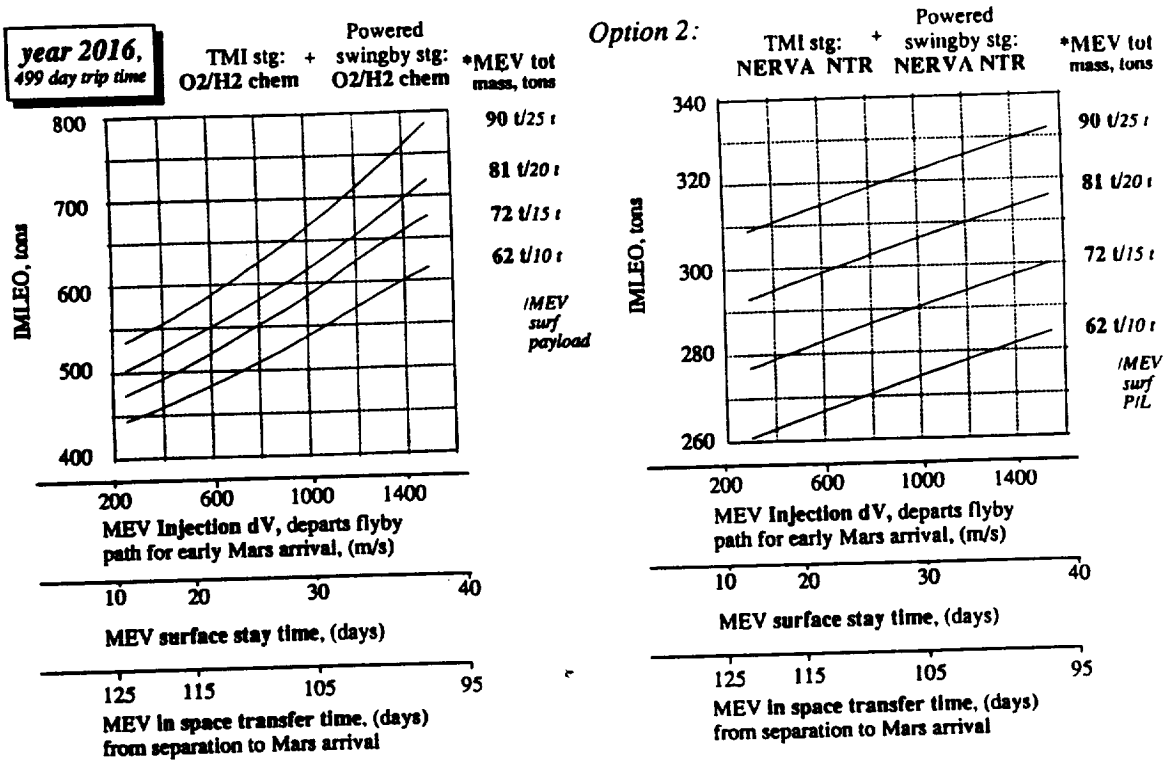
Hybrid vehicle: TMI stage: NERVA NTR/Mars swingby stage: cryogenic

• There is no Hybrid "stopover" mission vehicle for direct comparison

The 2016 and 2020 dash/flyby vs stopover mission IMLEO comparison summary chart (Figure 4-25) presents the savings in vehicle weight over the traditional stopover opposition missions.

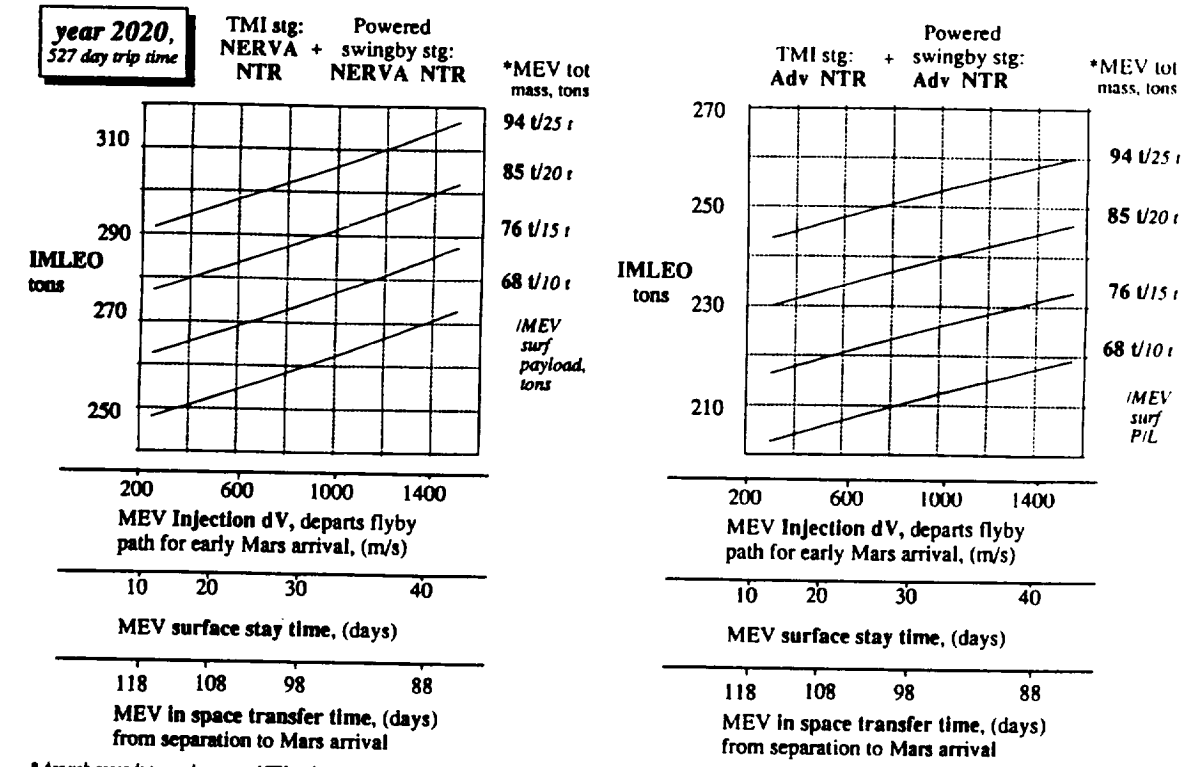


* Asc veh ascends to rendezvous w MTV on hyperbolic swingby path, thus MEV wt is function of swingby Vhp

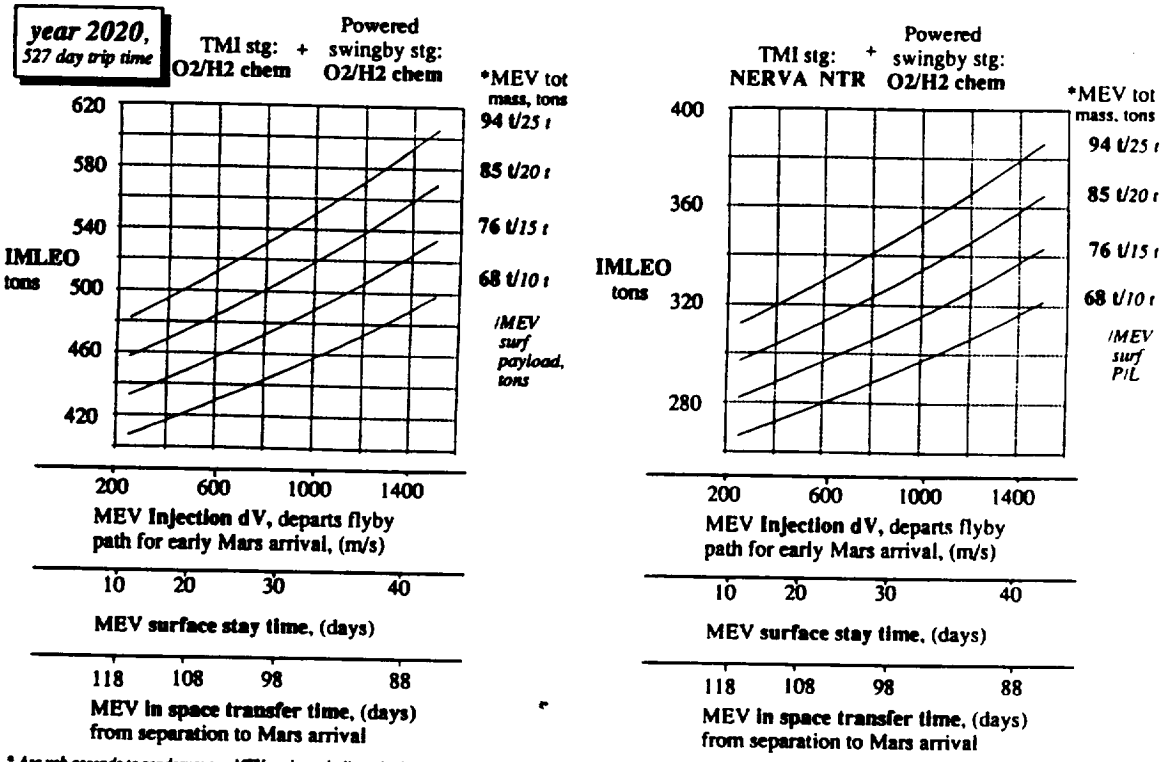


MARS FLYBY MISSIONS WITH SURFACE EXPLORATION

FIGURE 4-23



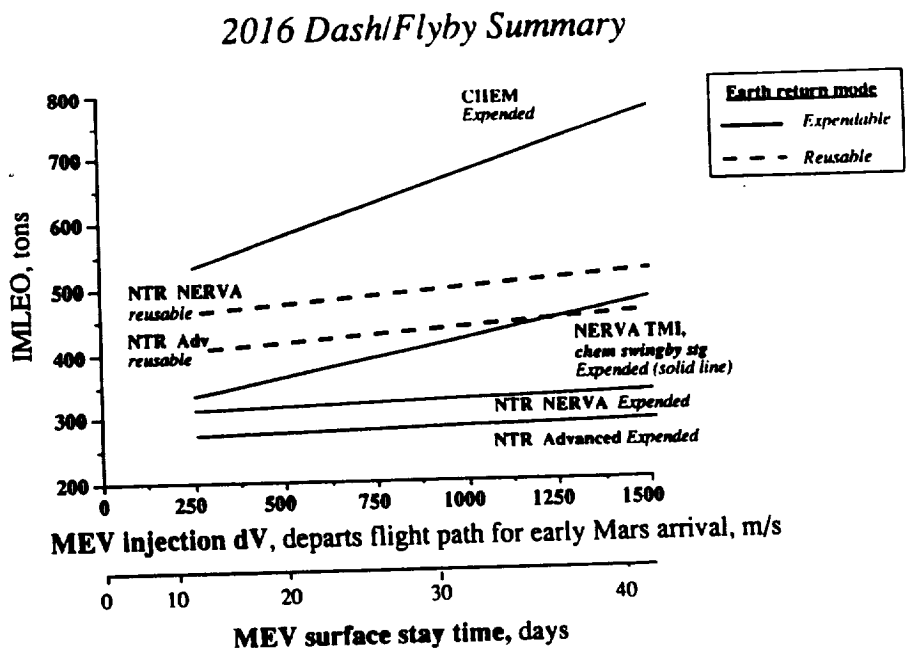
* Asc veh ascends to rendezvous w MTV on hyperbolic swingby path, thus MEV wt is function of swingby Vhp; = 3947 (m/s) for 2020



* Asc veh ascends to rendezvous w MTV on hyperbolic swingby path, thus MEV wt is function of swingby Vhp; = 3947 (m/s) for 2020

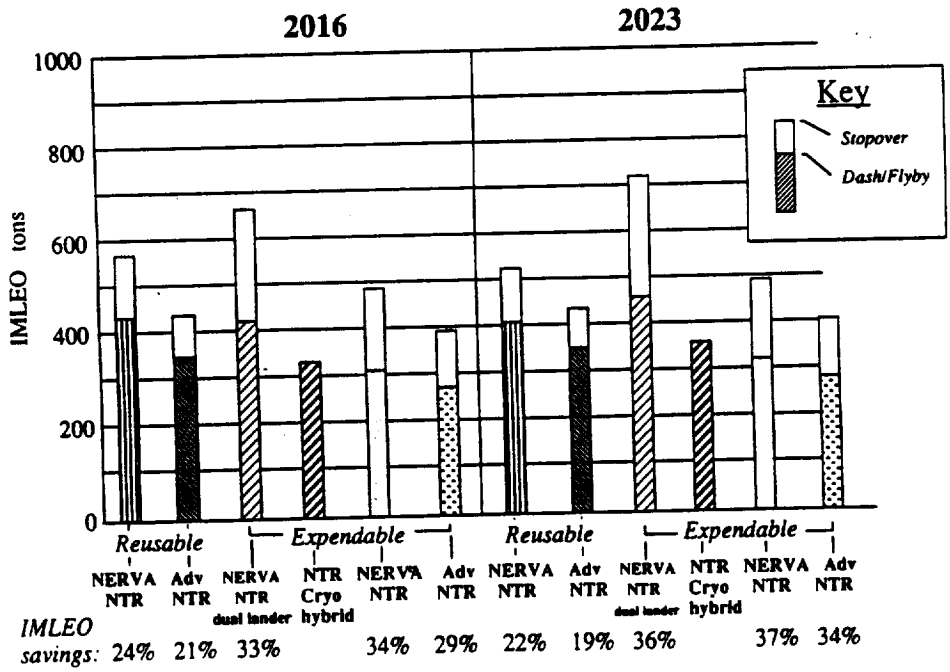
MARS FLYBY MISSIONS WITH SURFACE EXPLORATION (CONTINUED)

FIGURE 4-23



2016 DASH/FLYBY SUMMARY

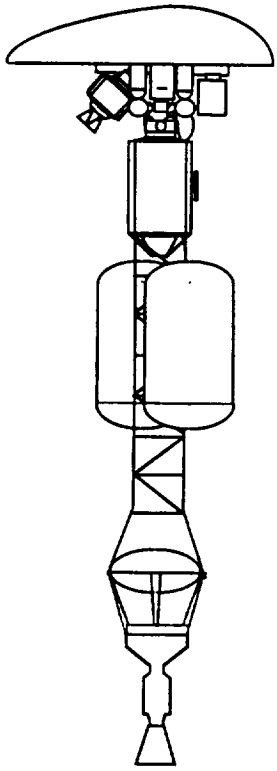
FIGURE 4-24



DASH/FLYBY VS. STOPOVER OPPOSITION MISSIONS

FIGURE 4-25

The 2016 dash/flyby NTR vehicle meet the mission goal requirements (trip time, surface stay time, and delivered payload mass) of the 2016 reference mission with the lowest IMLEO of any concept evaluated. The 324 ton, expendable, 499 day trip time vehicle required only 149 tons of propellant and only 35 tons of non-lander propulsion system inert weight to do the mission. This is for a near term 925 Isp NERVA derivative NTR. A vehicle sketch and summary weight statement is given in Figure 4-26.



Element	*mass (kg)*
MTV crew hab module 'dry'	27567
MTV hab consumables & resupply	5156
MTV crew habitat module total	**32723**
ECCV	**7000**
Powered swingby prop (dV=806 m/s)	5620
Powered swingby stg outbound boiloff	1839
Total Powered swingby propellant	6459
RCS propellant (dV=30 m/s)	603
Outb midcourse correction prop (dV=50 m/s)	386
Inb midcourse correction prop (dV=50 m/s)	323
Powered swingby tank inert	1857
Powered swingby prop & tank total	**9628**
MEV Mars capture & desc aerobrake	17508
MEV asc vehicle	23629
MEV descent stage	23863
MEV surface cargo	25000
MEV total	**90000**
MEV transfer leg supply mod	4000
Mars dep kick stg (asc stg/MTV stg rendez) inerts	3607
Mars dep kick stg propellant (dV=4374 m/s)	16661
MEV kick stg total	**20268**
NTR main stg slowdown burn prop (dV=500 m/s)	3486
NTR engine mass (NERVA; eng T/W=3.5)	9684
NTR engine radiation shield mass	4500
NTR stg tank struts mass	2400
TMI inert H2 tank wt (14% t fraction)	19730
TMI propellant load (dV=4271 m/s)	121180
TMI stage total	**140910**
IMLEO	**324599**

DASH/FLYBY 2016 NERVA NTR EXPENDABLE VEHICLE
FIGURE 4-26

Dash/Flyby Mission Aborts Scenarios - The dash/flyby trajectory provides the lowest risk early mission abort strategy. The early part of the mission is considered that period during the outbound transfer leg occuring before MEV/MTV separation. The excursion to the surface can be aborted by simply not separating the excursion vehicle. Since the spacecraft is on a round-trip flyby trajectory, it will return to Earth.

Two options are available to the MEV for aborting its surface mission once it has separated from the MTV stage. The MTV can itself do a Mars swingby, and thus eliminate the

aerocapture or direct entry maneuver if necessary. Either the MEV descent stage or ascent stage propulsion system may be able to provide enough delta V for swingby flight corrections and later rendezvous with the MTV (this abort profile has not been analyzed). The MEV could also do the aerocapture maneuver without descending to the surface. In that case the MEV could wait in Mars orbit for the MTV to arrive before doing its Mars departure burn.

If only CAB vehicles are considered, then the dash/flyby mode eliminates a high risk maneuver, the MTV aerocapture maneuver. For CAB stopover missions the necessity for the complex aerocapture maneuver means that the heavy MTV module/TEI stage must fly along a very precise path to stay within the aerocapture corridor bounds; if these bounds are violated the crew has no means of escape. The MTV stage in such a case would then either skip out into space, or plummet down to crash on the surface. Once out of the corridor bounds, this stage could not establish an orbit propulsively. In contrast, the MEV aerocapture may entail less risk because this is an aerocapture-to-landing maneuver with somewhat less stringent GN&C requirements. Also, the MEV descent or ascent propulsion stage has enough delta V and T/W capacity to abort the aerocapture by (1) descent abort to Mars orbit, or (2) attempting to alter a skip-out to a hyperbolic rendezvous trajectory. These abort profiles have not been analyzed.
Other advantages of the dash/flyby trajectory are:

- The dash/flyby mode provides very low IMLEOs (approximately 300 tons for the 2020 NERVA NTR expendable vehicle). This level of IMLEO may be low enough to eliminate the need for any new or uprated HLLV beyond that capacity already needed for the Lunar vehicles. The Lunar vehicle IMLEO figures are in the 220+ ton range.

- The dash/flyby mode requires much less total mission delta V variation, and thus less IMLEO variation, over the 2010-2025 time period than does the stopover missions. i.e., the mass requirements are nearly stabilized for all opposition years. The difference in the 'hard' years (2025) and the 'easy' years (2018) is not as pronounced.

- The dash/flyby mode eliminates the apsidal alignment problem inherent to MEV-MTV rendezvous occuring in Mars orbit.

- If CAB vehicles only are considered, the dash/flyby mode requires only one MEV-MTV rendezvous instead of the two required of the stopover missions.

- Since only an excursion module performs the planetary capture, its configuration may be more consistent with the requirements for atmospheric braking than the configuration of an entire MTV spacecraft which provides artificial gravity.

- The dash/flyby mission total propulsion system burn time is less, and thus the NTR reactor fission product buildup would be reduced over the fission product buildup of an equivalent trip time and opportunity year stopover mission.

- The dash/flyby mode provides a superior potential spent reactor disposal trajectory for nuclear propulsion systems, if the nuclear engine is only utilized for the TMI burn. The manned MTV stage returns to Earth after completing the powered Mars swingby (small chemical stage) and unpowered Venus swingbys. The spent nuclear TMI stage would be able to follow this path, but would instead fly past Mars without the necessary burn at Mars and swingby at Venus to return to Earth. In this scenario, the spent stage would follow a distant helocentric orbit without any later Earth vicinity interaction.

The disadvantages of the dash/flyby trajectory are:

- The dash/flyby mode requires the ascent vehicle to rendezvous with the MTV stage traveling on a hyperbolic swingby path instead of rendezvous with the MTV on an orbital path. The ascent vehicle delta V may be as much as doubled over that of a "stopover" mission. This larger delta V requirement necessitates more ascent stage propellant, although without requiring any new ascent stage subsystems. The rendezvous maneuver would require the ascent stage to match the speed of the MTV. The dash/flyby rendezvous would not in any other sense be more difficult.

- The dash/flyby mode requires a small supplemental hab for the MEVduring its partial outbound transfer leg.

- Some options call for small chemical kick stage to put the MEV on an early arrival path to Mars (ahead of MTV system). The kick stage can be dropped before MEV aerocapture, or used again with the ascent stage for the Mars departure burn.

- The ascent vehicle has no backup habitat system in orbit in case surface mission abort is necessary, unless the small MEV supplemental hab is aerocaptured with the MEV and left in orbit for that purpose.

4.3.5 Conjunctions

Conjunctions represent the class of trajectories that can be utilized to provide the lowest total mission delta V's, but also require very long (>900 day) trip times, and very long Mars stay times (>400 days). Conjunction mission total delta V's are not subject to as much variation with opportunity year as does the opposition class missions.

The advantage of aerodynamic braking at Mars is diminished given the much slower arrival Vhp for these trajectories. 900 day plus trip times characteristic of these conjunction class missions would require more development effort in the areas of habitat ECLS systems and artificial-g systems. Stay times on the surface in excess of 400 days necessitate prior establishment of surface habitat facilities or extensive payload for the purpose of supporting a long surface stay. Long stay times in Mars orbit exacerbate concerns about zero-g deconditioning and crew radiation exposure. H2 and O2 boiloff concerns exist for the surface ascent vehicle, and lower Isp performance storable propellants trade favorably for the > 400 day surface stay times.

Trade Objectives - Propulsion/vehicle concepts were not evaluated as to which produced lowest IMLEO. Most of the conjunction mission trades were evaluations of the IMLEO variation with variations in payload weight delivered to Mars.

Trade Results CAP Vehicles - The CAP vehicle represents the propulsion system that could be brought to first flight readiness the earliest, and with the least developmental cost. Two vehicle/mission evaluations typical for this set of trades are referenced below:

a. Human Expedition Mission - The human expeditions consisted of a crew of 3 staying 90 days on the surface. Thirty tons of surface cargo was delivered by the MEV. A vehicle sketch and weight statement in given in Reference 7.

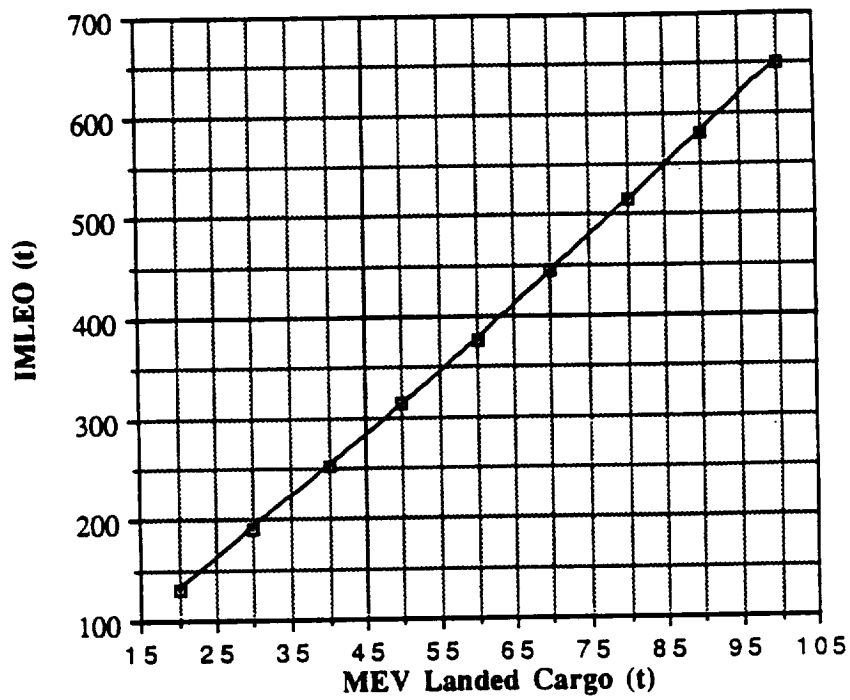
b. Evolution Emphasis Mission - The evolution emphasis mission consisted of a crew of 4. Five tons of surface cargo is delivered by the MEV which utilizes a storable propellant ascent stage to eliminate the boiloff risks associated with cryogenic propellants. This vehicle is part of a mission in which crew habitat facilities have already been emplaced on the surface on a previous flight. A vehicle sketch and weight statement in given in Reference 7.

Trade Results-CAP Unmanned Cargo Delivery Only - The conjunction missions offer an excellent mode for unmanned cargo delivery because of there low delta V's and because when no crew is onboard, short trip times are not as important. Analysis of an unmanned one-way CAP vehicle designed exclusively for cargo delivery was done in order to determine the cost in IMLEO for delivering cargo for these low delta V conjunction missions. Payload delivered by the MEV to the Martian surface is plotted vs transfer vehicle IMLEO in Figure 4-27. Cargo MEV mass vs delivered payload in given in Figure 4-27.

Results-NTR with Multilander Payloads - A NERVA NTR vehicle was designed for a low energy 2005 conjunction mission which is characterized by a 983 day trip time with 482 days a Mars. For this mission a ECCV was taken for crew return to Earth and the vehicle was expended. A nominal crew of 6 was taken. Both MOC and TEI propellant were carried in a single aft tank since the delta V for these burns are quite small when compared to the higher delta V of opposition trajectories. Three different payloads were carried for this evaluation, these being:

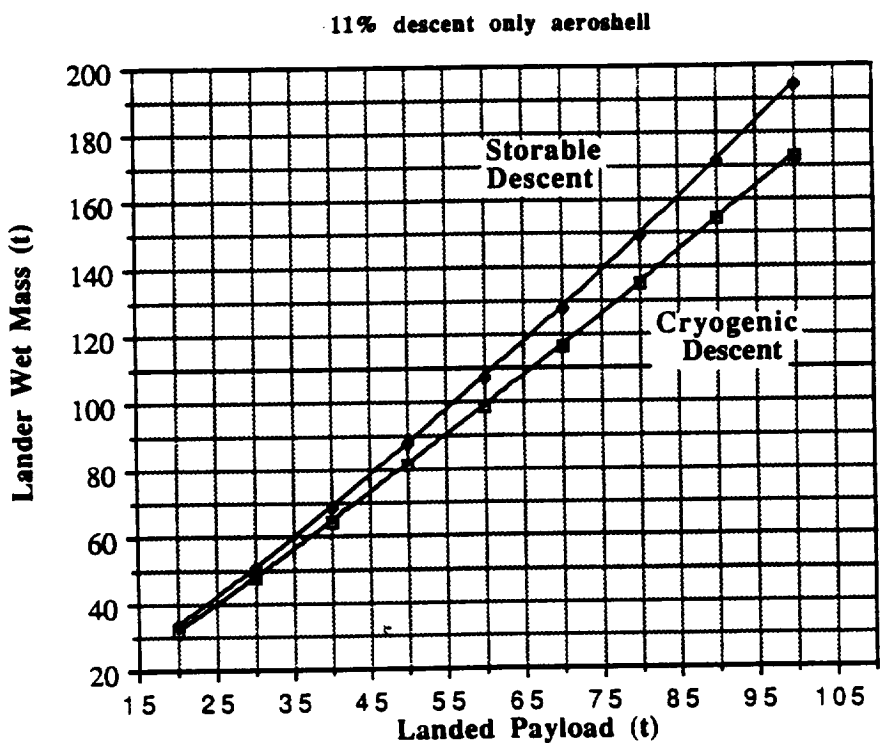
- two reference MEVs (73 tons each) and 30 tons of cargo to Mars orbit
- two mini MEVs (approximately 40 tons each) and 30 tons of cargo to Mars orbit
- two mini MEVs and 10 tons of cargo to Mars orbit

A vehicle sketch and summary weight statement is given in Figure 4-29.



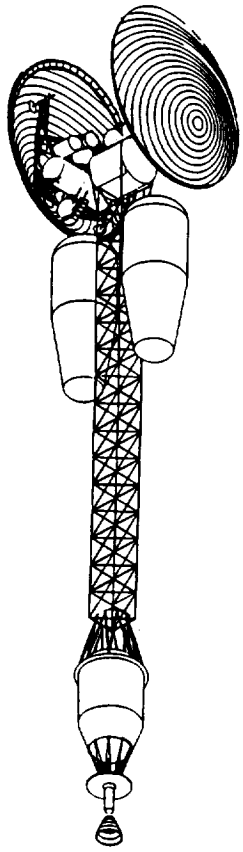
CONJUNCTION CRYO/ALL PROPULSIVE MISSION

FIGURE 4-27



CARGO LANDER SENSITIVITIES

FIGURE 4-28



Element	*2 Ref MEVs 30k cargo*	*2 mini MEV's 30k cargo*	*2 mini MEV's 10k cargo*
MEV total	73118	39752	39752
x 2	**146236**	**79504**	**79504**
MTV crew hab module 'dry'	34775	34775	34775
MTV consumables & resupply	19398	19398	19398
MTV crew habit module total	**54173**	**54173**	**54173**
*MTV frame,propulsion & shield wt	20033	20027	20027
*MTV Art-g added RCS hardware	650	650	650
*MTV Art-g added RCS prop	6348	5134	4588
TEI propellant (dV=4267 m/s)	19648	19185	19042
MOC propellant (dV=863 m/s)	35166	26955	24470
MOC/TEI common tank wt	9669	8543	8202
MTV propulsion/frame/prop total	**91514**	**80494**	**76979**
TMI tanks wt	36911	29732	27563
TMI propellant (dV=1179 m/s)	238510	183220	166520
TMI stage total	**275420**	**212950**	**194083**
ECCV	**8000**	**8000**	**8000**
MTV Cargo to Mars orbit only	30000	30000	10000
IMLEO	**605343**	**465121**	**422739**

* Art-g spinup wt additions

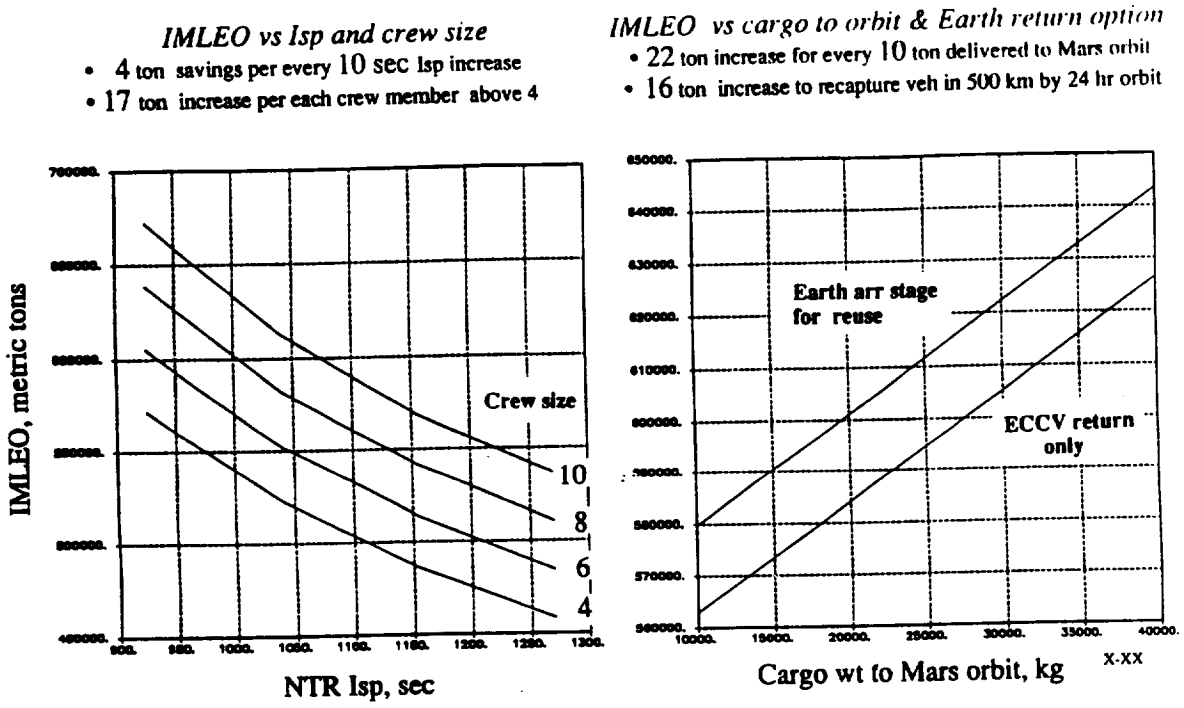
ART-G (MARS-G) NTR VEHICLE FOR 2005 CONJUNCTION MISSION
FIGURE 4-29

For case (1) the following sensitivities were evaluated:

- NTR Isp on IMLEO
- Crew size affect on IMLEO
- Cargo to Mars orbit affect on IMLEO
- Vehicle expended vs vehicle recovered modes

The results indicate that Isp is a very significant factor affecting vehicle IMLEO. For these conjunction NTR vehicles, a one second increase in Isp saves approximately 4 tons in IMLEO, Figure 4-30. Crew size is also a important factor. An increase in the crew size by 2 from 4 to 6, increases the vehicle IMLEO by approximately 35 tons. A 10 ton increase in the amount of cargo mass delivered to Mars orbit (not to the surface) requires 20 tons of additional IMLEO in the form of added TEI and MOC propellant. The IMLEO cost to capture at Earth for later reuse (rather than returning the crew with an ECCV and expending the vehicle) is relatively low. Utilizing the NTR engine for the EOC burn requires only 16 tons of additional IMLEO, which is approximately 3% of the initial total. This figure is a

convincing argument for flying completely reusable vehicles for the all conjunction missions, excepting the unmanned one-way cargo only missions for base buildup.

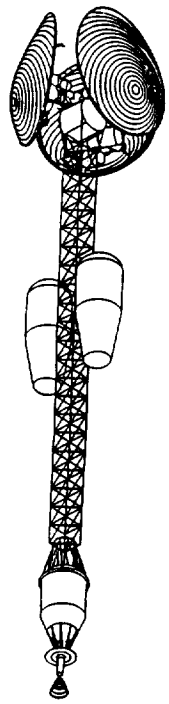


2005 CONJUNCTION NERVA NTR VEHICLE TRADES
FIGURE 4-30

Exploration Emphasis Three Lander NTR Vehicle - A NERVA NTR vehicle was designed for a low energy conjunction mission carrying three landers. Three different payloads were carried for this evaluation, these being:

- Three reference MEVs (73 tons each) and 20 tons of cargo to Mars orbit
- Three reference MEVs and 1 ton of cargo to Mars orbit
- Three mini MEVs (40 tons each) and 1 ton of cargo to Mars orbit

The results of this analysis are shown in Figure 4-31.



Element / *MEV type:*	*Ref*	*Ref*	*Mini*
payload taken & left in Mars orbit:	*20 t*	*1 t*	*1 t*
MEV descent only aerobrake	7000	7000	6000
MEV ascent stage	22464	22464	37366
MEV descent stage	18659	18659	**n/a
MEV surface cargo	25000	25000	5000
MEV total	**73118**	**73118**	**48366**
x 3	**219350**	**219350**	**145098**
MTV crew hab module 'dry'	34790	34790	34790
MTV consumables & resupply	18270	18270	18270
MTV transit science	0	0	0
MTV crew habitat module total	**53060**	**53060**	**53060**
Payload taken & left in Mars orbit	20000	1000	1000
MTV stg frame, struts, & misc inerts	4521	4521	4521
MTV stg RCS (dV=30)hardware & tankage	2034	1991	1834
NTR total engine wt	9684	9684	9684
NTR radiation shadow shield wt	4500	4500	4500
MTV nominal maneuver RCS prop	503	500	470
*MTV Art-g RCS spinup/down prop	7216	6851	5539
Outoound midc correction prop (dV=40)	1889	1784	1345
Inb midcourse correction prop (dV=40)	431	428	402
EOC propellant	n/a	n/a	n/a
TEI propellant (dV=920 m/s)	14280	14202	13404
MOC propellant (dV=1530 m/s)	72066	68104	51544
MOC/TEI common aft tank wt (1)	14564	14026	11711
MTV propulsion/frame/propel tot	**131690**	**126590**	**10450**
TMI propellant (dV=3900 m/s)	274530	259430	196520
TMI side tanks wt (2)	41582	39622	31456
TMI stage total	**316110**	**299050**	**227980**
ECCV	8000	8000	8000
IMLEO	**748200**	**707040**	**535590**

EXPLORATION EMPHASIS THREE LANDER ART-G CONJUNCTION CLASS NTR VEHICLE

FIGURE 4-31

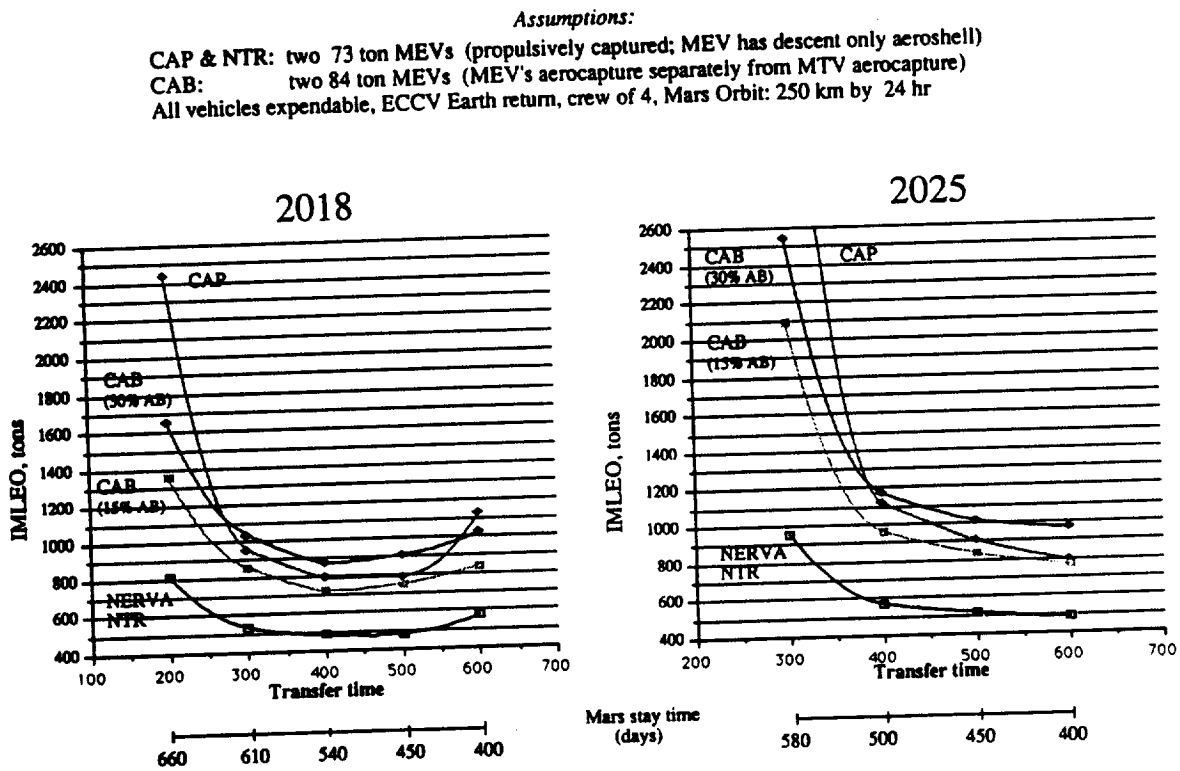
4.3.6 Fast Transfer Conjunctions

Fast transfer conjunction missions are characterized by the short outbound and inbound transfer trip times. The surface stay times for these trajectories are long (400 to 600 days), as well as the mission total trip time, which is on the order of 800 to 1000 days in duration.

Trade Objectives - The objective of this trade was to determine how IMLEO varies with transfer trip time.

Trade Results - The sum of the trip times for the outbound and inbound transfer legs is plotted vs vehicle IMLEO in Figure 4-32. The opportunity years of 2018 and 2025 are plotted because the former represents an 'easy' year (in total delta V) and the latter represents a 'hard' year. Thus all missions in the period of 2010-2025 are bounded by these two years. As Figure 4-32 illustrates, vehicle IMLEO increases dramatically for total transfer trip times less than 300 days for the year 2018. A dramatic increase occurs for

these trajectories with less than 400 days for the harder 2025 opportunity. Mission delta V's and departure dates are listed in Figure 4-33.



FAST TRANSFER CONJUNCTION IMLEO VS. TRANSFER TIME
FIGURE 4-32

4.3.7 Split Sprint missions

Introduction - The split sprint mission is characterized by the use of both a low delta V conjunction trajectory and a short trip time, higher delta V opposition trajectory. The conjunction trajectory is utilized by an unmanned cargo only one way transfer vehicle that leaves before the manned piloted vehicle carrying the crew return TEI propellant and the MEV. After the relatively long trip time is over and the vehicle is established in orbit, the smaller, piloted vehicle departs Earth orbit on a short trip time opposition trajectory. After arrival at Mars and cargo/piloted vehicle rendezvous, a surface mission is done, and the piloted vehicle returns to Earth after transferring TEI propellant from the cargo vehicle while docked in orbit.

	Mars stay time	Transfer trip time	Julian dates TMI	MOC	TEI	EOC	(1) TMI dV	Outb Deep Space burn dV	MOC Vhp	(2) Δ	MOC dV	TEI dV	EOC Vhp	(3) Δ	EOC dV Ellip orbit
2018 'easy year'	660	200	8300	8400	9060	9160	5.320	0	5.470	0	2.790	3.220	7.540	0	2.830
	610	300	8285	8435	9045	9195	4.100	0	3.480	0	1.510	1.990	3.590	0	1.030
	540	400	8270	8470	9010	9210	3.610	0	3.110	0	1.330	1.600	3.290	0	0.940
	450	500	8245	8465	8915	9195	3.540	0	3.020	0	1.140	2.130	3.140	0	0.900
	400	600	8270	8610	9010	9270	3.980	0	4.420	0	1.980	1.970	3.930	0	1.140
2025 'hard year'	620	200	0645	0745	1405	1505	8.750	0	10.850	3.850	7.390	8.750	9.690	0.690	4.170
	580	300	0640	0796	1376	1521	5.750	0	5.370	0	2.700	4.740	6.320	0	2.170
	500	400	0621	0831	1331	1521	4.460	0	3.520	0	1.490	2.150	6.050	0	2.030
	450	500	0600	0860	1310	1550	4.130	0	2.930	0	1.280	1.520	4.030	0	1.180
	400	600	0575	0895	1295	1575	3.730	0	2.470	0	1.060	1.750	3.820	0	1.120

(1) *g-losses not accounted for*

(2) All aerocapture veh's arriving Mars with Vhp>7 (km/s) use cryo propulsion to slow veh down to Vhp=7 for aerocapture

(3) ECCV's arriving Earth with Vhp>9.7 (km/s) use cryo propulsion to slow ECCV capsule down to Vhp=9.7 for entry

MISSION DELTA V DATES: FAST TRANSFER CONJUNCTIONS
FIGURE 4-33

The motivation for this two vehicle mission mode comes from the desire to do the Earth to Mars transfer of the heavy cargo (TEI propellant and MEV) on the lower delta V trajectory (to save IMLEO), and to decrease the crew risk to GCR hazards by reducing the trip time of the piloted trajectory. Unreasonably high IMLEOs prevent the alternative "all-up" single vehicle missions from achieving the short trip times (< one year) attainable by the piloted/no cargo vehicle.

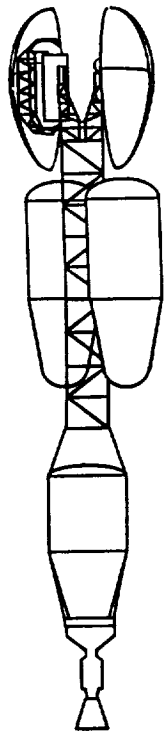
Trade Objective - Several split sprint vehicle weight statements were calculated to determine the weight sensitivity of the piloted vehicle to carrying TEI propellant and the MEV, vs the alternative of having the unmanned cargo vehicle carry these out beforehand. Options to the piloted vehicle payload and propellant capacities are listed below:

- All up mission piloted vehicle - The piloted vehicle carries all necessary Earth return propellant onboard, as well as an MEV. This vehicle is much like the all-up single manned/cargo vehicles utilized on standard opposition missions.

- Split - no MEV piloted vehicle - This vehicle is identical to the all up mission, except the piloted vehicle carried no MEV.

- Split - no MEV, contingency TEI - This vehicle carried only a contingency amount of Earth return propellant; just enough to do an low energy, long duration 'conjunction type' return. The piloted vehicle carries no MEV. A vehicle sketch and weight statement is given in Figure 4-34 for the unmanned, one way cargo only 2011 conjunction NTR vehicle that carries two MEVs and supplemental TEI propellant as cargo. The corresponding 2014 piloted NTR vehicle sketch and weight statement is given in Figure 4-35.

- Split - no MEV, no TEI propellant - All Earth return propellant is to be transferred onto the piloted vehicle from the orbiting cargo vehicle after rendezvous is make in Mars orbit. The piloted vehicle did not carry the MEV.

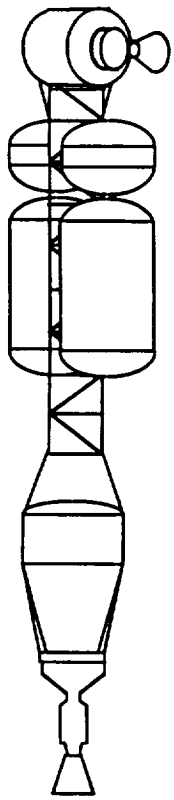


2011 one way transfer 252 day mission

Element	*mass (kg)*
MEVs (2; one cargo, one piloted)	**146236**
TEI prop for transfer to piloted veh (dV=2647 m/s diff in early return dV=4588 and later return dV=1941)	34519
boiloff of TEI transfer prop	8465
Total TEI prop as outb cargo	**42974**
MTV crew habitat module	**0**
MTV frame & aft tank struts	4626
RCS prop (dV=100 m/s) & inert wt	3531
Reactor/engine weight	9684
Radiation shadow shield weight	0
Prop transfer hardware	700
Total	**18541**
Earth Orbit Capture (EOC) prop	0
Trans Earth Inject (TEI) prop	0
TEI tank wt (1)	0
Total TEI stage	**0**
MOC propellant (dV=2532 m/s)	83380
MOC tank (single aft tank)	19462
Total MOC stage	**102842**
TMI propellant (dV=3672 m/s)	182860
TMI tanks (2)	29686
Total TMI stage	**212546**
ECCV	0
IMLEO	**523139**

2011 SPLIT SPRINT ONE WAY CARGO NERVA NTR VEHICLE

FIGURE 4-34



2014 conjunction
586 day mission for early Mars dep (prop transfer in orbit) 919 day mission for late Mars departure

Element	*mass (kg)*
MEV	**0**
MTV crew hab module 'dry'	29317
MTV consumables & resupply	10988
MTV transient science	1000
MTV crew habitat module	**41305**
MTV frame & aft tank struts	4671
RCS prop (dV=50 m/s) & inert wt	3179
Reactor/engine weight	9684
Radiation shadow shield weight	4500
Prop transfer hardware	1000
Total	**23034**
Earth Orbit Capture (EOC) prop	0
Contingency TEI prop (dV=1941 m/s)	25135
TEI tank wt (1)	10799
Total TEI stage	**35934**
MOC prop (dV=2785 m/s)	48631
MOC tanks (2)	12253
Total MOC stage	**60884**
TMI propellant (dV=3886 m/s)	108580
TMI tanks (2)	20041
Total TMI stage	**128621**
ECCV	7000
IMLEO	**296798**

2014 SPLIT SPRINT PILOTED NERVA NTR VEHICLE
FIGURE 4-35

4.3.8 Major Propulsion Element Development Effort Trade

Evaluating vehicles for individual missions has been presented in the preceding sections. The question remains as to what level of technical development effort would be necessary to provide a set of vehicles that could realistically accommodate the variety of mission types and payload combinations covering the complete SEI time frame of 2005 to 2030. In order to estimate the development effort of the propulsion related systems of such a vehicle set, the following trade was done. It is subjective, rather than quantitative in nature. Any early assessment of the effort necessary to the develop the major propulsion system elements of the CAB, NTR, NEP and SEP vehicles is, to a large degree, dependent on the judgment of the evaluator. Even so, this assessment is really only intended to give a indication of a relative ranking among the vehicles; it cannot be considered as a technically precise trade.

The objective was to take a top level look at SEI propulsion requirements from the viewpoint of a total program (2005-2030). This is in contrast to the common practice of evaluating the 1) Lunar, 2) short trip Mars opposition, and 3) long trip, long stay Mars conjunction missions separately.

Each major propulsion element necessary to the accomplishment of three specific missions just mentioned above are listed in Figure 4-36. Four different propulsion scenarios (vehicle sets/combinations) were assessed, in a manner described below, so that a ranking could be assigned among them to see which of the four would require the least 'development effort' to accomplish a complete SEI program. The vehicle sets are as follows:

- Chemical Lunar, chemical Mars opposition (zero-g) & conjunction (Art-g, tether system)
- NTR Lunar, NTR Mars opp (zero-g) & conj (Art-g, veh rotation about its Cg, no tether)
- Chemical Lunar, w NEP Mars opposition (zero-g) & NEP conj (Art-g, tether system)
- Chemical Lunar, w SEP Mars opposition (zero-g) & SEP conj (Art-g, tether system)

For each one of the individual propulsion elements that make up the three mission 'vehicle set', a technology development effort value has been assigned which reflects the estimated effort required to bring any such propulsion system up to flight readiness. As an example, the NTR reactor/engine has been assigned a value of 6 - 'most development effort'. Even though NTR propulsion technology (NERVA class), was demonstrated 20 years ago, new environmental testing concerns and new facilities buildup would increase development effort.

Results - The exclusive use of NTR propulsion for all three missions types produces the lowest estimated development effort score [sum=13], followed by chemical Lunar/chemical Mars [sum =19], chemical Lunar/SEP Mars [sum=18], and chemical Lunar/NEP Mars [sum=27]. Obviously, reducing the total number of distinct propulsion elements necessary to do the missions in order to minimize the overall SEI expenditure on propulsion is important. These scores tend to indicate the profitability in the long run for the use of a single propulsion system for all planetary space transfer missions, even at the expense of mission specific (or individual mission) vehicle optimization.

Moon	zero-g opposition	Mars	artificial-g conjunction	Propulsion Element	Development Effort Factor
				$_{Lunar}$Chemical/$_{Mars}$Chemical sys	
				Lunar	
				1 LTV propul stg	2
				2 LTV aerocapt brake	3
				3 LEV propul stg	2
				Mars zero-g vehicle	
				4 MEV propul stg	2
				5 MEV/MTV aerocapture brake	3
				6 MTV propul stage	2
				7 TMI propul stage	3
				Mars artificial-g vehicle	
				8 Art-g tether system	2
				8 distinct propulsion elements with development factor scores:	**19**
				$_{Lunar}$NTR/$_{Mars}$NTR sys	
				Lunar	
				1 LEV propul stage	2
				Common Lunar & Mars zero-g	
				2 Common LTV/MTV NTR propul stage	6
				3 Radiation handling/monitoring/shield	2
				4 MEV propulsion stage	2
				5 MEV descent heat shield	1
				Mars artificial-g	
				no necessary additions	
				5 distinct propulsion elements with development factors scores:	**13**
				Lunar — **$_{Lunar}$Chemical $_{Mars}$NEP**	
				1 LTV propulsion stg	2
				2 LTV aerocapture brake	3
				3 LEV propulsion stg	2
				Mars zero-g vehicle	
				4 MEV propulsion stg	2
				5 MEV descent heat shield	1
				6 NEP reactor	6
				7 Radiation handling/monitoring/shield	2
				8 Dynamic conversion equip	2
				9 Radiators	2
				10 Electric thrustors	3
				- Separate crew carrier to NEP 'spirial up' altitude	0 (use LTV)
				Mars Artificial-g vehicle	
				11 Artificial-g tether system	2
				11 elements w sum of devel factors scoring:	**27**
				Lunar — **$_{Lunar}$Chemical $_{Mars}$SEP**	
				1 LTV propulsion stg	2
				2 LTV aerocapture brake	3
				3 LEV propulsion stg	2
				Mars zero-g vehicle	
				4 MEV propulsion stg	2
				5 MEV descent heat shield	1
				6 SEP Solar array	3
				7 Electric thrustors	3
				- Separate crew carrier to SEP 'spirial up' altitude	0 (use LTV)
				Mars Artifical-g vehicle	
				8 Artificial-g tether system	2
				8 elements w sum of devel factors scoring:	**18**

Legend: *(1) least development effort; (6) most development effort*
Expected total resources that must be expended for such a propulsion element to achieve flight readiness

MAJOR PROPULSION ELEMENT LIST FOR SPECIFIC VEHICLE SETS

FIGURE 4-36

4.3.9 Advanced Propulsion Trade Summary

Mission goals and requirements may conflict with one another. A trade demonstrating the impact of opposing mission goals can be illustrated with the plot of Mars vehicle IMLEO vs. trip time for the non-swingby opportunities given earlier (see Figure 4-10). A low IMLEO is desired to keep launch costs down; a short trip time is desired to reduce potential GCR risk to the crew. However, the relationship between the two goals as they are implemented into the vehicle design process almost appears to be one of mutual exclusivity, i.e. the two goals of low IMLEO and short trip time drive the design in diverse directions. To complicate the difficulty, neither an upper bound indicating an 'unacceptably high' IMLEO nor a 'maximum allowable' trip time is as yet definable. What can be established from this example is just how much IMLEO varies with variations in trip time for each of the vehicle concepts. In this way a methodology for quantifying this IMLEO trip time relationship is established as a means for accessing the cost (in vehicle weight) of reducing crew exposure time to GCR via reducing trip time. This point has been given as a means of showing that the 'yardsticks' of evaluation are often interdependent.

4.3.9.1 Preferred Concepts

Several mission categories do exist, and have already been presented, wherein certain vehicle/propulsion systems demonstrated superior performance in that category. Several of these were assessed as preferred concepts for particular mission types. Identifying and defining these areas where particular concepts differentiated themselves as being superior was a key emphasis of the STCEAM study trades. For given niches, or categories of mission scenarios, an individual 'preferred system concept' proved to be the most effective in meeting the primary objective(s) or requirement(s) for the mission type under investigation. These concepts are given in Figure 4-37.

PRIMARY goals - >	*Minimum Up Front Costs*	*Short Transfer Trip Times*	*Minimum IMLEO*
SECONDARY goals			
- > *Early First Flight*	**CAP** (conj) **NERVA NTR** (opposition)	N/A	**NTR** or **SEP**
Frequent Trips	**NERVA NTR** *(Evol to Adv NTR)* or **SEP**	**NTR** or **NEP**	**NEP**
Long Surface Stays (Conjunctions)	**CAP**	**NTR** or **SEP**	**NTR** or **SEP**

PREFERRED CONCEPTS
FIGURE 4-37

4.3.9.2 Concept of Maximum Utility: NTR

Although no overall 'winner' is presented as the system to accomplish all the goals of a comprehensive SEI program, an overall assessment was made as to which of the preferred concepts was best suited to cover the broadest range of the various SEI mission types. The NTR propulsion system/vehicle concept was assessed as that system of maximum utility.

a. The NTR vehicle was frequently assessed as a preferable concept for several of the various mission scenarios analyzed.

b. NTR system can provide both an early first flight date, and a favorable pathway for evolving to higher performance in the out years, without extending beyond subsystems enhancements to the early system designs. The evolution to high temperature capable

carbide fuel elements was identified as the subsystem pathway for eventually achieving the 3100K temperatures and the 1000+ sec Isps associated with high performance advanced NTR systems.

c. Vehicle flexibility to accept variations in performance requirements is good; NTR readily adapts to changes in mission delta V's since these are implemented as changes in propellant tank capacities.

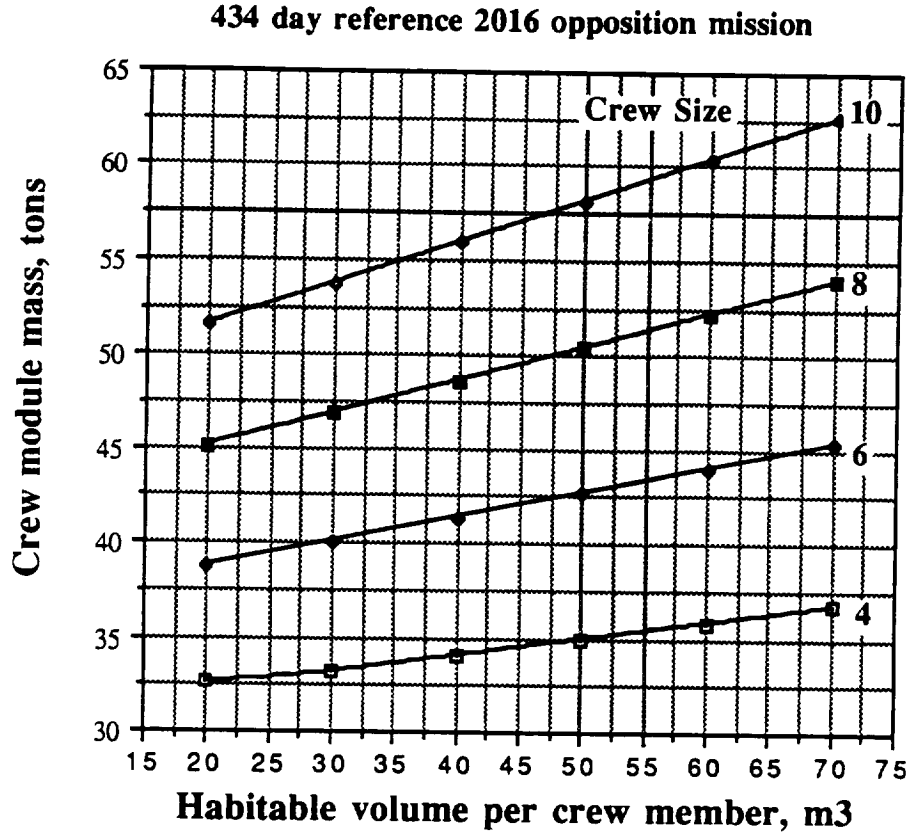
d. The use of an all propulsive NTR system allows for total elimination of high energy aerocapturing technology.

4.4 System Trades Common to All Propulsion Concepts

4.4.1 Crew Module Weight vs Crew Size vs Habitable Volume Trade

The objective was to quantify the relationship between crew habitat module weight, crew size and habitable volume per crew member (free volume). The data presented in the parametric chart (Figure 4-38) applies to the crew habitat module used for all the reference vehicle trades and advanced propulsion trade vehicle weight statements. A detailed description of this crew habitat module is discussed in Section 2.

The module used for a crew size of 4 (reference) was 7.6 meters in diameter and 9 meters in length, and was sized to have a nominal habitable volume (free volume) per astronaut of approximately 50 m3. The habitable volume, or free volume, is defined as that internal volume in the module that a crew member can occupy, such as crew quarters, hall ways, work areas, bathrooms, medical station, etc; any place that some one can sit or stand or lay is considered as habitable volume. Habitable volume per crew member is simply the total habitable volume divided by the number of crew.



TRANSFER VEHICLE CREW HABITAT MODULE SENSITIVITIES
FIGURE 4-38

4.4.2 Reference Crew Habitat Design Trade

It is known that the needs of the crew dictate the amount of habitation volume that would be considered necessary, or desirable for a long duration Mars mission, although the actual amount is hard to quantify. Payload limitations are dictated by the performance of the propulsion system and the total propellant load. This limitation would restrict the transfer module system to some maximum allowable weight for a given mission. Minimizing module weight is seen as very desirable, and beneficial to overall vehicle performance from the propulsion system designer's standpoint, while providing the crew with generous accommodations is desirable from an operational standpoint. Some compromise has to be made between the two. Realistically assessing the needs of the crew, be they real or perceived, is a task of major proportions, requiring input from many disciplines outside that of spacecraft design and operations engineering. Relevant data and experience will be

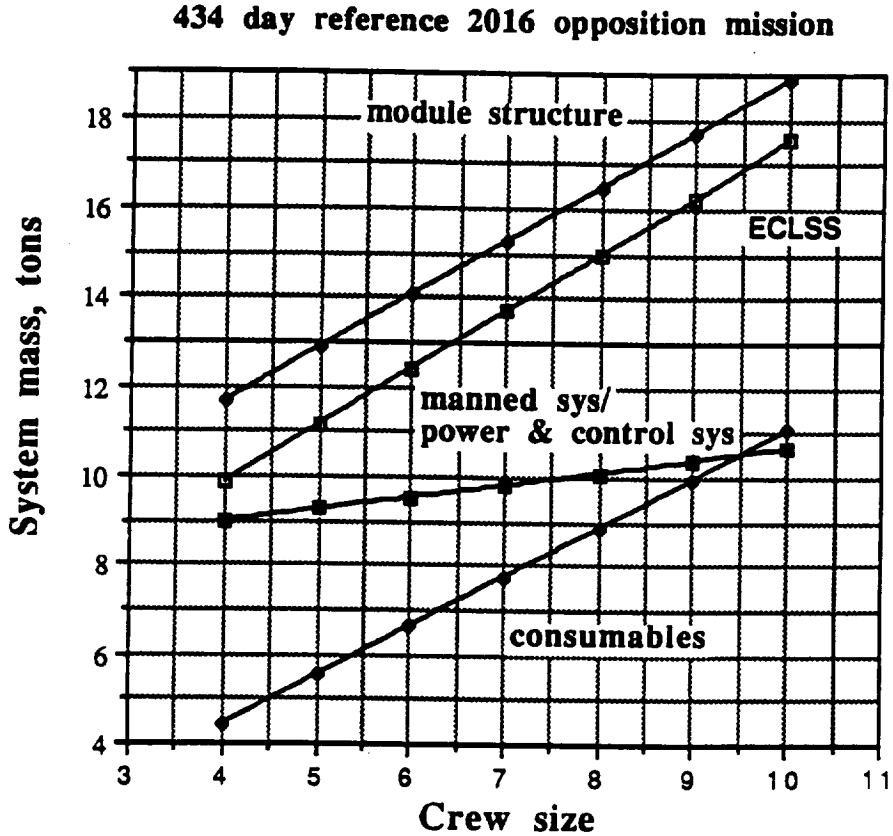
gained from the space station missions, especially in the area of health risks associated with living for long periods in a weightless environment. Current information that could be considered applicable comes mainly from past space missions, such as Skylab, and from space system analogs, which include submarine operations and Antartic winterover expeditions. Persons involved in these activities share with astronauts some of the problems associated with long duration confinement in isolated environments.

The total volume of the crew module includes the habitable volume as well as the internal volume taken up by the onboard structure, equipment, and consumables. In order to correlate total vehicle weight variations with the habitat module size and weight variations, crew number was varied from four up to ten, and the habitable volume per crew member was varied from 20 to 70 cubic meters. Also of interest was the weight contributions made to the total module from the ECLS systems, the module structure, manned systems, health care/recreation systems, galley, and crew quarter sections, among others.

The crews habitat module is an environmentally protected enclosure intended for long duration crew activity and habitation functions (eating, sleeping, recreating, relaxation, medical operations, work activities, EVAs, navigation, etc.). The habitat module contains all the systems necessary to provide a shirt sleeve, productive environment conducive to long duration space mission task. The reference 2016 opposition mission was 434 days in duration, and the module weights calculated reflect the consumables and contingency supplies necessary to this duration mission. Preliminary weight estimations have been developed.

The relationship between the crew module mass, crew size, and habitable volume is shown in Figure 4-38. For an increase in crew size by one, the resulting increase in MTV crew habitat module mass is approximately 10%, due primarily to the increased consumables, ECLSS, and other module subsystems necessary to accommodate the extra crew member. Increases in a module's habitable volume, however, only involves increased living space (not additional or more robust ECLSS subsystems) which can be accommodated solely by slight increases in the overall crew module external structure. An increase of allowable internal habitable volume from 40 m^3 per astronaut to 50 m^3 only increases crew habitat module mass by approximately 1%. The conclusion reached is that module weight is much more sensitive to crew size changes than to personnel living space requirements.

The weight contribution to the module from the ECLS system, structure, manned systems, power and control systems and consumables is shown in Figure 39.



CREW HABITAT MODULE SYSTEM SENSITIVITIES
FIGURE 4-39

The environment control and life support system (ECLSS) weight and volume estimates are taken from individual NASA/Boeing Space Station Freedom habitation module system and subsystem design estimates. Estimates of individual subsystems were collected and then modified for the Mars transfer mission. A quick assessment of the planned Space Stations individual ECLSS systems capability, capacity, and longevity was done to estimate the additional robustness, reliability and maintenance measures, that have to be added to the more difficult mission requirement. The lack of a periodic resupply interval differentiated the SSF system from the MTV system; system and subsystem weight increase estimates reflected this difference. The ECLS system serves the following functions:

- Maintains cabin atmosphere with regard to temperature, humidity, pressure and composition

- Provides/processes water for potable and hygiene systems
- Processes/stores biological wastes
- Supports EVAs
- Provides cooling of equipment, fault detection, and suppression of fires.
- Provides for thermal conditioned storage of food and pharmaceuticals

The life critical systems within the ECLSS were configured to meet the reliability/safety requirement of dual fault tolerance. The ECLSS is interfaced to the data management system architecture for control/monitoring and caution/warning systems. Issues of concern were the following:

- Dependant failures and failure propagation
- Maintenance policy and implementation
- Redundancy, resupply levels, backup systems
- Spares quantity and accessibility
- Continuous/intermittent operation modes

Items for which weight and volume figures were calculated are listed below:

- Potable water system
- Hygiene water system
- TCS/THC system
- Atmospheric Revitalization System
- Urine processing
- Atmosphere conditioning system
- Fire detection, isolation and suppression
- Fault detection system
- EVA supplement system
- Filters
- Mounting and distribution hardware for these systems
- Film locker
- Leak detection system
- Refrigerator/freezer
- N_2 leakage makeup
- Food and packaging
- Emergency air repressurization

Items of the structure for which weight and/or volume figures were calculated are listed below:

- Primary shell
- Secondary shell
- End cones
- Floor/support
- Bulkhead
- Radiation shielding
- Airlocks
- Hatches
- Berthing hardware
- Equipment mounting
- Consumables storage
- Tunnels
- Mechanisms
- Other storage
- Utilities
- Spares

Items of the manned systems for which weight and/or volume figures were calculated are listed below:

- Health care
- Recreation
- Exercise
- Crew quarters
- Personal effects and storage
- Galley
- Personal hygiene
- Full body wash
- Laundry
- Dishwasher
- Waste management system
- Trash compactor
- Media/library

- Photography
- Science station equipment and supplies
- Lighting
- EVA suits

Items of the power/control system for which weight and/or volume figures were calculated are listed below:

- Vehicle control work station
- Data management systems
- GN&C systems
- Internal power system
- External power system
- Power distribution system
- Communication systems
- External RMS

The relative impact of these systems to total habitat module weight is shown in Figure 4-39.

4.4.3 Vehicle Artifical-g Trade

Artificial-g incorporation to a NTR conjunction vehicle is illustrated in Figure 4-40. This data shows the impact to IMLEO for a 2005 conjunction NTR vehicle. The long NTR vehicle spins about its CG to produce artificial gravity at the ends. RCS propellant is used to spinup the vehicle to the required RPM and to despin the vehicle back to the zero-g stage for propulsive burns. Section 2 of this report contains a complete discussion on artificial gravity issues.

4.5 MEV/LEV Trades

Detailed information on the reference MEV configuration issues are provided in Section 2. Important lander trade categories are shown in Figure 4-41 and are discussed below.

4.5.1 Reference MEV Propellant Trade

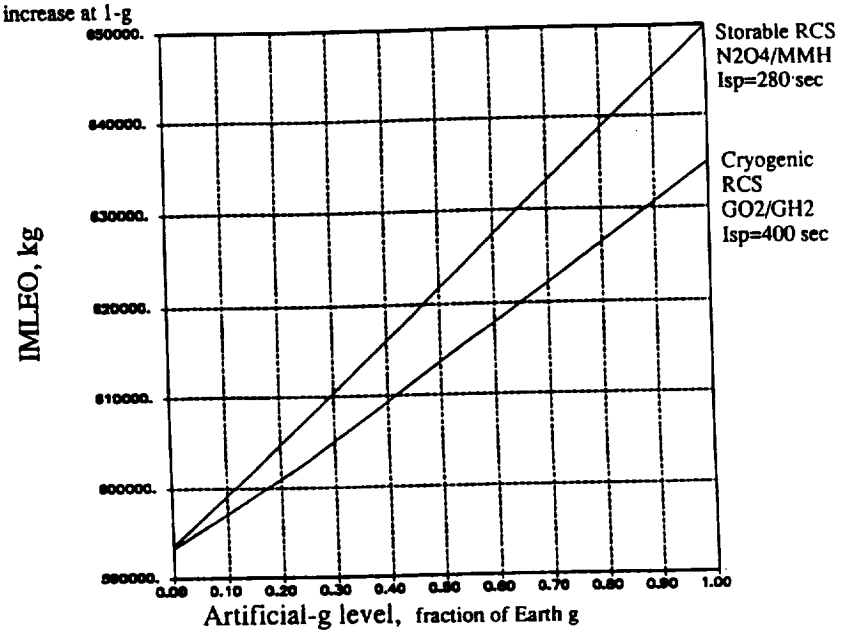
Trade Objective - The inhouse Boeing Vehicle Synthesis Model was used to generate parametric data to evaluate the effect of propellant choice on ascent vehicle total mass. The ascent vehicle under evaluation was the reference design with a crew size of 4. The following five propellant combinations were examined:

1. Storage N2O4/MMH pump feed (Isp = 340 sec)
2. Metallic gel storable N2O4/MMH/A1 (50% by weight), pump feed (Isp = 366)
3. Cryogenic/storable LO2/MMH/A1 (35% by weight), (Isp = 374 sec)
4. Cryogenic LO2/Methane, (Isp = 380 sec)

5a. Cryogenic LO2/LH2, Advanced space engine, 200:1 nozzle (Isp = 460 sec)
b. Cryogenic LO2/LH2, Advanced space engine, 400:1 nozzle (Isp = 475 sec)

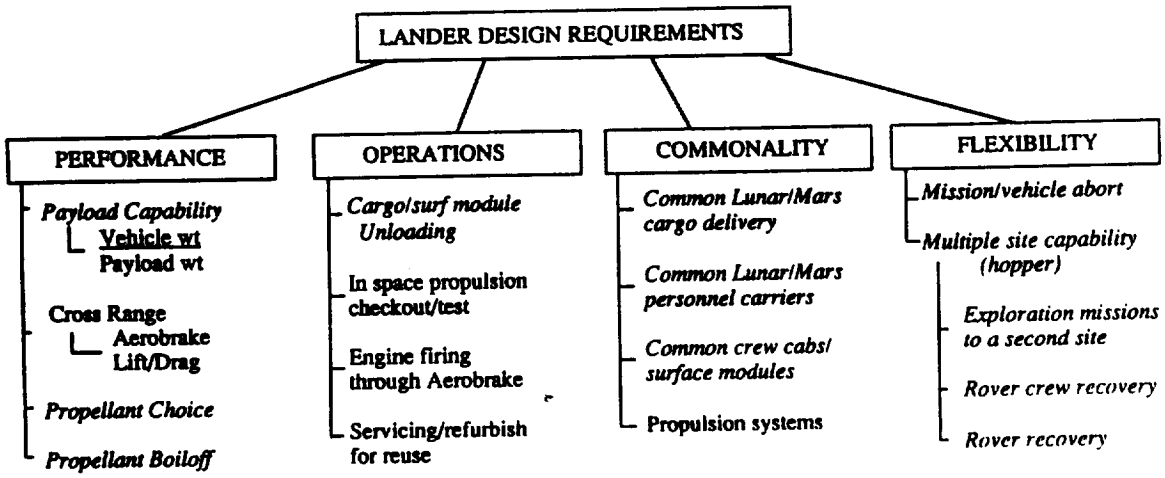
Results - The ascent vehicle propellant boiloff rates were calculated using Boeing's "Crystore" program which is based on empirical data documented in several MLI studies of the late 1960's to mid 1970's. Results are shown in Figure 4-42, and indicate that even for stay times as long as 600 days, the boiloff of cryogenic LO2/LH2 can be kept low enough with advanced passive thermal insulation (two inches of MLI, vacuum jacket, and vapor cooled shields with an outboard vent temperature of 50K) to keep the total ascent stage mass less than that for an identical ascent stage utilizing the storable N2O4/MMH (Isp = 340) combination as propellant. This assumes that no loss of thermal insulation performance takes place in LEO, during the Earth-Mars transfer leg, and while on the Martian surface stay due to unexpected insulation and/or tank penetrations or any insulation and/or tank penetrations or any insulation system degradation.

RESULTS:
- 25 t IMLEO increase of 1-g over 1/3 g (Mars-g)
- 6 t IMLEO increase for storable biprop RCS (Isp=280 s) over GO2/GH2 RCS (Isp=400 s) at Mars-g
- 15 t increase at 1-g



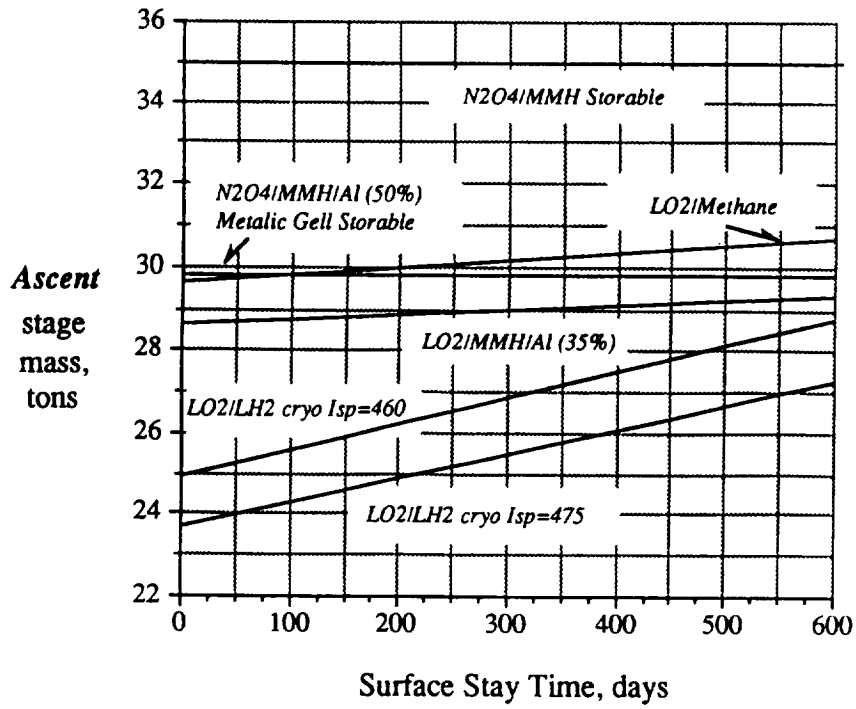
2005 CONJUNCTION NERVA NTR VEHICLE ART-G TRADE

FIGURE 4-40



MARS/LUNAR LANDER TRADE CATEGORIES

FIGURE 4-41



REFERENCE MEV PROPELLANT TRADE

FIGURE 4-42

4.5.2 "Mini" MEV Propellant Trade

The mini MEV was designed to be one half the weight of the reference 84 ton MEV, so that two surface missions could be carried out for the same mass. The mini MEV concept provides much less cargo and stay time capability than the large reference MEV. The mini MEV is characterized by its 6-10 day stay capability and minimum cargo allotment.

The five propellant combinations mentioned above for the reference MEV trade were also evaluated for the mini MEV.

Results - The cryogenic propellant case provides for the minimum mass MEV system for all surface stay durations of 300 days or less. At 300 days, the additional cryogenic propellant carried onboard to make up for the boiloff losses, causes the total vehicle weight to increase to that weight estimated for a MEV utilizing the metallic gel storable propellant (Isp=366). Aluminum added to the MMH fuel enhances engine Isp, but would necessitate some engine modifications needed to compensate for the erosion caused by the small aluminum particles, which would be suspended in a gelled propellant form. Not until the required surface duration extends to approximately 600 days is the cross over point reached. At that point, the cryogenic mini MEV system is as heavy as the mini MEV system

utilizing the standard storable N2O4/MMH propellant without the metallic additive. Results are presented in Figure 4-43.

4.5.3 MEV Aeroshell Trade

The objective of this trade is to illustrate the advantage gained by utilizing aerobraking for the Mars descent maneuver. The vehicle reference MEV was compared to a MEV that utilized an all propulsive (engine thrust) descent to the surface. For that case no aeroshell was used, however a descent heat shield was used to protect the vehicle from high entry heating rates. This descent heat shield was assumed to be 5% of MEV mass. Both MEVs carried a 25 ton cargo payload to the surface. The reference MEV utilized a 7 ton aeroshell that dropped off of its own weight after the descent engines were fired during terminal descent. The difference in total weight between the two is shown in Figure 4-44. Both MEVs descended from a 250 km by 1 sol parking orbit, (total descent delta V is 5519 meters per second). For the reference case 4557 m/sec is taken out by the aerobrake, while the remaining 962 m/sec velocity reduction is done with engine thrust. For the all propulsive MEV, all 5519 m/sec of delta V are taken out by engine thrust. Almost 140 tons of additional descent propellant is required if a descent aeroshell is not used.

4.5.4 MEV Ascent to Altitude Trade

The objective of this trade is to illustrate the MEV weight cost of ascent to a circular parking orbit of varying altitudes (Figure 4-45). The MEV carried a crew of 6 and utilized a descent only aeroshell.

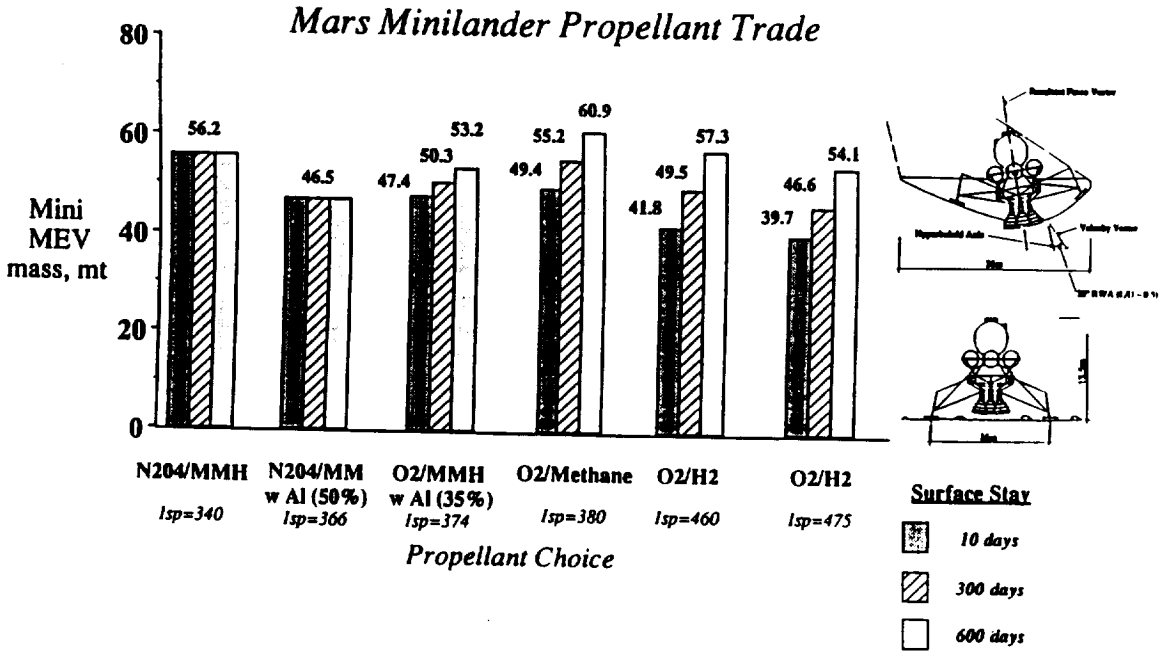
4.5.5 Cargo MEV Trade

A descent only cargo MEV was evaluated by varying cargo load delivered to the surface as well as propellant choice for the descent engines which fired after most of the descent delta V was taken out by the aerobrake. The cost in lander weight for these varying cargo loads was shown previously in Figure 4-28.

4.5.6 MEV Surface vs Excursion Habitats Trade

Objective - The objective was to determine which is lighter: sizing the excursion vehicle crew cab for the entire surface duration or using a one day excursion vehicle cab along with

a separate surface habitat. This trade was conducted to determine the stay time after which it is lighter to provide a separate surface habitation module which remains behind, than to embed this function in the excursion vehicle crew cab, which ascends back to orbit. The operating mode for this study is an excursion scenario (including "campsite" options), with no permanent base available.



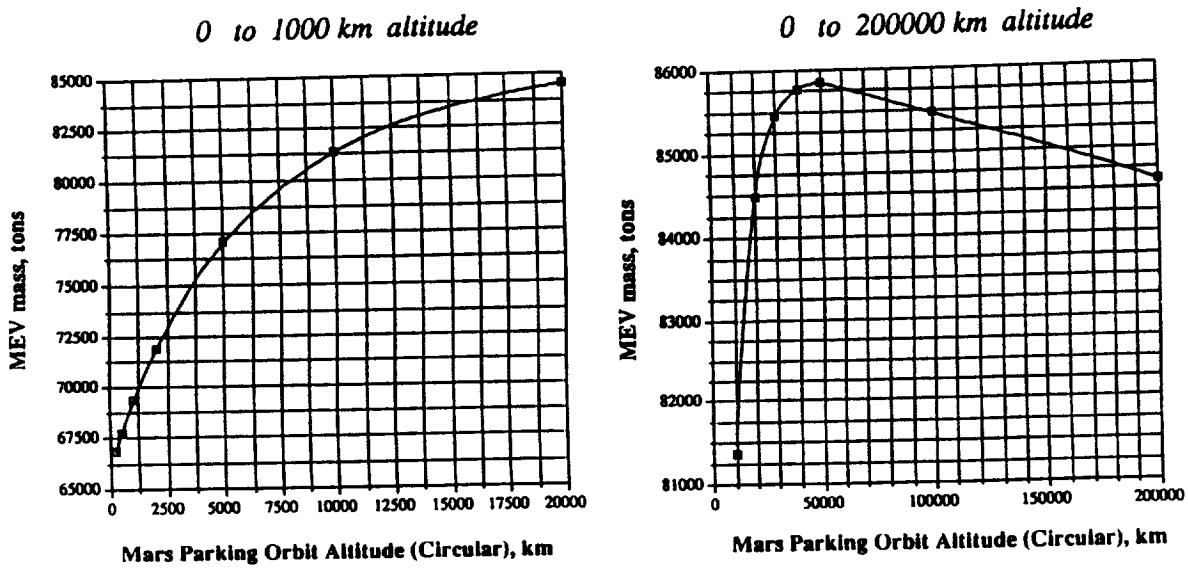
MARS MINI-LANDER PROPELLANT TRADE

FIGURE 4-43

Element	*Reference MEV [descent only aeroshell]*	*MEV w/o Aeroshell [all propulsive descent]*
Ascent cab	3478	3478
Ascent stage inerts	3173	3173
Ascent RCS prop	172	172
Ascent propellant	15639	15639
Total ascent vehicle	**22462**	**22462**
Surface cargo	25000	25000
Descent stage inerts	5422	12924
Descent RCS propellant	1173	3763
Descent propellant	12061	153710
Base heating shield (5%)	n/a	10893
Descent only aeroshell (15%)	7000	n/a
Total descent stage	**50656**	**206290**
MEV total	**73118**	**228752**

MEV WITH AND WITHOUT AEROSHELL TRADE

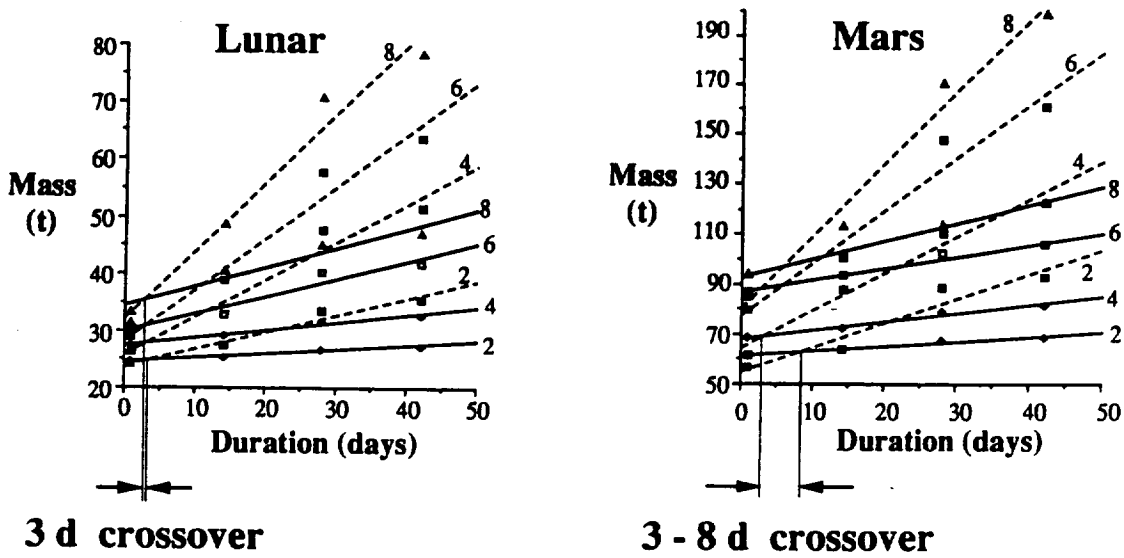
FIGURE 4-44



MEV WEIGHT VERSUS MARS CIRCULAR PARKING ORBIT ALTITUDE
FIGURE 4-45

Crew sizes of 2, 4, 6, and 8 were evaluated. 2 kg food/person/day and 6 kg water was used in the calculations. An open ECLSS was assumed with three repressurizations possible. Other assumptions included carrying 2000 kg of science equipment (left on the surface) and 500 kg return payload (samples). Propellant tanks were sized for a 25 ton load except when in excess of 25 tons is needed. A 5% fuel contingency, and 2% boiloff were assumed. For the Lunar cab-plus-hab case, the ascent vehicle returns to LLO without landing legs and some structure. In the Lunar cab only case, the entire vehicle returns to LLO for reuse in LOR mode. For the Mars case, separate ascent and descent stages were used for all cases, as well as a 20% aerobrake for entry, jettisoned prior to landing.

Results - A separate habitation module is the preferred solution for both Lunar and Mars surface stays exceeding a few days, as shown in Figure 4-46.



EXCURSION VEHICLE CAB DURATION TRADE DATA
FIGURE 4-46

4.5.7 Common Mars/Lunar Lander Trade

The objective here was to design a common Mars/Lunar lander that could operate either as an unmanned cargo carrier, or as a piloted vehicle carrying personnel and cargo to and from the surface of Mars or the Moon. The primary design criterion was to keep the complete propulsion stage identical for all vehicles. The vehicle inert weight is a function of several items, foremost among these being the propulsion requirement to provide a minimum vehicle T/W ratio of approximately 1.6 for the landing and ascending burn phases while retaining a single engine out margin. Other important parameters include the stage structural frame and landing leg weight, both of which are a function of the load that each must support. Stage inert weight is also a function of the number of tanks and their size as selected for both the MPS and RCS propellant loads which include boiloff allowances for surface stay time in the case of the piloted sorties.

For this preliminary design task, the decision was made to size the common inert stage based on the piloted Mars lander mission with a crew of 4 and with a 30-day surface stay time. A total cryogenic propellant load of 31 tons was necessary for this case. To provide a T/W of 1.6, three 30k lbf advanced cryogenic engines (Isp=475) were selected. One large fuel tank, and four small oxidizer tanks were selected and sized to hold the 31 ton maximum propellant load. Adding tank insulation, meter shields, vapor cooled shields, propellant line weights, frame structure, landing legs and a mass growth allowance to the engine/tank set produced a total stage inert weight of 7.4 tons. The following comments will serve as a brief description for the six vehicles given summary weight statements in Figure 4-47.

Commonality across the vehicles is realized by using a common propulsion stage: identical engines, structure, MPS and RCS tank sets, etc.as mentioned earlier. However, differences do exist in the following areas: aerobrakes, cargo load, tank propellant loads (off loaded tanks in the case of the Mars cargo and 30 ton Lunar cargo vehicles)

Element *Stay time*	**Mars** *Cargo* *(desc only)* *n/a*	**Mars** *Manned* *(single stg)* *365 days*	**Mars** *Manned* *(single stg)* *30 days*	**Lunar** *Cargo* *(desc only)* *n/a*	**Lunar** *Cargo* *(desc only)* *n/a*	**Lunar** *Manned* *(single stg)* *6 months*
Ascent cab	0	4847	4847	0	0	4847
Stg inerts	*7396	8656	*7396	*7396	*7396	*7396
Aeroshell	7500	7500	7500	n/a	n/a	n/a
Surf Cargo	30000	500	500	30000	**45000	**23000
Asc prop	n/a	28001	23389	n/a	n/a	7193
Desc prop	8125	9188	7713	21502	30126	24277
RCS prop	1704	2121	1854	705	988	914
Total wt	**54725**	**60810**	**53227**	**59602**	**83510**	**67627**

Manned: crew of 6, immediate transfer to surf hab module
Stay time: for Asc stg propellant boiloff calculation only
** *Maxium surface cargo load for these lunar cases when all tanks are full (not off loaded as in column 4)*
* *Commonality across vehicles is realised by using a common propulsion stg: identical engs, structure, MPS & RCS tank set differences consist in: aerobrakes, cargo load, tank prop load (off loaded tanks in columns 1 & 4)*

COMMON MARS/LANDER VEHICLE - CARGO AND MANNED VERSIONS
FIGURE 4-47

Column 1 Mars cargo: 7.5 ton aeroshell & 30 ton cargo to the surface requires only 8.1 tons of descent propellant (1/4 of available tank volume)

Column 2 Mars piloted with 1 year surface stay: 7.5 ton aeroshell, 4.8 ton ascent cab (crew of 6), 500 kg of cargo, and 1 year boiloff allowance requires 37 tons total descent and ascent propellant, which is above the common stage tank set load capacity, thus excluding it from the group sharing the common propulsion stage.

Column 3 Mars piloted with 30 day surface stay: requires 31 tons of propellant; this load was selected as the capacity for sizing the tanks for the common vehicle inert stage - used for all except case (2).

Column 4 Lunar cargo: 30 tons to the surface implies off loaded tanks: only 21.5 tons of descent propellant required (tanks 2/3's full)

Column 5 Lunar cargo: 45 tons to the surface are possible if the tanks are filled to their 31 ton maximum capacity.

Column 6 Piloted Lunar: ascent cab with 23 tons of surface cargo are possible if tanks are filled to their 31 ton maximum capacity.

For the Mars missions a 7.5 ton aeroshell decelerates the cargo for the majority of the descent delta V. Part way through this aerobraking phase the nozzles of the engines are extended through the brake doors. Supplemental braking is provided by these engines until the brake drops off from its own weight, after which the engines alone provide terminal descent to final touchdown. The Lunar cargo case is identical to the Mars cargo case except for the absence of the aeroshell. For the manned sorties the lander functions as a single stage descent/ascent vehicle. The entire vehicle ascends to orbit leaving behind only the surface cargo for the piloted Lunar case, while leaving the cargo and the landing legs for the piloted Mars case.

4.5.8 Reusable MEV Concept Trade

The concept is for a totally reusable MEV and assumes the availability of Mars surface *in situ* oxygen production for its ascent burn O2. The small landing aeroshell has 30% less aerobrake area than the reference 2 stage expendable MEV aeroshell. This concept only 'works' with Mars surface oxygen being available. The vehicles maximum gross mass is about the same as a traditional staged MEV without the use of Mars oxygen. Details may be found in Section 3.

4.5.9 Secondary Requirements influence on MEV/LEV Design

The 84 ton reference MEV (with 25-ton surface habitat) and the 40 ton 'mini' MEV were developed early in this study, each especially suited to its descent and ascent task. However, viewing a LEV or MEV system as merely a vehicle for descent to, and ascent from, a planetary surface without due consideration given to other secondary requirements of significance would be inadequate. As a complete man rated transportation element, a lander may be used in a secondary capacity to enhance some other phase of the mission. In assessing how the design might provide for capabilities beyond that needed for the primary descent/ascent task, the following items were considered:

- Lander design and how it affects cargo unloading to the surface
- Techniques for cargo jettison during descent abort situations
- LEV capability enhancement to allow it to serve as an backup system to the LTV for Lunar departure and return to Earth
- Secondary capability for short suborbital flights to distant sites, for additional exploration, or recovery of off site failed systems.

4.5.9.1 Lander Cargo Off-Loading Trade

The impact of cargo unloading operations and/or restrictions were left unaddressed until later in the study. In retrospect it was determined that cargo off-loading operations play a very important role in insuring a truly successful lander mission. Assessments indicated the importance of the following:

a. Ease of cargo unloading is imperative to successful surface operations. The MEV/LEV should provide for a functionally simple and mechanically uncomplex capability

for off-loading cargo, without placing on the entire surface system the requirement for either an unnecessarily heavy surfaced based cargo unloader, or for a unloading procedure that is overly complex, or time consuming.

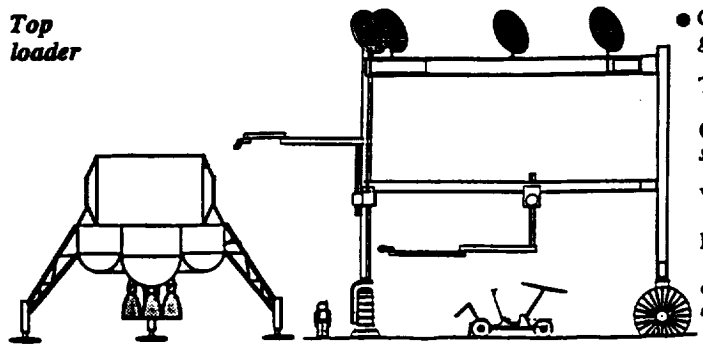
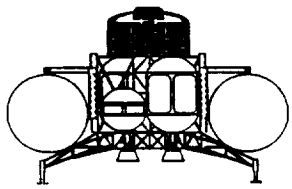
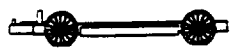
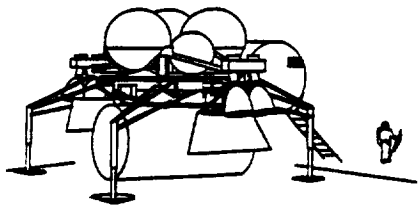
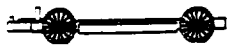
b. A single Lunar or Mars lander concept should be able to effectively handle a range of cargo loads as well as varying personnel and cargo combinations.

Cargo Off-Loading: Current NASA Configurations - An evaluation of the current NASA baseline LEV design, as well as some of its contemporaries in the literature, revealed that those designs did not provide simple off-loading capability, but actually drove the surface based systems to provide relatively large surface unloaders. This was seen as a potentially severe burden to surface operations.

Lander designs evaluated in the Cargo Off-Loading Operations Trade - Current LEV designs can be currently classified into two groups: (1) Top loaded, and (2) Side loaded cargo types, Figure 4-48.

Top loaded cargo landers, as their name implies, require unloading of cargo from above. Large mobile surface cranes or gantry-type off-loaders are required. They must drive up to, then over (on top of) the lander, grasp the cargo at its Cg, lift off the cargo and raise it up and away from the lander, then drive off with the cargo in this raised position. A surface unloader required to do this task would be heavy, and require a dedicated flight just for its emplacement and assembly on the surface. The 'over top' or gantry-type off-loader has an estimated wt of 12 tons and an assembled frame size of roughly 15 m by 15 m, and would have to be delivered and assembled on the surface before any other payloads could be delivered with the top loaded lander type. Since the lander alone is helpless to unload itself in the event of a surface unloader failure, it would be rendered incapable of ascent to orbit, and the crew stranded if such a failure occurred. This lander concept was therefore judged as placing a relatively high risk and weight burden on the system. A design providing more efficient cargo off-loading is preferable.

Side loaded cargo landers, of which the NASA LEV baseline is an example, must divide the total cargo load into two sections, each held to the sides of the vehicle with large side mounts. These two cargo sections must be of approximately equal weight for the vehicle center of gravity (CG) to be balanced on the descent flight phase. An over-the-top gantry

Lander type	Vehicle sketch	Off-loader sketch	Off-loader and/or transporter type, weight, number of dedicated flights to deliver off-loader or transporter
Top loader			• Off-loader type: *gantry type overtop lifter* Transporter: *same* Cargo division necessary? *No; single pallet or surface module* Weight: *12+ tons* Delivery flights: *1-2* *off-loader size and delivery/assembly are major drawbacks*
Side loader			• Off-loader: *None required; cargo lowered directly to surface* Transporter: *Flat-bed; pulled to location by rover, or self propelled. lifting crane may be necessary for placement on transporter* Cargo division necessary? *Yes; cargo must be divided into two equal sections; major difficulty. single surf module hard to accommodate* Weight: *2-3 tons* Dedicated flights: *may be possible to carry a transporter as one side load*
Bottom loader			• Off-loader type: *None required; cargo lowered directly to surface* Transporter: *Flat-bed pulled to location by rover, or self propelled* Cargo division necessary? *No; accommodates large surface modules* Weight: *2-3 tons* Dedicated flights: *None; large transporter can be carried with cargo underneath lander* *Most efficient cargo delivery type*

CARGO UNLOADING CHARACTERISTICS OF CANDIDATE LANDER DESIGNS

FIGURE 4-48

type unloader would not be necessary for this lander type. It is also possible that a mobile crane unloader would not be necessary either, since the two side mounted cargo pallets could be lowered directly to the surface (or on to flatbed transporters driven underneath). This would prove to be an important benefit since the lander would not be incapacitated for ascent if a surface transporter failed. However, the need for dividing the cargo payload into two equal pieces does not reflect the reality of surface system payloads (they are not all

divisible). A single heavy (20-30 ton) surface habitat module would be hard to accomodate. Providing for undivided cargo delivery would be preferable.

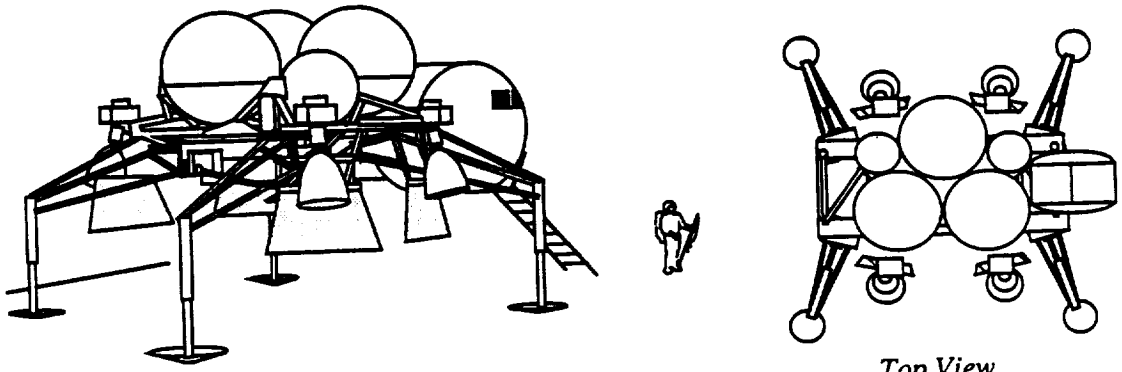
During discussions concerning the impact of cargo unloading on lander design, a lead member of the US Army Civil Engineering Research Lab group, currently involved in the study of planetary surface cargo unloading and transport, made the following statement:

"The primary need in providing ease and efficiency of cargo unloading is the direct access of at least two adjacent sides of the cargo from underneath."

Obviously, the most important requirement for a lander system, from the surface cargo unloading systems point of view, is the ease of getting the cargo load onto a surface transporter for immediate or later access to its contents.

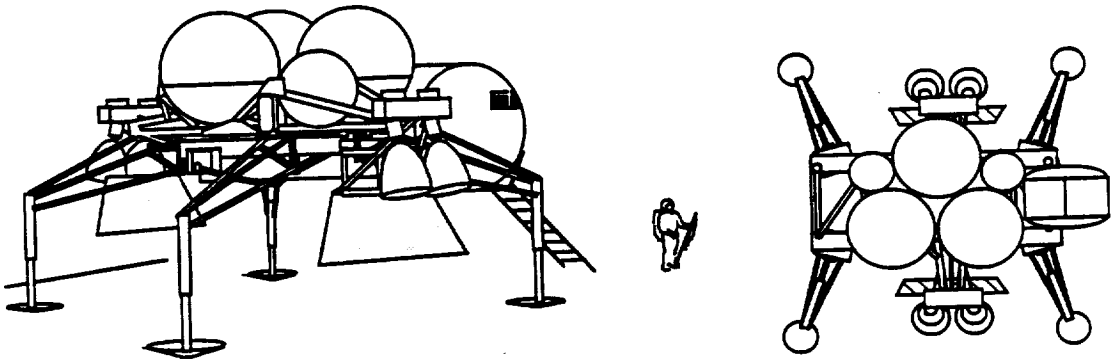
Satisfying the requirement: unloading from underneath - Placing the cargo underneath the propellant tankage/propulsion system, instead of above it, is the distinguishing characteristic of the undercarriage cargo lander configuration. The vehicle utilizes four or six engines. Three candidate configurations are illustrated in Figure 4-49. The engines are positioned several meters above the ground plane and also are out of the way of the cargo. This 'over top' engine positioning prevents the engines from blocking or enclosing the cargo, and allows access to the cargo from any two adjacent sides as needed by a flat bed surface transporter. Any alternative approach of lowering the placement of the engines near the ground level position would enclose the cargo and thus prevent access to the two adjacent sides. The cargo module or pallet is held at the bottom of the vehicle, attached from above to the vehicle's cargo bay. Directly above the cargo bay lies the base of the engine extension frame structure and the propellant tanks. The engine nozzles extending out away from the cargo bay section are canted slightly outward from the cargo and incorporate plume impingement shields to prevent any exhaust gas impingement on to the vehicle or cargo when landing. These shields are mounted on hinges and can be easily pulled up and out of the way in order to clear all sides for cargo removal. The Lunar LEM also utilized impingement shields for its RCS.

When the lander is used to carry personnel as well as cargo, a small crew cab is also attached underneath the cargo bay in a position adjacent to the cargo. In the case of delivery of a large surface crew module the small ascent crew cab can be joined to the surface module to provide for contiguous placement and use of the two modules. After a piloted



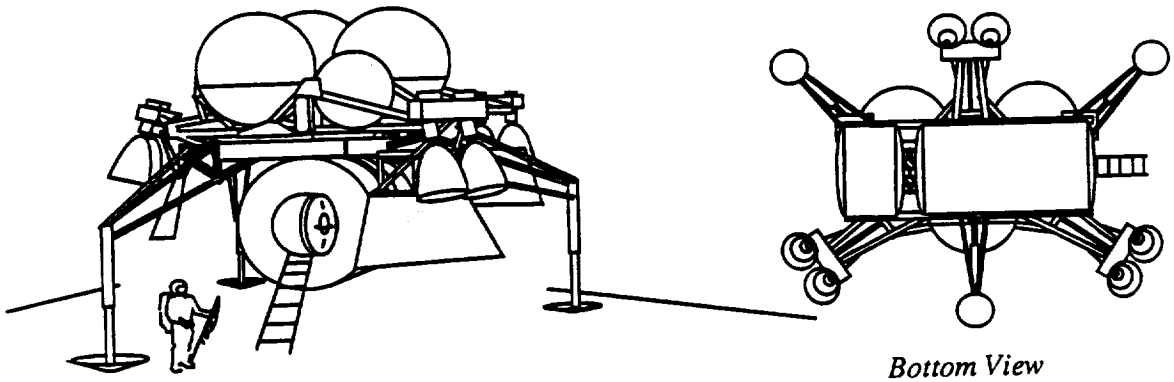
Top View

Ascent cab on side; cargo underneath (cargo not shown)
single stage vehicle: 4 engine design: Top: corner engine placement
Below: paired engine extensions. Engine out: shut off opposing engine



Top View

Ascent cab and cargo underneath (cargo shown)
single stage vehicle: Below: 6 engine design: 1 pair at each of 3 extensions.
Engine out: throttle healthy engine on affected extension up to compensate



Bottom View

UNDER CARRIAGE CARGO LANDER CONCEPTS
FIGURE 4-49

surface mission, the surface module is off-loaded and the small crew cab remains on the vehicle for ascent. The undercarriage lander provides the capability to unload from underneath and for crew ingress and egress from a underneath position directly to the surface without climbing down a long ladder. This affords a significant operational advantage that is not available from either of the other lander options. Specific advantages are given below:

- Easiest access to the cargo by surface vehicles
- Capability to off-load the cargo to the surface or to a surface transporter without the need for any surface off-loading vehicle whatsoever
- Undivided cargo delivery
- Easiest access to the surface for the crew
- Best crew visibility of the surface for descent and touchdown maneuver
- Easiest cargo ejection for emergency descent abort maneuvers
- Provides contiguous placement of surface and cab crew modules
- Easiest cargo on-loading from the surface for return or translation of cargo
- Easiest accommodation to a variety of cargo weights
- Incorporates lessons learned from terrestrial cargo delivery helicopter operations; examples shown in Figure 4-50

Gravity assist unloading - Utilizing the undercarriage cargo concept, cargo down-loading can be accomplished in a one step, relatively risk free operation; the cargo is lowered down by hoists to the surface solely by the the pull of gravity, no lifting cranes are needed. Utilizing this approach, Lunar or Martian base build up might be done entirely without the use of any lifting cranes or gantry off-loaders whatsoever. Since the unloading operation can be completely independent of any surface equipment, an emergency, unplanned ascent to orbit can be effected immediately. Vehicle ascent would not be affected by surface equipment failure.

Descent abort cargo jettison - In the same way an emergency descent abort could also be done in one step: cargo release and drop, again utilizing the force of gravity for separation. The top-loader and possibly the side-loader designs require some form of higher risk cargo ejection scheme.

TERRESTRIAL CARGO DELIVERY HELICOPTER ANALOGS
FIGURE 4-50

Engine out considerations - The severity of anticipated engine out situations on descent and the resultant off vehicle CG complications are actually independent of whether the engines are placed above are below the cargo; this, by itself, is not a driver. Severity of engine out complications and strategies for their compensation are mainly related to the following:

- Engine spread (the distance the engines are spread out from one another in the radial direction)
- Number of engines
- Engine position relative to the vehicle CG
- Available engine gimbal angle
- Supplemental control from RCS

Engine out recovery capability need not be more difficult for the 'overtop' engine undercarriage cargo configurations than for the traditional engine 'on bottom' position if the driving factors mentioned above are equivalent for both designs. Both techniques of differential throttling and engine gimbaling can be used to compensate if a engine fails.

Summary - A landers potential usefulness between surface touchdown and lift off is an important ingredient to achieving a highly successful surface exploration or site buildup mission. In retrospect, the judgement was made that lander design must extend beyond merely satisfying the primary descent/ascent requirement to also meet other surface mission related requirements. This more complete design philosophy can be summarized in the following statement:

The role of the MEV/LEV circumscribes more than the descent/ascent task, and incompasses the delivery of cargo and personnel in such a way as to maximize the effectiveness of the surface mission, as well as providing for an immediate means for cargo jettison during descent abort to orbit.

4.5.9.2 Second Site Exploration Capability

A concept aimed at increasing exploration potential for a given payload mass delivered to Mars or to the Moon, incorporates suborbital flights to distant sites for additional exploration. A lander capable of visiting two sites before ascending back to orbit would provide several benefits to planetary exploration missions. Surface 'hopping' is herein defined as a suborbital flight of a few hundred kilometers. Such a capability could, for

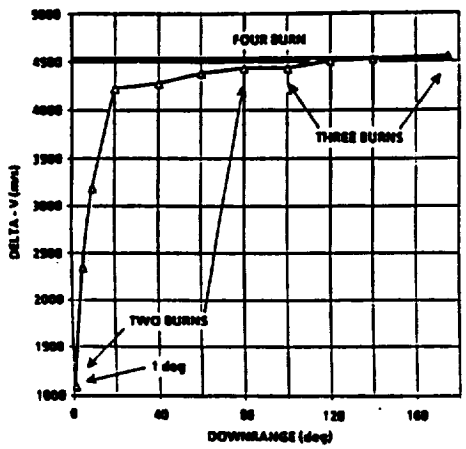
Mars, provide both the desirable 30 day surface mission (based out of a large surface habitat module of ~20 tons) typical of the reference MEV mission, and also a short 3-5 day duration exploration of a second distant site. This shorter mission was typical of the 'mini' MEV mission described earlier. A key design objective was to satisfy this dual requirement, providing the capability for an efficient cargo delivery system that could also serve effectively as a 'hopper' for a second short duration exploration sortie.

The undercarriage cargo lander concept, designed to simplify cargo unloading operations at the surface, is utilized in a piloted/cargo configuration for evaluation of its capability for this dual delivery/exploration task. Both a small ascent crew cab and a larger surface habitat module are linked together and carried under the vehicle propulsion/frame carriage for the descent maneuver and initial surface stay phase. The large module is then lowered to the surface from the propulsion carriage, with the small ascent crew cab remaining in position underneath. Now lighter by at least a factor of two, and having a takeoff T/W at least a factor of two higher (when compared to its earlier initial descent from orbit), a suborbital flight to site two is undertaken. Vehicle characteristics advantageous for second site capability include the capability of off-loading the first module without the aid of an unloader, use of a single propulsion stage, and highly responsive flight controllability for higher risk flights into cannons or on to mountain plateaus.

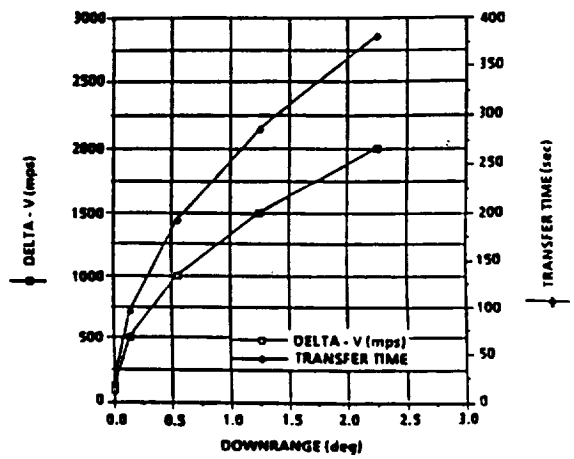
Lunar/Mars Surface to Surface Suborbital Transfers - To make a proper evaluation of a landers capability as a suborbital flyer, the delta V requirements for such excursions must be known, and a means for relating some form of cost figure for any such capability. For this study lander weight vs distance flown and payload carried was the method for assessing a cost to the system for second site exploration flights.

Delta V calculations for suborbital transfers - A suborbital transfer is defined as a series of maneuvers targeted to transfer a LEV or MEV from an initial surface location to a final location. Two different approaches exist. The first approach entails executing a series of powered maneuvers to transfer from an initial basepoint on the planet surface, through a specified surface relative transfer angle, to another location. The first burn of this "burn-coast-burn" scenario achieves the required apoapsis altitude and the final burn achieves the required constraints for landing. In this case the apoapsis of the transfer orbit is optimized as a function on the desired downrange transfer angle - this is labeled in this section as the "two burn," or "multiburn" transfer.

The second approach consists of a single-powered maneuver where the LEV/MEV will hover and horizontally transfer through the desired transfer angle - this is labeled as the Hover/Horizontal Transfer (HHT). The engines fire throughout the entire flight to maintain a constant hover altitude. The vehicle does not 'coast' for this flight. Delta V is plotted against surface transfer range for both maneuver approaches in Figures 4-51 and 4-52 (Reference 30).

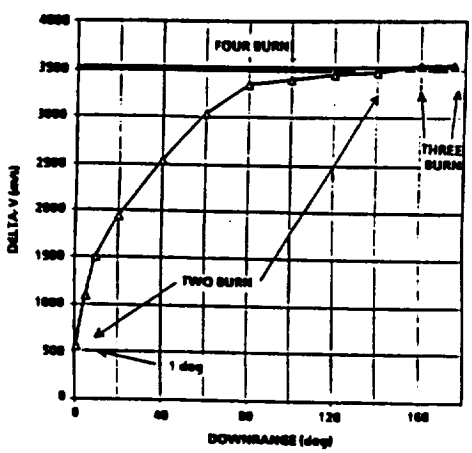


Downrange: *Mars: 1 deg = 60 (km)*

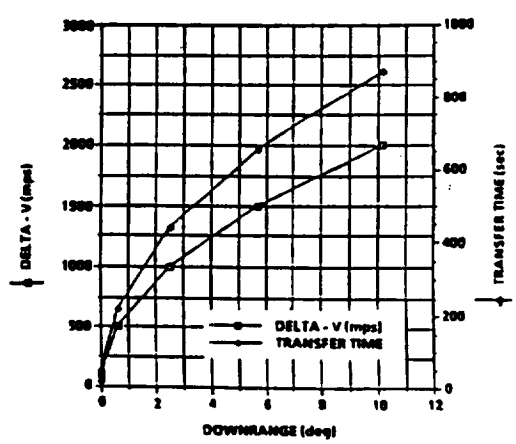


Downrange: *Mars: 1 deg = 60 (km)*

MARTIAN SURFACE TO SURFACE SUBORBITAL TRANSFERS
FIGURE 4-51



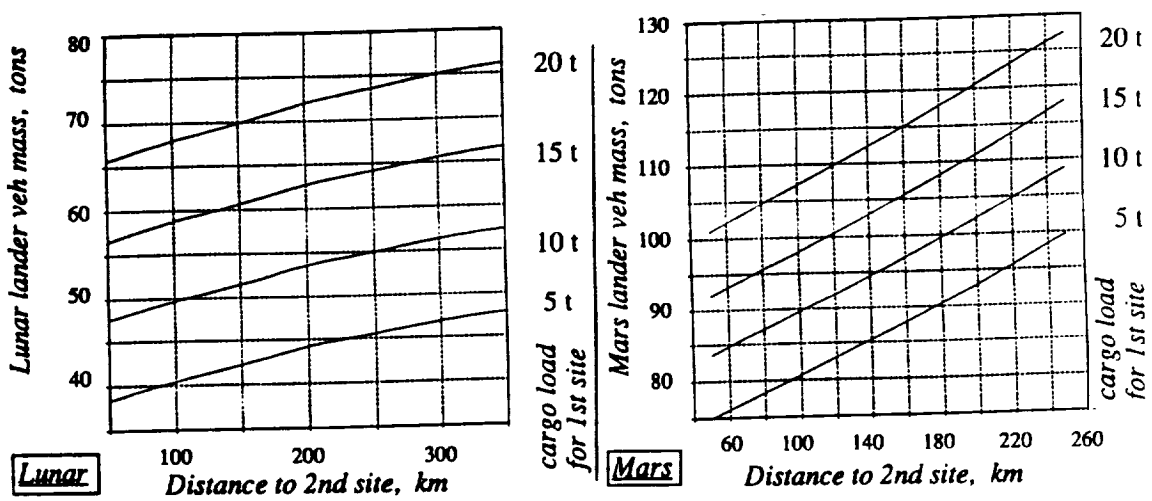
Downrange: *Lunar: 1 deg = 30 (km)*



Downrange: *Lunar: 1 deg = 30 (km)*

LUNAR SURFACE.TO SURFACE SUBORBITAL TRANSFERS
FIGURE 4-52

Trade Results - MEV system costs for second site transfer. To provide for a mission incorporating an initial stay surface habitat weight of 20 tons and a second site distance of 100 km, the total lander weight in Mars orbit would be 108 tons. This mass is roughly 24 tons (28%) more than the 84 ton, 2 stage MEV used as the reference vehicle for this study, which was incapable of a second flight. This is illustrated in Figure 4-53 where total vehicle weight variation with cargo and distance is given for 5, 10, 15, and 20 ton surface modules, as well as distances traveled up to 260 km. Although an attempt to estimate the added scientific benefit or value that might be gained from a second site visit will not be made herein, it should be noted that the cost in IMLEO for bringing a second lander along as an alternate way to visit a second site would assuredly be more than the additional 24 tons of mostly added propellant (as calculated above). Even the spartan 'mini' MEV that provides a crew of 4 (with minimal cargo) about a week of surface stay weighs in at approximately 40 tons.



LUNAR/MARS LANDER SECOND SITE EXPLORATION CAPABILITY
FIGURE 4-53

LEV system costs for second site transfer: nominal case - A second site LEV mission is described as follows. An LEV lands and does an initial surface exploration, or cargo delivery mission. .The LEV, with or without its initial payload, flies suborbitally to a

second site for further exploration. The payload or surface module is dropped (if it had not been dropped earlier), culminated by ascent to orbit.

The cost in LEV weight for a range of payloads and transfer-to-second-site distances is shown in Figure 4-53. For a specific example, the total lander weight in Lunar orbit would be 50 tons, if 10 tons of cargo were delivered to the surface initially and then taken an additional 100 kms distance to a second site. This is roughly 6 tons (14%) more than a 44 ton LEV which could carry that same payload on descent though incapable of a second excursion later, before ascent. Again it can be noted that the estimated alternative cost in IMLEO for visiting that second site with a separate LTV/LEV system at a later date would be at least an order of magnitude higher.

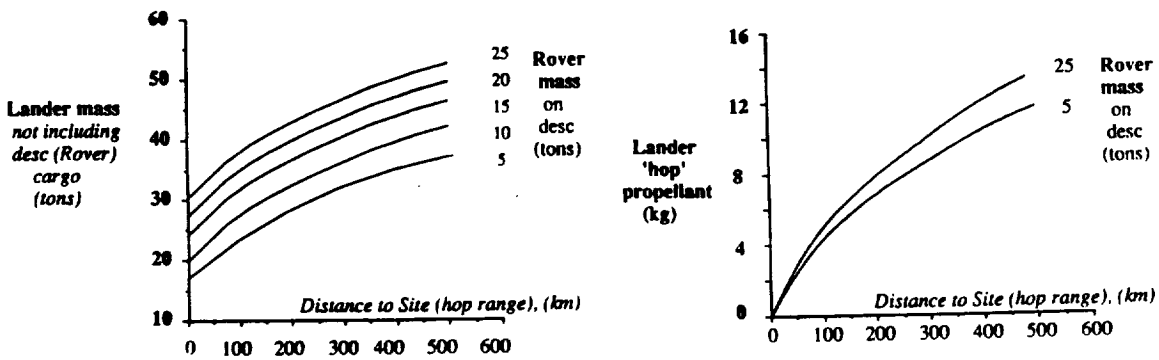
General Summary - For SEI mission strategies that stress multiple planetary exploration missions that include site visits within as much as several 100 kms of each other, the approach of designing MEV/LEV systems that can accommodate short suborbital excursions, is less expensive in terms of mission cumulative IMLEO than utilizing multiple landers. The use of surface rovers also must be included if the overall trade space is to be covered.

4.5.10 Specific Second Site Missions

Specific missions were identified and evaluated in order to make a finer judgement on the practicality of providing a LEV with second site capability. It should be noted that LEV systems have all the necessary flight and propulsion hardware on board to do a suborbital hop. Design changes would principally be changes to allow the accommodation of more propellant, as well as provisions for off-loading an initial cargo pallet or module without the use of a separate surface cargo unloader (this latter item has been described in detail earlier in Section 4.5.9.1). To this end, the undercarriage cargo lander concept, designed to simplify cargo unloading operations at the surface, is utilized in its piloted/cargo configuration for an evaluation of its capability for this dual task of descent/ascent and second site flight. Both a small ascent crew cab and a larger surface crew habitat module can be linked together and carried under the vehicles propulsion/frame carriage for any or all of the mission phases; descent, hop, and ascent. From the propulsion carriage, a large crew module or cargo module can be lowered down to the surface, or hoisted up (on-loaded) from the surface while the small ascent crew cab remains in position underneath.

4.5.10.1 Rover Crew Recovery

The use of a LEV capable of a suborbital flight to the site of a failed rover for rescue of the stranded crew was evaluated. For this analysis the effect on lander weight was calculated for variations in distance to the site as well as to the rover weight. (The rover itself was not 'on-loaded' and returned to base. It was assumed that the lander would originally deliver the rover to the surface.) Once the rover was delivered and the rover surface mission was underway, the lander would remain on site, or at the base, if and until it was needed to rescue the crew in case of some rover failure. The LEV and rover masses relating to second site distance is provided in Figure 4-54.



ROVER CREW RECOVERY IMPACT OF DISTANCE, (LEFT) MASS, (RIGHT) PROPELLANT
FIGURE 4-54

The lander is utilized in a piloted/cargo configuration for the original delivery of the rover cargo, then in its piloted only configuration (no cargo in the undercarriage bay) for the suborbital hop to and from the rover failure site.

A 1000 km pressurized Lunar rover would weigh approximately 25 tons. Driving at a rate of 4 km/hr for 10 hrs per day (24 hr period) during the 14 day daylight period and at 2 km/hr for 10 hrs per 24 hr period during the lunar night would require about 33 days to reach its 1000 km limit before starting its return trip. If, for example, the rover failed at a

distance of 500 km, a second rover sent out to affect a rescue would need around 33 days to reach and return the crew to the base.

A smaller "daylight" pressurized rover, weighing approximately 12.5 tons would have a maximum range near 240 km. Driving a 4 km/hr for 10 hrs. per day during daylight would mean that about 6 days of travel would be necessary to reach its 240 km radius limit. If the rover failed at this distance, a second rover might require about 5 days of travel time to reach and return the crew. It was assumed that for rover rescue attempts of 300 km or less, a recovery rover crew of two, alternating driving and resting periods among themselves, might be able to drive continuously (24 hrs per day) at 4 km/hr in order to reach a stranded crew and return in the 5 day period. For distances of about 300 km or more (one week and up to travel time) a rover crew might not be able to keep up the pace of around the clock continuous operation.

For the 25 ton long range rover at 500 km, the 33 day recovery time is assessed as impractical, entailing too long a wait and thus too high a risk to the stranded rover crew. A LEV capable of flight to a second site could accomplish just such a recovery in less than one day.

For the smaller "daylight" rover, the 240 km, 5 day recovery period is certainly more reasonable than the 33 days for 500 km. However, for cases when a rover crew member is injured or when the rovers ECLS system has failed, or is failing, even 5 days may prove to be excessive; anything longer than a one day rescue may be to no avail.

The cost for recovery for a second flight capable LEV or MEV would merely be the added propellant necessary to do the suborbital rescue flight. Analysis showed that 14 tons of propellant would be necessary for a LEV to fly out to, and return from a 500 km distance, while about 9 tons would be needed for the 240 km distance.

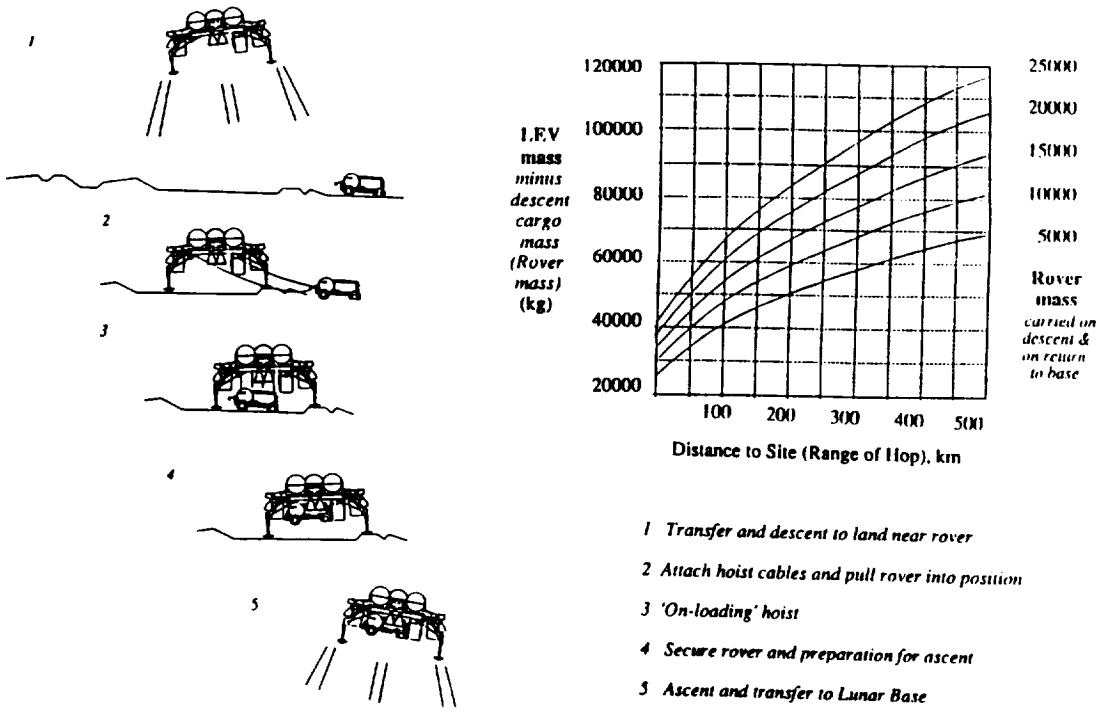
Accomondating additional propellant appears to be an order of magnitude less expensive than building and delivering a second pressurized rover (a very high technology item) just to provide a recovery vehicle for a rover mission.

When the difference in recovery times is taken into consideration as well, the approach of using the lander as the rescue vehicle is assessed as preferable to the alternative of delivering a second rover, and utilizing it to effect recovery. The landers capability for suborbital hopping would be beneficial for other reasons as well.

4.5.10.2 Pressurized Rover Recovery and Return To Base

The use of a LEV capable of a suborbital flight to the site of a failed rover for rescue of the crew and return of the rover itself was evaluated. For this analysis the variation on LEV weight was calculated for variations in distance to the site as well as to rover weight. The rover itself was 'on-loaded' and returned to base for this analysis. Again, it was assumed that the LEV would initially deliver the rover to the surface. Once the rover was delivered and the rover surface mission was underway, the lander would remain at the Lunar base, until it was needed to rescue the crew and return the rover for repair .

On the LEV mass vs distance to second site plot given in Figure 4-55, rover weights of 5, 10, 15, 20, and 25 tons are listed. The undercarriage cargo lander concept is again utilized in its piloted/cargo configuration for the original delivery of the rover cargo, then in its piloted only configuration for the suborbital hop to the rover site, and again in its piloted/cargo configuration on return. The LEV lands near the rover, hoist cables are attached, the rover is pulled into position under the LEV cargo bay, and is hoisted up ('on-loaded') and secured. The alternative to LEV rover recovery would be the utilization of a second rover to pull the failed rover back, or to abandon the rover permanently.



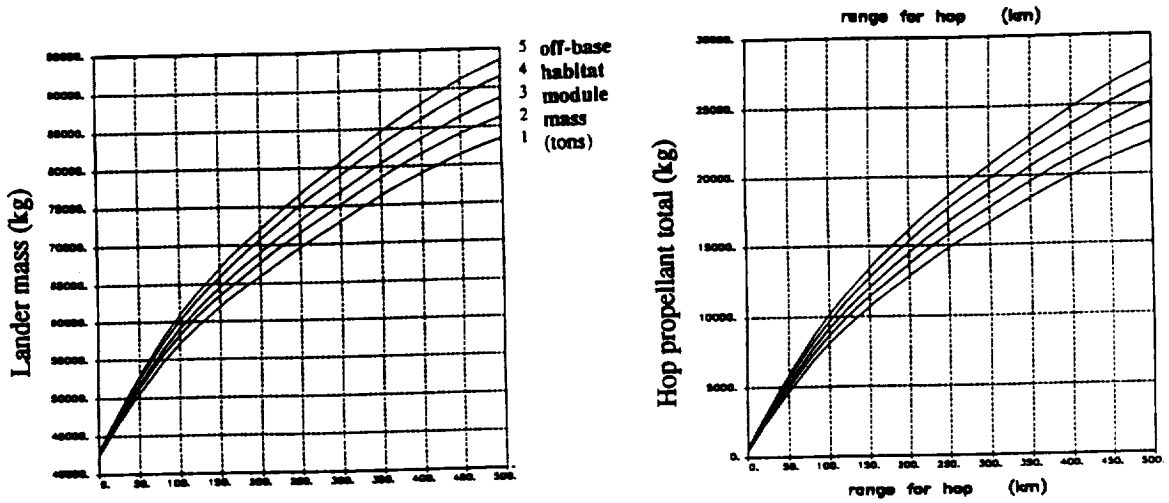
ROVER RECOVERY IMPACT ON LEV DESIGN AND OPS
FIGURE 4-55

4.5.10.3 Pressurized Rover Replacement

Consideration was given to replacing a heavy (15-25 ton) long range, pressurized rover with the combination of a LEV and a light, short range unpressurized rover for surface exploration. For this analysis a LEV carried 10 tons of cargo on its descent task. This cargo was off-loaded for use at the Lunar Base. At the base was a small crew habitat module especially equipped and suited for use with a lander on an "off-base" mission. That is, the extra module provided for exploration surface stays of durations longer than the excursion crew cab could sustain (a few days) by itself.

For this analysis the variation of LEV weight was determined for variations in distance to the site as well as to the mass of the supplemental off-base module. This module was carried both ways; picked up at the base, utilized during the surface exploration, then returned. Additionally, this off-base module served as supplemental base crew habitat module when not in service on an exploration flight. Module weights of 1, 2, 3, 4, 5 tons were evaluated. Again the undercarriage cargo lander concept is utilized in its piloted/cargo configuration. The cost to LEV mass and the hop propellant load requirements, given range and module mass, is shown in Figure 4-56. With a light weight, unpressurized

rover, much actual surface exploration could be done at a distant site in a relatively short time with an LEV.



PRESSURIZED ROVER REPLACEMENT MISSION, IMPACT ON LEV DESIGN
FIGURE 4-56

4.5.10.4 Lunar Vehicles Trades

LTV/LEV Habitat Trade - An LTV/LEV habitat trade module study was conducted to size crew modules for varying crew sizes and mission durations. Two types of transfer modules were evaluated, an aerobraked module and a direct entry (Apollo-type) module, as well as a single module concept for transfer and excursion (direct entry at Earth) and excursion modules. Crew sizes of 2, 4,6, and 8 for transfers of 24 days and surface stays of 1, 14, 28 and 42 days. Sizes for these 36 modules were generated from historical spacecraft data, and mass statements were generated from SSF and STCAEM estimates. Results of this trade study provide estimates as to the size and mass of lunar modules.

A trade study was also performed to determine what point it becomes more mass efficient to have a separate surface habitat along with an excursion module, if a base is not available and missions of the excursion/exploration class are being performed. Results show that 3-8 days is the crossover point.

C-4

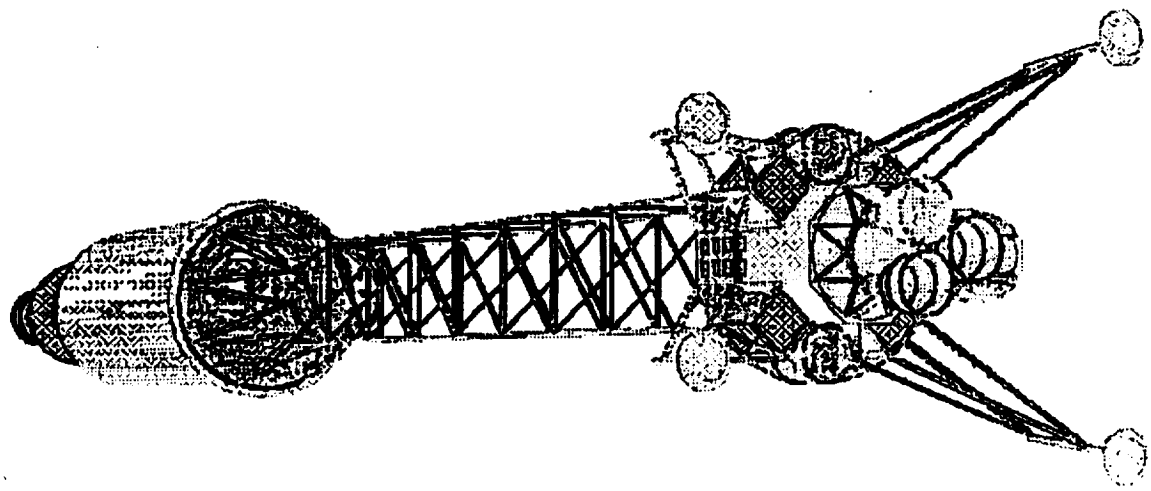
Reference Lunar NTR Vehicle Definition - A Lunar NTR vehicle was developed in order to evaluate some of the benefits of NTR propulsion as applied to the Lunar missions, including:

- High Isp performance
- Ability to do all missions without an aerobrake
- Complete LTV reusability
- Reusability for the LEV (no aerobrake packaging constraints as with the cryogenic propulsion system)
- Propulsion system commonality with NTR Mars vehicle (tank sizing only difference)

Two vehicles were developed; a single lander version, which does the same transfer mission as the present reference LTV/LEV CAB vehicle, and a double lander version which transfers two landers of any type per Lunar mission. The engine, shield, and truss system is identical to the reference NERVA NTR vehicle to provide commonality. Possibly the best proof of concept and flight qualification methodology for a Mars vehicle propulsion system is to demonstrate it in space as a Lunar vehicle. A Lunar round trip is strikingly identical to a Martian round trip from a space transfer propulsion systems point of view if no aerobraking is required. The primary difference is the magnitude of the delta V's and the H2 boiloff for the longer Mars missions. The Lunar NTR vehicle configuration is shown in Figure 4-57.

General vehicle characteristics:

- All propellant in single aft tank; no drop tanks
- Single 75000 lbf ref NERVA eng; T/W=3.5, Isp=925, alternate engine: PBR; T/W=15, Isp=925
- Shadow shield: 4500 (kg); identical to Mars NTR vehicle shield
- 45 m SSF type truss (2400 kg) provides separation distance



NTR LUNAR VEHICLE
FIGURE 4-57

Baseline single lander Lunar NTR vehicle:

- No aerobrake packing constraints; empty LEV returned to LEO for reuse
  Allows checkout, refurbishment & resupply to be done in LEO
- IMLEO = 197 tons, NERVA NTR; 46.5 t LEV, 8.7 t mod
  IMLEO = 170 tons, advanced NTR (t/w=15 PBR)

Double lander version:

- Cargo lander mission done jointly with piloted lander mission
- Single NTR LTV does the job of two CAB LTV's
- Piloted LEV returned to LEO for reuse; cargo LEV left on surface
- Robust propulsion system can accommodate wide range of payloads

Burn times - Earth departure: 26 minutes, Lunar capture: 6 minutes, Lunar departure: 4 minutes, Earth capture: 9 minutes; 45 minutes total. This short burn time keeps fissionable product buildup small. Crew occupies the solar flare radiation shelter already required for the LTV module during these short burns.

Reference Lunar NEP Vehicle Definition - A parametric analysis of a NEP vehicle was performed to gain an understanding of the capabilities of a NEP cargo vehicle for lunar applications. Some of the trades performed included:

- IMLEO vs Isp
- Propellant Mass vs Isp
- Payload Faction vs Isp
- Trip Time vs Isp

The analysis was performed for a vehicle with a specific dry mass (alpha) of 10 kg/kW and 15 kg/kW. Power levels analyzed were from 1 to 6 MWe. The payload was set at 100 t for the entire analysis. A lower powered vehicle resulted in a lower initial mass, but the transfer time became increasingly longer for the lower powered vehicles. There may or may not be a time constraint on the vehicle's spiral from LEO to lunar orbit. A time of 180 days (6 months) was assumed to be the maximum amount of time the transfer should take. The Isp can also be varied with a lower Isp (5,000 sec) resulting in a higher initial mass, but shorter transfer time and a higher Isp (10,000 sec) resulting in a lower initial mass at the cost of transfer time. The results show that a 4 MWe vehicle (10 kg/kW) operating at 6,000 sec Isp can transfer 100 t of payload from LEO to lunar orbit in 6 months or less. The total initial mass of this vehicle at departure would be less than 170 t. Additional details may be found in Reference 11.

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5. INTEGRATION AND COMPATIBILITY

5.1 Integration

During the course of the current study, the Integration task was involved in the vehicle configuration interference assessment and design refinement after the initial configuration conceptual design had been established. This included modifications in response to requirement refinements as defined by other tasks and subsystem integration. It covered two forms of operation ; the first was a standard architectural hardware analysis and revision of the conceptual design, the second was a computer aided design (CAD) three dimensional modeling of the systems and subsystems to locate visually any physical interference in the designs. This is a brief description of the work performed on each of the propulsion options and landing vehicles. This portion of Task 4.0 is also related to work performed under Tasks 1.0 and 2.0, Figure 1-1.

5.1.1 Design Integration

Configuration concepts for all the vehicles were modified as the course of the study progressed and new data was generated that affected the basic design. Cases in point are the designs for the NEP and SEP vehicles that underwent extensive changes. These changes occured while other operations evaluations were in progress, therefore for consistency of evaluation, all operations and manifests have been done on vehicles existing as of August 30, 1990. The refinement of mission requirements and vehicle parameters in response to environmental conditions that affected the designs and systems is outlined in Section 5.1.3. Design sketches for each vehicle are shown in References 6-11.

Subsystem initial placement was done under other tasks, but the final layout and adjustments were performed under this task as an interplay between the overall design and the computer aided modeling.

5.1.2 CAD Modeling

The cryogenic/aerobrake vehicle is equiped with a trans-Mars injection stage (TMIS) consisting of 5 LH2 and 5 LO2 tanks, launched in a tank set of one hydrogen tank and one

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oxygen tank. Integration of individual stages and plumbing interference checks were performed for the nominal configuration.

Separately, the integration of MEV to MEV connection structure within the aerobrake and an evaluation of the aerobrake wake envelope packaging were done. When the MEV landing legs were retracted and it was placed in the aerobrake, then substantial interference assessments and center of gravity (CG) analyses were conducted. This MEV work was universal to all vehicle designs due to its consistent use of the MEV in all missions. Packaging adjustments were made to maintain the CG and wake enclosure constraints for both the MEV and the MTV to assure the vehicle sections would be balanced and protected over the duration of aeroentry and /or aerocapture.

The nuclear thermal rocket propulsion option had 5 hydrogen tanks that were integrated with the main vehicle truss that had to be checked for maintaining the CG position during the jettison of expendable tanks and alignment / interconnections of the tanks for different mission modes. Subsystem integration and a nuclear shielding analyses were performed to ensure the safety of the crew and maximum equipment survivability for the duration of the mission(s). The position of the transit habitat and MEV (manned systems) on the vehicle were driven by radiation shielding requirements from the reactor only (no GCR considerations) for both a single and dual engine case, and CG constraints.

The work performed on the reuseable high L/D MEV (L/D $\geq$ 1.0) entailed modeling a new aerobrake and sizing an upper shroud to conform with packaging protection within the wake envelope. All subsystem integration of landing mechanisms and interior structure on the high L/D reuseable MEV was accomplished on both crew and cargo mission modes. The integration of interior payloads about the appropriate CG was analyzed for the cargo missions. Additionally the reaction control system (RCS) thrusters and propellant tanks were integrated with the vehicle.

The nuclear electric propulsion vehicle required very extensive plumbing and electrical systems integration. The primary reactor loop included plumbing between the radiators and various pumps and had to be routed individually. The decay heat removal loop required plumbing from pumps to auxiliary radiators which also had to be individually routed. Interference checks and redundancy analysis were performed on these systems. The main cycle consisted of various pipes and pumps from the turbo-alternators to the main cycle condensors and radiators whose plumbing systems and connections had to be determined

and plotted. The alternator coolant loops were also analyzed for associated interference and packaging requirements. Electrical cables from the alternators to the power conditioning units of the thruster pod assembly were packaged inside the truss structure of the vehicle and then layed out to be examined in three dimensions. Integration of the electrical systems to the ion thrusters was performed. The transit habitat, airlocks and MEV were integrated with the vehicle and the CG position assessments were made. Ray traces of the radiation impingement beyond the shield were made to determine shadow areas to assess crew safety and ascertain areas of probable maximum equipment survivability. This vehicle involved the most complicated integration process of any of the candidate vehicles.

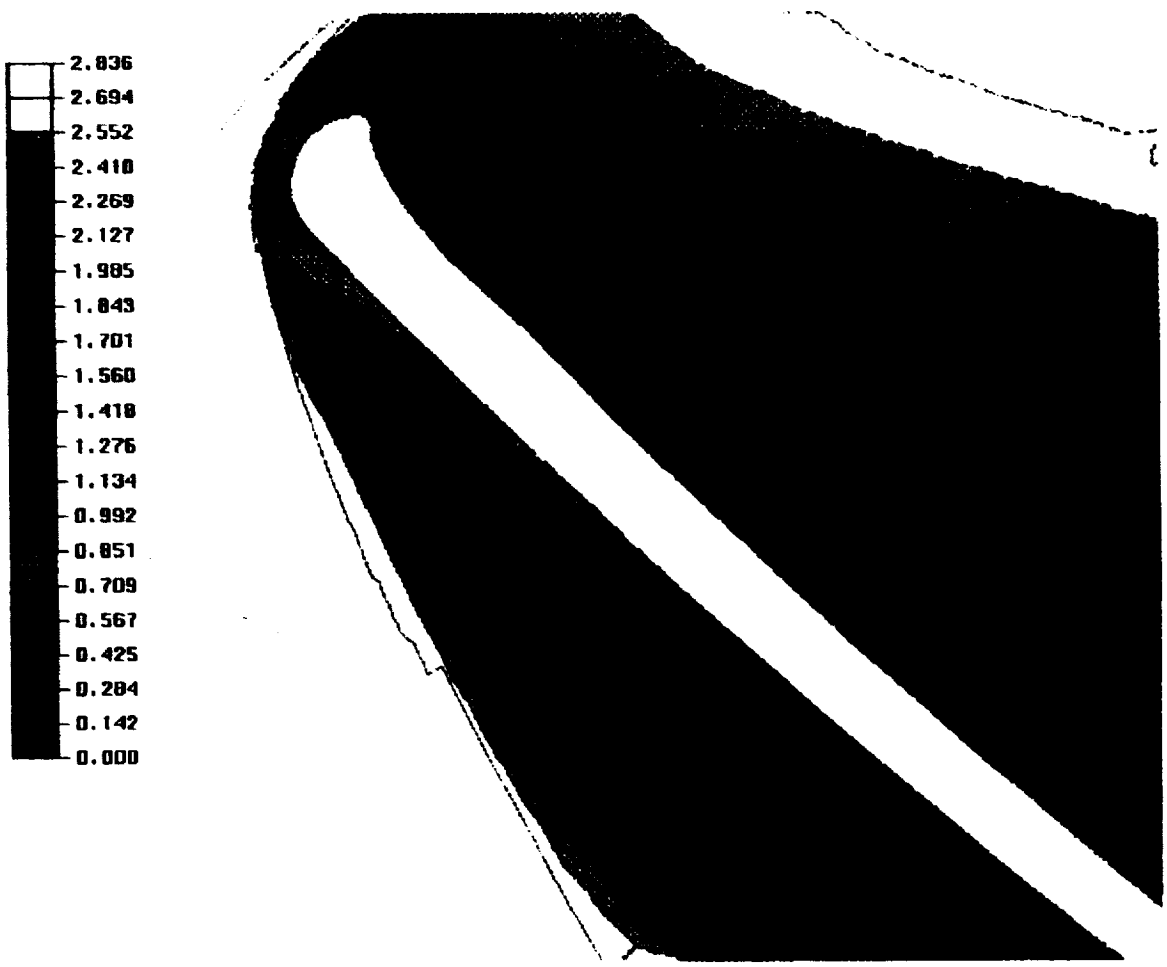
The piping and wiring systems for the artificial gravity NEP and zero -g NEP are identical. However, the ion thrusters are positioned differently so electrical cabling was rerouted and new interference checks were made. The habitat system and MEV were repositioned to conform to the CG constraints involved in artificial gravity at several g-levels. Radiation shield analysis was also performed on this vehicle to ensure safety of crew and maximum equipment survivability.

The solar electric propulsion vehicle required integration of all electrical subsystems and the adaptation of the ion thruster pod developed for the NEP vehicle. Cables were routed to power conditioning units and interference checks were made. Solar panel positioning and packaging analysis were performed on both the original concept and the one done in the latter portion of the contract. Transit habitat and MEV positioning were driven by CG requirements. These final positions were checked for working clearances and structure connections by computer modeling.

5.1.3 Flowfield Analysis

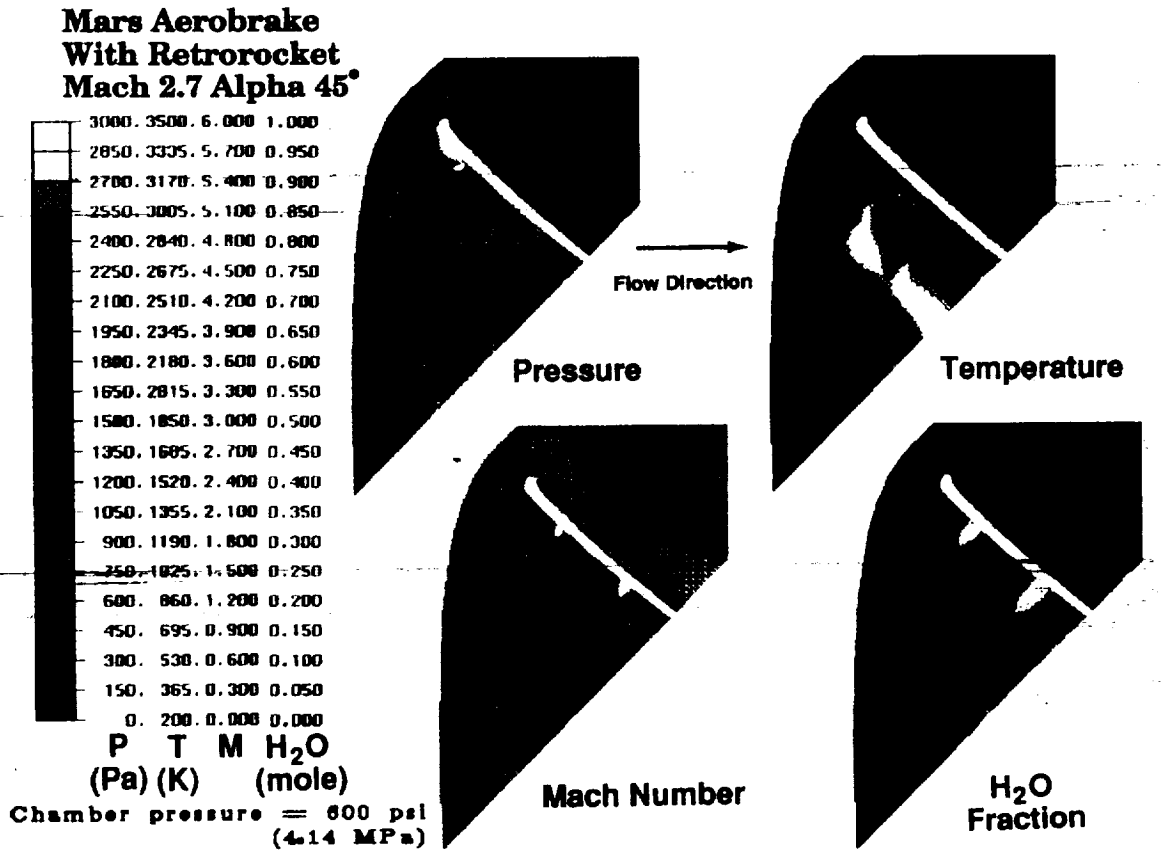
In going beyond the initial mission analysis requirements definitions for baseline aerobrake performance parameters, several areas of inquiry were pursued. These included the forebody and aftbody flows for both low L/D (0.5) and high L/D (1.0+) aerobrakes. The high L/D aerobrake was of particular interest for use as a reuseable MEV. The flow field including surface heating and pressure distributions, associated with the high L/D shape upon descent when the descent engines are fired were evaluated to see if the firing would have adverse effects on the aerodynamics and control of the vehicle. This was investigated using computational fluid dynamics (CFD) analysis. Three-dimensional flowfield calculations were made about the aerobrake using the General Implicit Flow Solver (GIFS,

Reference 31), which is a Navier-Stokes, three dimensional, reacting gas plume code. The analysis was performed for the following set of conditions: an angle of attack of 45°, the shape of the high L/D aerobrake, Mars atmosphere, engine start altitude of 9 km descent velocity of 600 m/sec (Mach 2.7) in the Martian atmosphere. A grid size of 55 x 31 x 41 was required to properly model the aerobrake flow field. The Mach number distribution around the aerobrake, (no back enclosure, engines off) is shown in Figure 5-1.



MACH NUMBER CONTOURS - SYMMETRY PLANE
FIGURE 5-1

With engines on at full thrust (600 psi engine chamber pressure), the flow field, pressure, temperature, Mach number and H_2O distributions along the length of the vehicle is shown in Figure 5-2. It can be seen that the the bow shock of the vehicle has been pushed far from the body and a severe pressure gradient has developed at the nose of the vehicle. With this thrust level, however, the pressure distribution does not prove to be an issue as there is enough control authority from the engines to counteract the flow induced moment placed on the vehicle.



MARS AEROBRAKE FLOW FIELD WITH RETROROCKET, PO = 600 PSI
FIGURE 5-2

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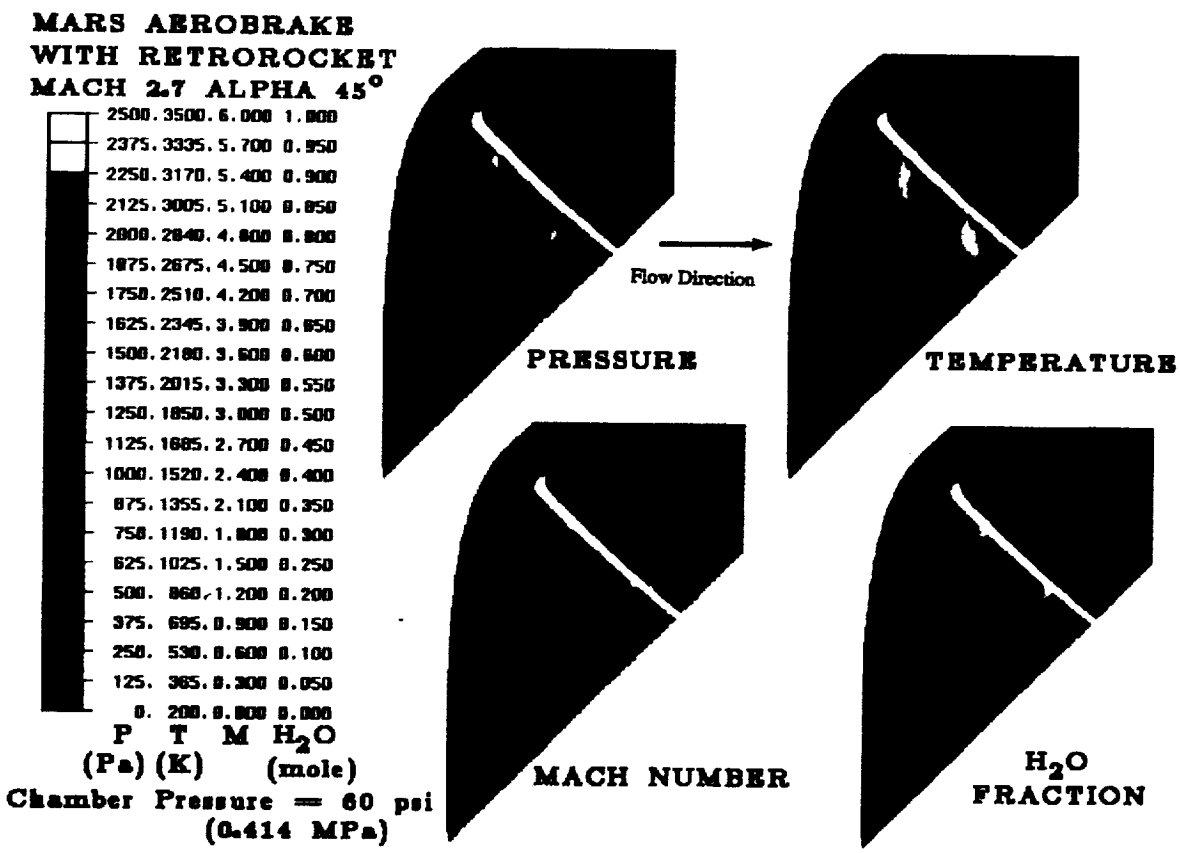
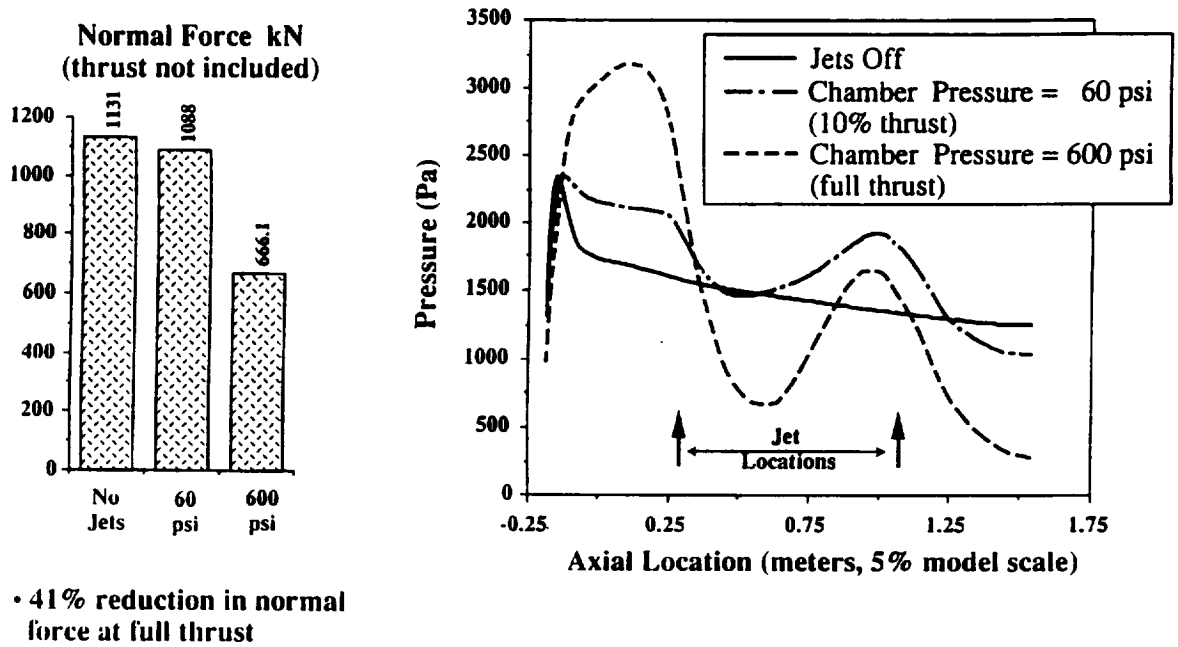
An additional calculation was made to determine the flow about the vehicle at engine "start" conditions, Figure 5-3. With the engine pressure at 60 psi, the control authority might not be enough to counter a severe induced moment. The surface pressure distribution along the centerline of the vehicle is shown in Figure 5-4. As it illustrates, for the "start" conditions, the deviation is less severe than the full thrust condition, thereby resulting in a lower induced moment change.



MARS AEROBRAKE FLOW FIELD WITH RETROROCKET, PO = 60 PSI
FIGURE 5-3

As is expected, there is a change in vehicle performance, particularly a reduction in the lift and drag forces with engine thrust. For the high thrust case, the effects are significant with the normal force, being reduced by 40%, with corresponding changes in lift and drag, Figure 5-5. In turn, these changes impact the descent delta V requirements.



HIGH L/D CFD FLOW FIELD ANALYSIS EFFECTS OF ENGINE FIRING
FIGURE 5-4

	Force On Solid Wall			Total forces	
Case	Lift	Drag	Thrust	F_n	F_a
No Jets	815 kN	784 kN	0	1,131 kN	-22 kN
with jets, 60 psi	786 kN	752 kN	132 kN	1,220 kN	-30 kN
with jets, 600 psi	468 kN	474 kN	1,073 kN	3,163 kN	4 kN

HIGH L/D DESCENT - FORCES
FIGURE 5-5

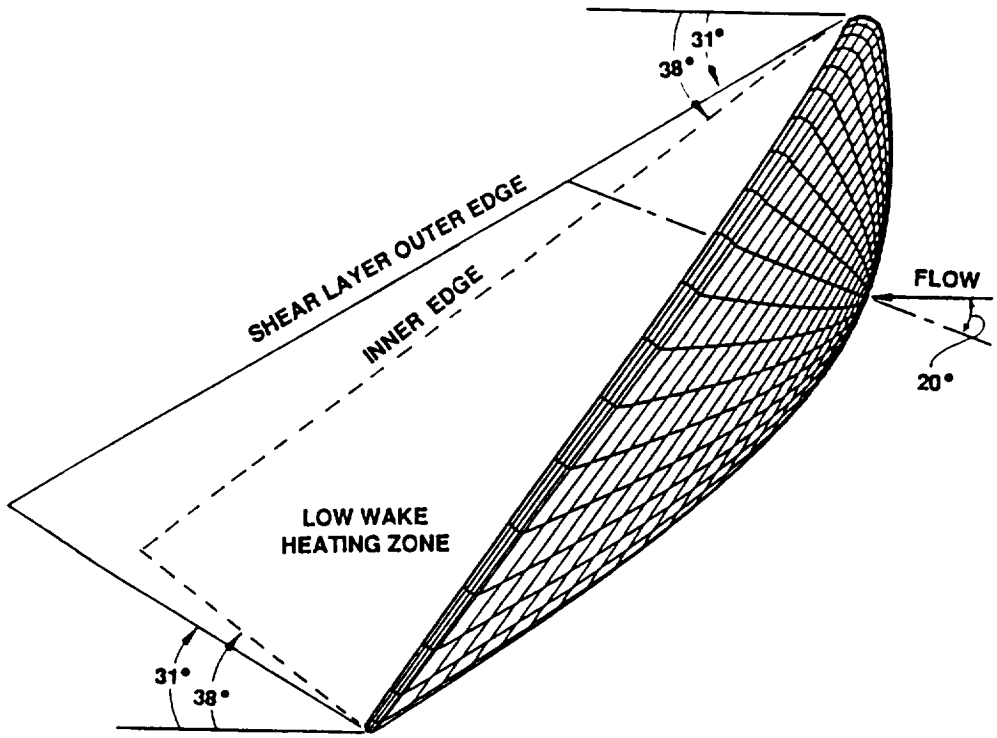
An analysis was conducted for the low L/D aerobrake to determine the extent and severity of the wake closure about the vehicle upon aerocapture. This is an effort to determine the protective cone which would contain the packaging envelope for crew modules, infrastructure, engines and cargo. The study involved examining the base flow heating

regime, which includes both convective and radiative heating for a C3 of 30 km^2/sec^2 MTV aerocapture at Mars. The trajectory used for this analysis was shown in Figure 3-22, where the maximum stagnation point heating was 83 W/cm^2. Radiative heating rate predictions in the brake are based on relationships derived and used for the Aerobrake Flight Experiment (AFE). The radiative heating rate of 1.5 W/cm^2 was estimated at the peak heating condition that occurred at 114 seconds. This value is only 2.6% of the peak radiative stagnation point value. For the convective heating rate calculation the assumed wall temperature (Tw) was 1367 K. A $\pm$ 10 % change in the Tw value resulted in only a ~2% change in heating rate, therefore the wall temperature does not have a large impact on the computed heating rate. The convective heating rate of 1.7 W/cm^2 was estimated for the same peak heating conditions. The total contribution of convective and radiative heating rates amounts to 3.2 W/cm^2, which could be met by a thermal protective system (TPS) material.

The wake closure zone was also determined for the point at which peak base heating occured. Wake shear layer angle predictions were computed based on calculations made for the expansion flow around the lip of the MTV aerobrake. Flowfield properties around the lip were estimated using the BLIMPK program and a shock shape approximated using AFE wind tunnel schlieren photos (Reference 32). The wake closure angle is effected by flow conditions around the lip, which in turn is determined by body shape and angle of attack, Figure 5-7. The wake closure is estimated at 38° per side, with a 31° shear edge outer layer and an additional 7° for the viscous region, Figure 5-6. This closure angle is considerably larger than the value of 22°, which was previously estimated. The heating and wake closure angle may be reduced by flying the brake at a decreased angle of attack (flat on) at a lower value of L/D and a modified aerobrake shape.

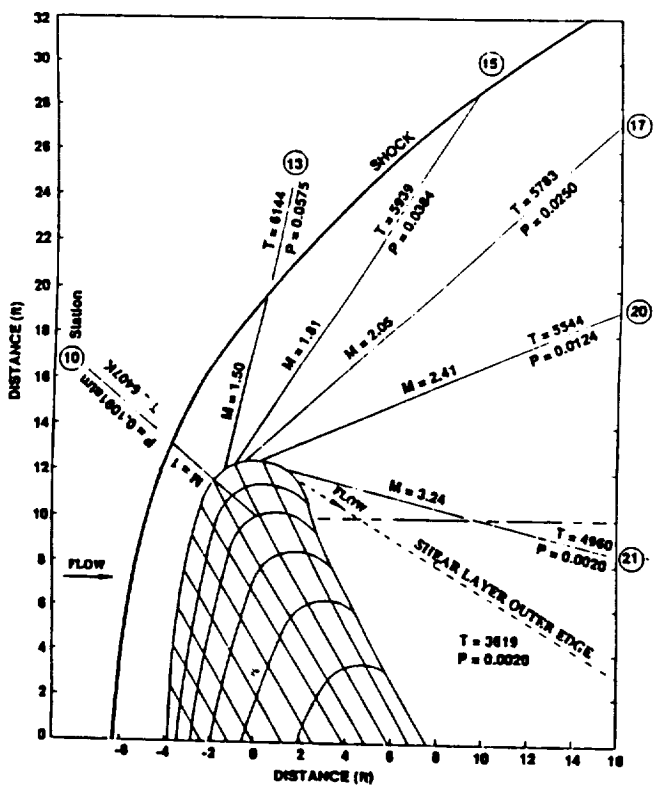
5.2 Manifesting and Processing

The first portion of this activity was devoted to establishing a single point baseline operation and processing scenario. From this baseline, deviations and changes could be established to accommodate the other advanced propulsion concepts in on orbit and ground processing (see Section 6), ETO manifesting, and operations sequencing. The baseline vehicle for use in this method was the nominal cryogenic/aerobrake. The activities performed in this area are directly related to those performed in Task 5, Figure 1-1.



WAKE CLOSURE ZONE AT PEAK HEATING

FIGURE 5-6

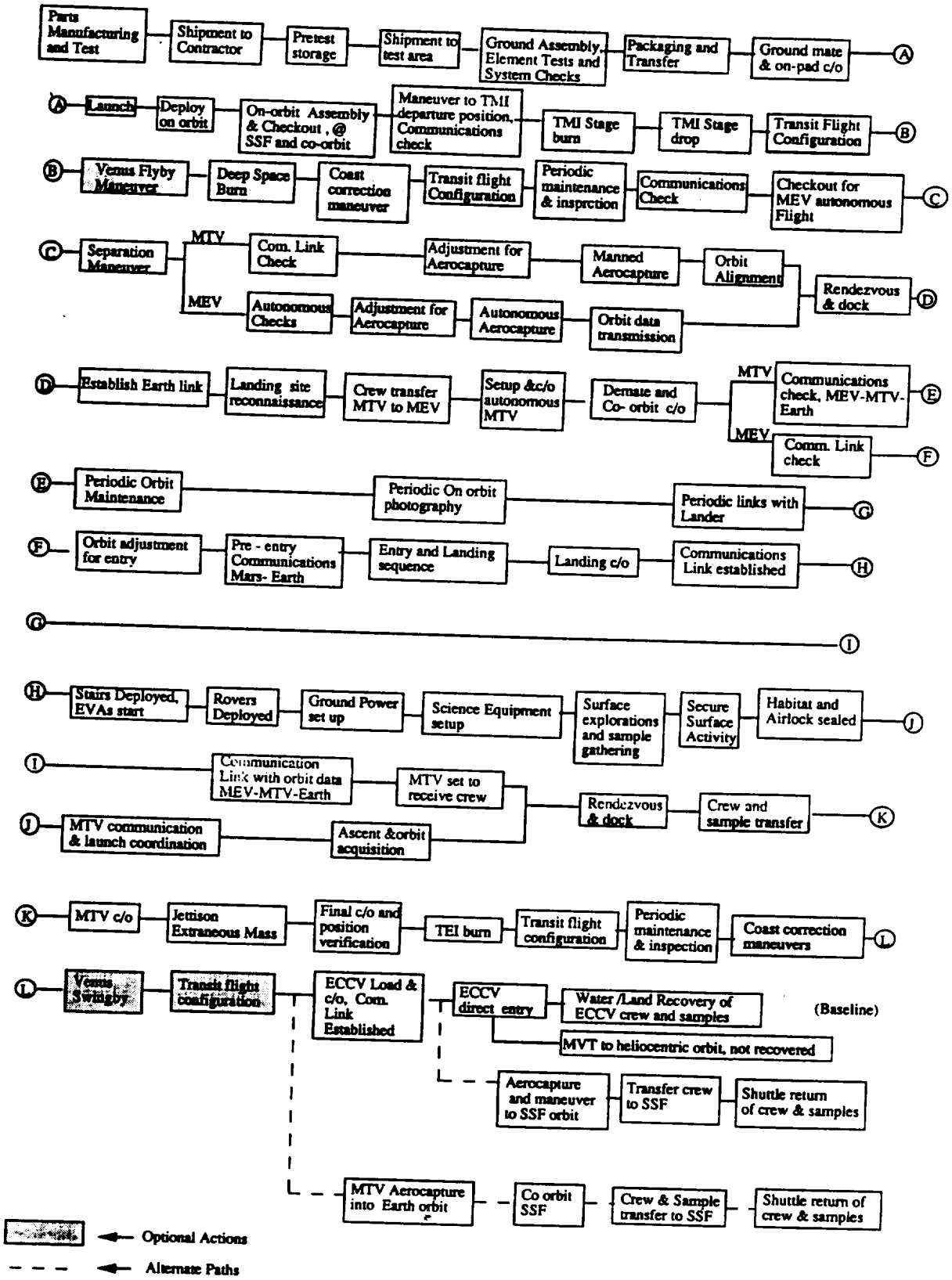


ESTIMATED BRAKE SHOULDER FLOW FIELD PROPERTIES

FIGURE 5-7

For each of the vehicles an overall, top-level operations sequence was developed. An example of the cryo/aerobrake operations sequence is shown in Figure 5-8. The ground processing, launch and on orbit operations constituted several blocks on the overall operations sequence, however, these were considered to be critical junctures requiring an indepth look. In this study, some options and variations concerning node point location, assembly node support framework and services, deployment scenarios or ETO fleet mix were evaluated; some options were identified for future work. The present emphasis on a baseline with deviations helped isolate places where gaps in understanding exist. The baseline systems for cryo/aerobrake manifesting and packaging are given in Figure 5-9 for the HLLV.

The study also concentrated on integration compatibility analyses for the advanced propulsion vehicles, nuclear thermal rocket (NTR), nuclear electric propulsion (NEP), and solar electric propulsion (SEP). Integration compatibility tasks include manifesting and packaging studies, launch sequencing, and interface definition. These analyses were based on the corresponding work previously performed for the reference cryo/aerobrake vehicle; i.e., the main assumptions and ground rules as well as any identical manifesting conditions (HLLV, etc.) were taken directly from this previous work. These analyses were performed for different launch vehicles and launch assumptions as well as for several assembly node concepts and ground processing scenarios (see Section 6). During the course of this study, it became quite obvious that the choice of the heavy lift launch vehicle (HLLV), assembly method (robotic, EVA, etc.), assembly node, and vehicle configuration were all inter-related and one cannot be optimized at the exclusion of the others. However, useful results may be obtained by establishing reference assumptions:



MARS MISSION OPERATIONAL TASK FLOW CRYO/AEROBRAKE

FIGURE 5-8

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· HLLV : Shroud Size- 30 meters x 10 m dia., 140 t throw weight

	Flight #	1	2	3	4	5	6	7	8
	Launch Vehicle	HLLV	HLLV	HLLV	HLLV	HLLV	HLLV	HLLV	HLLV
Cryo/Aerobrake	Manifest	MEV Aerobrake (Ninja Turtle), Descent stage, surface cargo, assembly equipment, consumables	Ascent stage, interconnect structure (MEV-MTV), TEI departure stage	MTV Hab Module, ECCV, MTV Aerobrake (Ninja Turtle), interconnect structure (MTV-TMIS), assembly equipment, consumables	TMIS tankage, Engine set, Structure, assembly equipment, Top-off equipment	TMIS tankage and propellant	TMIS tankage and propellant	TMIS tankage and propellant	TMIS tankage and propellant (wet tanks and top-off)
	Mass	67.8 t	138.7 t	100 t	140 t	119.7 t	119.7 t	119.7 t	140 t

	Flight #	1	2	3	4	5	6	7
	Launch Vehicle	HLLV	HLLV	HLLV	HLLV	HLLV	HLLV	HLLV
NTR	Manifest	MEV Aerobrake (Ninja Turtle), MTV Hab, Ascent module, Descent module, Main truss, assembly equipment	Mars Departure Structure, Surface payload, Mars Departure Tankage	TMI Tank	TMI Tank	Engine, Shield structure, Mars Arrival/ Departure Tank	Reactor Engine and Shield, Mars Arrival/ Departure Tank	Propellant for top-off
	Mass	132.8 t	121.1	140 t	140 t	85.3 t	94.3 t	26.6 t

MARS MISSION MANIFESTS - CRYO/AEROBRAKE AND NTR, 140 t HLLV
FIGURE 5-9

a. The baseline HLLV was chosen with a 10 meter diameter, 30 meter length shroud and a payload mass capability of 140 metric tons. Similar analyses were conducted using different launch vehicles (a mixed launch vehicle fleet of 84 and 120t capacity, and the Shuttle C with Shuttle Z launch set) for comparison with this baseline, Figures 5-10 and 5-11.

b. The assembly method studies depended on extensive on-orbit robotics with manned presence reserved for contingencies and crew module check-out only. These robotics would be a combination of ground and space-based teleoperations along with built-in autonomous capability.

c. Assembly node concepts were developed for a broad range of on-orbit infrastructures. These concepts exist at varying levels of maturity but served as bases for sequencing analyses.

d. The work performed, has been based on the NTR, NEP, and SEP as defined by the vehicle concepts as of August 30, 1990 . Establishing this benchmark allowed effort under these tasks to be consistent and thorough without being constantly

	Flight #	1	2	3	4	5	6	7-10	11
Cryo/Aerobrake	Launch Vehicle	HLLV 1	HLLV 1	HLLV 1	HLLV 1	HLLV1	HLLV 2	HLLV 2	HLLV 1
	Manifest	MTV Hab Module, Surface Payload, Assembly Equipment	MEV Aerobrake (Ninja Turtle) Habitat refurbish/ consumables, Assembly Equipment	Descent Module, Ascent Module, structure	MTV Aerobrake (Ninja Turtle), Habitat refurbish/ consumables, Assembly Equipment, ECCV, TEI Tanks & Engines	TEIS Propellant, consumables	TMI Propellant & Engines	TMI Propellant and tanks	Top-off Propellant and equipment, Habitat refurbish/ consumables
	Mass	78.8 t	40.1 t	49.1 t	76.4 t	84 t	117.7 t	114.5 t	84 t

	Flight #	1	2	3	4	5	6	7	8	9
NTR	Launch Vehicle	HLLV 2	HLLV 1	HLLV 1	HLLV 1	HLLV1	HLLV 1	HLLV 1	HLLV 1 or 2	HLLV 1 or 2
	Manifest	Descent Module, Ascent Module, Surface Payload, MEV aerobrake (Ninja Turtle), Assembly Equipment	Mars Departure/ Earth Arrival Structure, Main Truss, MTV Hab	Engine/ Shield Structure, Mars Departure/ Earth Arrival tank (13.5 t off-loaded),	Mars Arrival/ Departure Tank (1 of 2)	Mars Arrival/ Departure Tank (2nd of 2)	Earth Departure Tank (1 of 2, 69.3 t off-loaded)	Earth Departure Tank (1 of 2, 69.3 t off-loaded)	Off-loaded Propellant from Flight 3 &6	Off-loaded Propellant from Flight 6 & 7. Reactor Engine and Shield
	Mass	112.2 t	46.6 t	84.0 t	82.9 t	82.9 t	84 t	84 t	79.5 t	84 t

MARS MISSION MANIFESTS - CRYO/AEROBRAKE AND NTR 84-120 T HLLV

FIGURE 5-10

	Flight #	1	2	3	4	5	6	7-12
2015 Mars Departure	Launch vehicle	Shuttle Z	Shuttle C	Shuttle Z	Shuttle C	Shuttle Z	Shuttle Z	Shuttle Z
	Manifest	MTV aerobrake, assembly equipment	Crew systems, ECCV, structure, comsats, consumables	TEIS O2 propellant, tankage	TEIS H2 propellant, tankage & engines	MEV aerobrake, assembly equipment	Ascent vehicle, descent vehicle, surface payload	MTV - Earth departure propellant & engines
	Mass	41.0 t	50t	91t	15t	12t	75t	91t each launch

	Flight #	1	2	3	4	5	6 - 9
2017 Mars Departure	Launch vehicle	Shuttle Z	Shuttle C	Shuttle Z	Shuttle C	Shuttle C	Shuttle Z
	Manifest	MEV 1 aerobrake, assembly equipment	50t payload, descent stage for MEV1	MEV 2 aerobrake, assembly equipment	50t payload, descent stage for MEV 2	MEV1-2 & TMIS inter -connect structure, Nav kit	TMIS engines and departure propellant
	Mass	12 t	70t	12t	70t	20t	91t each launch

	Flight #	1	2	3	4	5	6-11
2018 Mars Departure	Launch vehicle	Shuttle Z	Shuttle C	Shuttle C	Shuttle Z	Shuttle Z	Shuttle Z
	Manifest	MTV aerobrake, assembly equipment	Crew Systems ECCV, structure, consumables	TEIS propellant tankage O2 & H2	MEV aerobrake, assembly equipment	Ascent vehicle descent vehicle, surface payload	MTV- Earth departure propellant & engines
	Mass	41t	50t	69 t	12t	75 t	85 t each launch

MARS MISSION VEHICLE MANIFESTS BY YEAR, SHUTTLE C/Z

FIGURE 5-11

changed to keep in step with the configuration. Consequently, this work lags the current vehicle concepts subsequent to that date. Modifications can easily be added to the documented concepts as an advancement or enhancement.

Once the above guidelines were set, Integration Compatibility Analyses were performed for the three advanced propulsion vehicles. A significant asset to the activity was that the major systems, which comprised the NEP, SEP, and NTR, were designed to fit within a 10 meter shroud in either their integrated or packaged configuration.This allowed manifesting and packaging studies to proceed without initial major impacts to the vehicles. An illustration is provided of the inter-relationship of design with assembly sequence. The manifests lists for the NEP is given in Figure 5-12. The packaging for each NEP assembly launch is shown in Figures 5-13 and 5-14 for both an HLLV and mixed fleet Earth to orbit transfer vehicle sets. The payload mass per assembly launch for both the HLLV and mixed fleet sets is shown in Figure 5-15. In spite of the design planning, several areas were identified as potential problems and/or problem-solutions which warrant future concentration.

An assumption was made that the aerobrake for each of the vehicles must be assembled in orbit (the NEP, SEP, and NTR use propulsive captures at Mars and Earth, but use an aerobrake for landing on the Martian surface). The current packaging scheme for an aerobrake required two dedicated HLLV flights in addition to that required for the MEV itself. This impact may indeed be mitigated by the integrated aerobrake launch concept (Figure 5-16). If the MEV can be partially integrated with its aerobrake at launch, the three assembly flights currently required may become only one. As the aerobrake dedicated HLLV flights currently use only a small percentage of the 140 t capability, some other solution, like an integrated launch, would be more effective.

The aerobrake launch exemplifies the many assembly flights which are volume, and not mass, limited. Many of those flights which are mass critical actually involve propellant launches. A significant input to the propellant question (at least for the cryo and NTR vehicles) comes from an on-orbit cryo depot trade study (Reference 33). Due to the majority of volume restricted flights, the ideal launch vehicle may prove to have a large shroud with only moderate lift capability or the vehicle assembly flights may possibly be accomplished by some mixed launch fleet. A mixed fleet may also improve Earth-to-orbit (ETO) launch rates, depending on the mission scenarios chosen. Details of how the vehicles would use a mixed fleet are given in References 6 to 11.

NEP Component	Quantity	Dimensions (meters)	Total Mass (metric tons)
• MEV			
Aeroshell	1	28 x 30 x 7 box	9.51
Descent System (incl 2 rovers)	1	9.5 x 20 x 4 box *	32.83
Ascent System	1	9.5 x 9.5 x 5.5 box	24.83
Surface Payload Module	1	13 x 4.4 (dia) cylinder	25.00
Surface Payload Module Airlock	1	2.9 x 3 (dia) cylinder	4.50
		Subtotal =	96.67
• MTV			
MTV Hab Module	1	10 x 8 (dia) cylinder	40.30
MCRV	1	3 x 4 (dia) cylinder	7.00
MTV-to-MEV Tunnel and Airlock	1	6 x 3 (dia) cylinder	7.00
Main Truss	2	7 x 7 x 7 box (deployable) *	4.60
Power Conditioning	1	2 x 1 x 1 box	1.80
Communications	2	2 x 1 x 2 box	0.60
Attitude Control	2	2 x 2 x 2 box	5.70
Avionics	1	2 x 2 x 2 box	2.50
Power Conditioning and Control	4	5 x 2 x 2 box	32.80
Thruster Pods	4	25 x 2 x 5.5 box	54.80
Propellant and Tanks	5	4.1 (dia) sphere	185.00
		Subtotal =	342.10

NEP Component	Quantity	Dimensions (meters)	Total Mass (metric tons)
• Power Generation and Heat Transport			
Reactor #1	1	1 x 2 (dia) cylinder	7.4
Reactor #2	1	1 x 2 (dia) cylinder	7.4
Shield #1	1	0.8 x 2.25 (dia) cylinder **	5.4
Shield #2	1	0.8 x 2.25 (dia) cylinder **	7.2
Primary Heat Transport System			
Auxiliary Cooling Subsystems		9.5 x 9 (dia) cylinder *	71.5
Boiler			
TurboAlternators	1		
TurboPumps			
Rotary Fluid Management Device			
Piping and Auxiliaries			
		Subtotal =	98.9
• Radiators			
Main Cycle Radiators	2	15 x 6 (dia) cylinder *	10.7
Auxiliary Radiators	2	15 x 4 (dia) cylinder *	5.1
		Subtotal =	15.8
		NEP Total =	553.47

* These represent launch package dimensions, not mission configuration
** Total Power Generation and Heat Transport launch package dimensions = 15 x 9 (dia) cylinder

NEP COMPONENT MANIFEST DATA

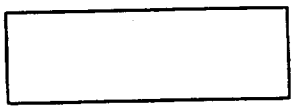
FIGURE 5-12

Ground Rules and Assumptions

- Heavy Lift Launch Vehicle (HLLV) with 140 metric ton capability and 10m x 30m shroud
- Sequencing based on External Tank-derived Assembly Platform concept
- Some TBD volume is available in nosecone of HLLV
- No specific FSE/OSE or CG constraints identified (heavier payload located at bottom of stack)
- NEP configuration (volume and mass) current as of 3rd Quarterly Briefing
- MEV Aeroshell is assembled on orbit (in ten pieces) and requires two dedicated HLLV flights

Assembly Mission One

- MEV Aeroshell (5 out of 10 pieces)

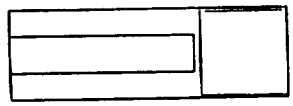


Assembly Mission Two

- MEV Aeroshell (5 out of 10 pieces)

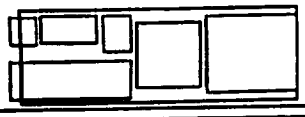
Assembly Mission Three

- MEV Descent System (incl 2 rovers)
- MEV Ascent System



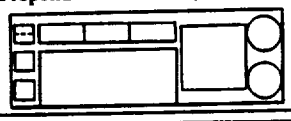
Assembly Mission Four

- Mars Surface Payload Module
- Mars Surface Module Airlock
- MTV-to-MEV Airlock and Tunnel
- MCRV
- Main Truss (1 of 2)
- MTV Hab Module



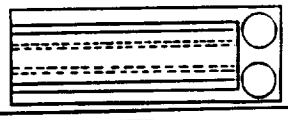
Assembly Mission Five

- Attitude Control (2)
- Communications (2)
- Avionics (1)
- Power Conditioning (1)
- Power Distribution and Control (4)
- Main Cycle Radiators (1 of 2)
- Main Truss (2nd of 2)
- Propellant and Tanks (2 of 5)



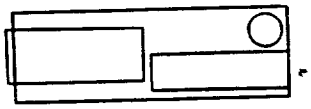
Assembly Mission Six

- Thruster Pods (4)
- Propellant and Tanks (3rd & 4th of 5)



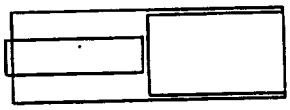
Assembly Mission Seven

- Main Cycle Radiators (2nd of 2)
- Auxiliary Radiators (1 of 2)
- Propellant and Tanks (5th of 5)



Assembly Mission Eight

- Auxiliary Radiators (2nd of 2)
- Reactor/Shield/Turbo Assembly



NEP MANIFESTING AND PACKAGING - 140 mt HLLV

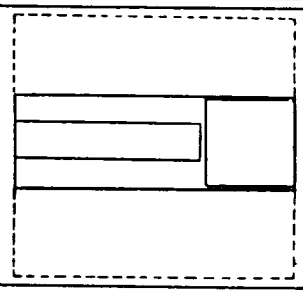
FIGURE 5-13

Ground Rules and Assumptions

- Heavy Lift Launch Vehicle (HLLV) mixed fleet consists of:
  - HLLV #1: 84 metric ton payload capability with 10m x 30m shroud
  - HLLV #2: 120 metric ton payload capability with 7.6m x 30m shroud
- Sequencing based on External Tank-derived Assembly Platform concept
- Some TBD volume is available in nosecone of HLLV
- No specific FSE/OSE or CG constraints identified (heavier payload located at bottom of stack)
- NEP configuration (volume and mass) current as of 3rd Quarterly Briefing
- MEV Aeroshell is assumed to be integrated at launch ("Ninja Turtle" concept) with other payload packaged in shroud

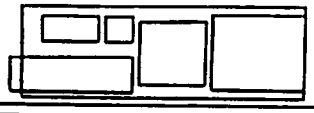
Assembly Mission One (HLLV #1)

- MEV Aeroshell (externally mounted)
- MEV Descent System (incl 2 rovers)
- MEV Ascent System



Assembly Mission Two (HLLV #1)

- Mars Surface Payload Module
- Mars Surface Module Airlock
- MTV-to-MEV Airlock and Tunnel
- Main Truss (1 of 2)
- MTV Hab Module

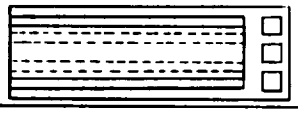


Assembly Mission Three (HLLV #1)

- Attitude Control (2)
- Communications (2)
- Avionics (1)
- Power Conditioning (1)
- Power Distribution and Control (1)
- Main Cycle Radiators (1 of 2)
- Main Truss (2nd of 2)
- MCRV
- Propellant and Tanks (1 of 5)

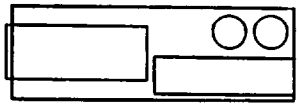
Assembly Mission Four (HLLV #1)

- Thruster Pods (4)
- Power Dist & Control (3 of 4)



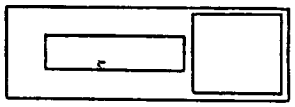
Assembly Mission Five (HLLV #1)

- Main Cycle Radiators (2nd of 2)
- Auxiliary Radiators (1 of 2)
- Propellant & Tanks (2nd, 3rd of 5)



Assembly Mission Six (HLLV #1)

- Auxiliary Radiators (2nd of 2)
- Turboalternator/generator Assembly

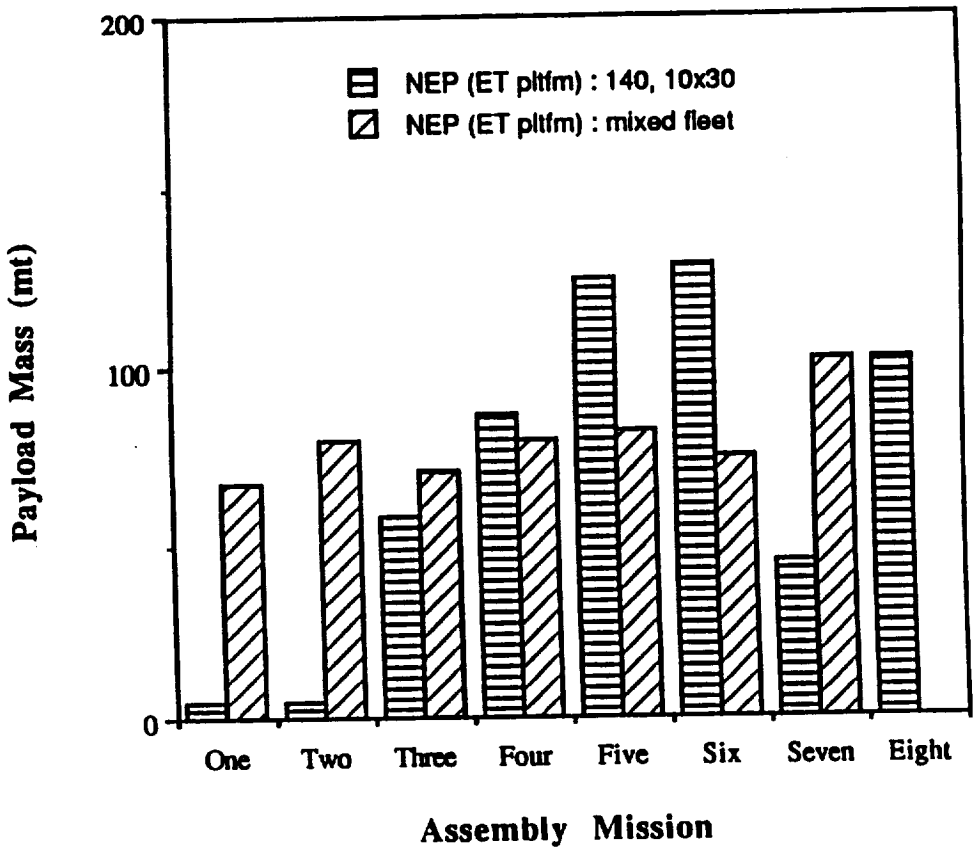


Assembly Mission Seven (HLLV #2)

- Reactors (2 of 2)
- Shields (2 of 2)
- Propellant & Tanks (4th, 5th of 5)

NEP MANIFESTING AND PACKAGING, MIXED HLLV FLEET

FIGURE 5-14

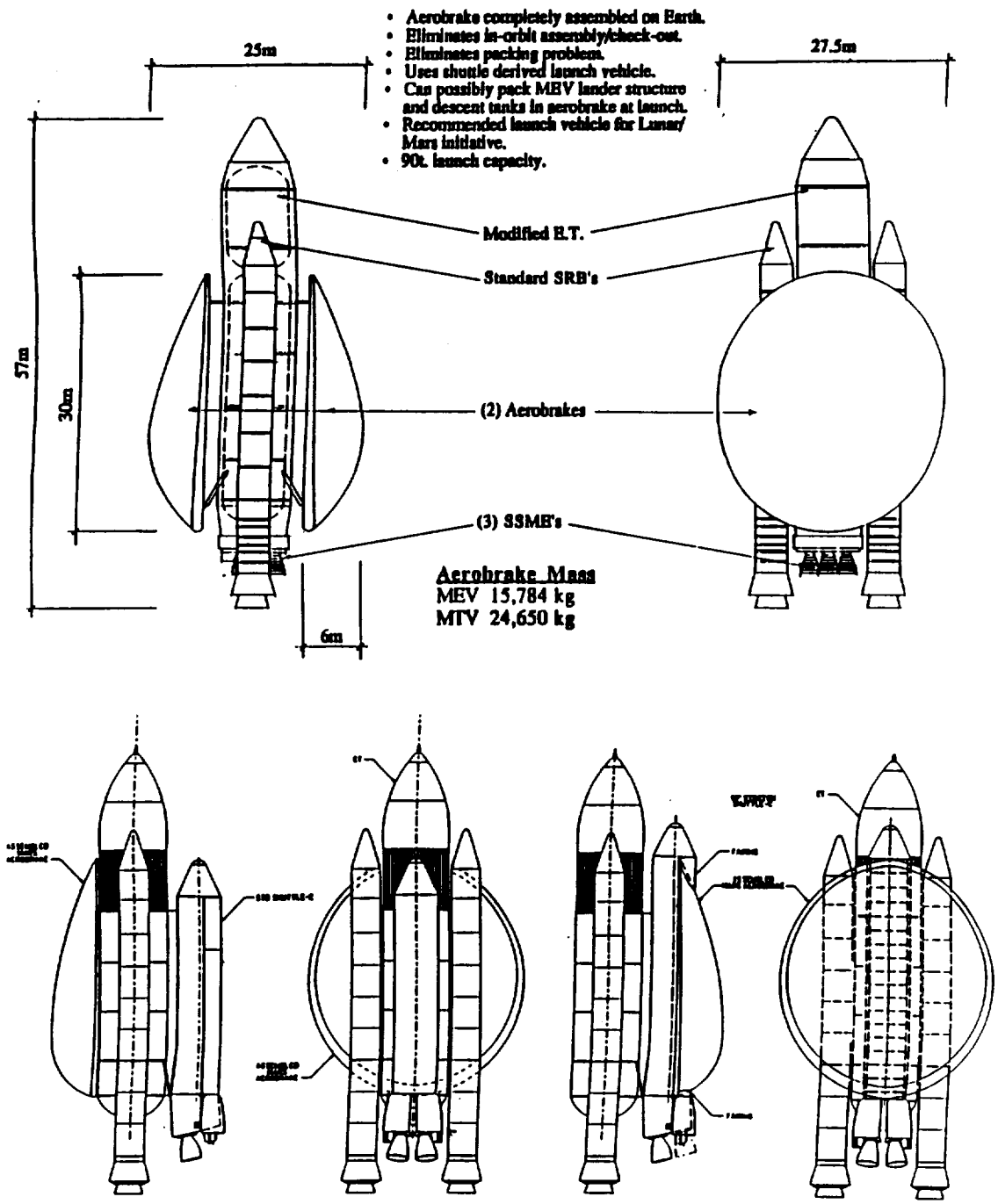


NEP MANIFESTING AND PACKAGING, PAYLOAD MASS PER ASSEMBLY
FIGURE 5-15

Defining concepts for different on-orbit infrastructures to examine their effect on vehicle manifesting and launch sequencing was also performed. Much of this effort was expended in considering different assembly node concepts. In the preliminary work, manifests and sequences were derived for several of these concepts to show their interdependence.

There also appears to be a need for some intermediate on-orbit vehicle for transferring cargo from the HLLV to the assembly site, transporting crew from SSF, etc. A cargo transfer vehicle (CTV), capable of both pressurized and unpressurized transfer is seen as a possible candidate. The CTV is basically a vehicle which would be sized to carry out the tasks of HLLV unloading/maneuvering and vehicle fueling, within the confines of low Earth orbit. The CTV definition has involved developing a list of CTV functions and providing some performance and operations information for each; this may provide some

basis for preliminary mission definition and configuration development. The actual capabilities of a CTV would impact how each vehicle may be packaged and, ultimately, how it would be designed.



AEROBRAKE LAUNCH OPTIONS, (TOP) SHUTTLE (BOTTOM) SHUTTLE-C
FIGURE 5-16

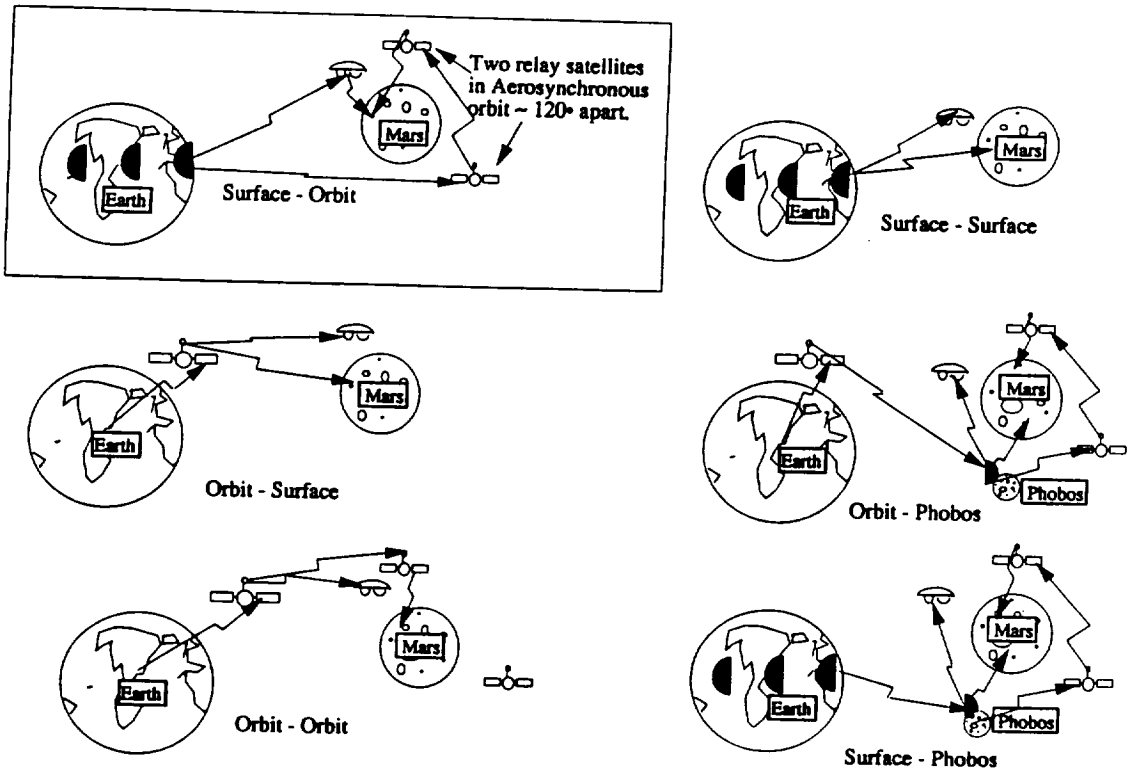
Another aspect of this work involves the interfaces between subsystems and systems of the integrated vehicle as well as those between the vehicle and the launch vehicle, CTV, assembly node, etc. (reference integration section above). These concerns directly translate into the sequencing necessary for verifying these interfaces both on the ground and in orbit. The ground processing is inter-related to manifests being defined here. One example of this interface verification need produced the assumption that the MTV habitat module should be kept on the ground as long as possible for checking the interfaces between it and the entire vehicle with which it communicates throughout the mission. However, it was discovered that this goal tends to complicate launch sequencing and to decrease some of the efficiency of HLLV packaging. Alternatives such as verifying only physical connections early with the actual module and then using simulators later for some of the data interfaces may alleviate these problems and allow the habitat to be launched at the optimum time. This affects the test plan and philosophy chosen for a particular vehicle.

Obviously, much of the current work has not been constrained to any particular vehicle or on-orbit concept as the SEI Program is still fluid. Once a specific Mars vehicle, HLLV (or fleet), schedule, on-orbit assembly infrastructure, and assembly methodology are chosen, these types of analyses will mature. However, these continuing parametric studies accompanied by the identification of issues and resolutions which they produce will prove to define the preliminary concepts and capabilities which should be pursued.

5.3 Communications

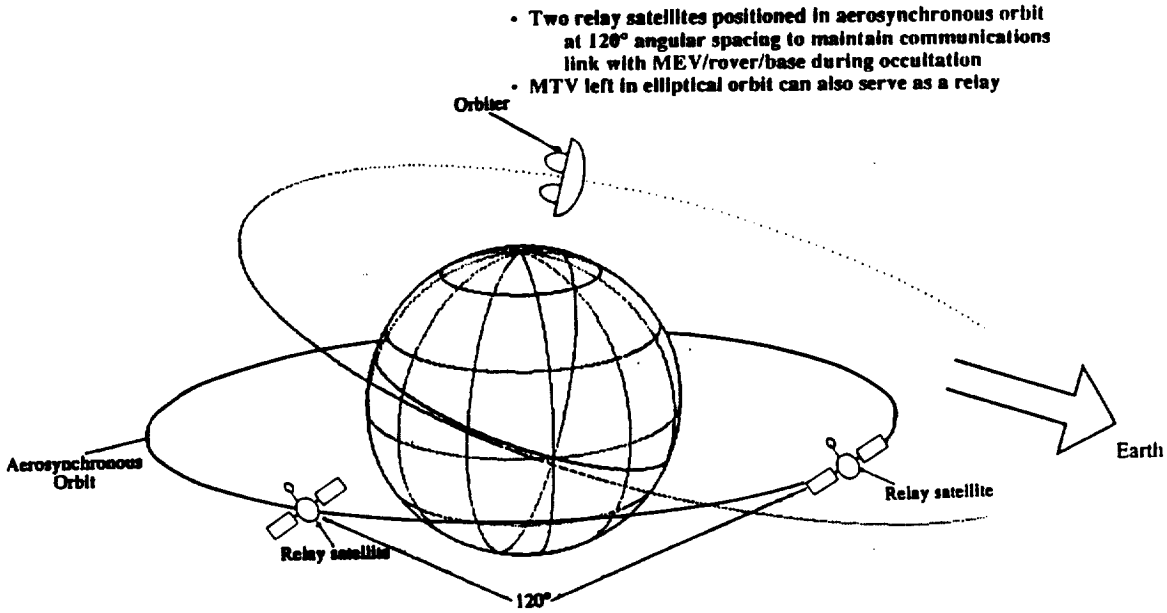
Early in this study rudimentary link analyses on several types of systems were performed to examine their suitability for Earth-Mars communications. For these analyses the "Optical Communications Link Analysis Program, version 2.01" was used (Reference 34). The frequencies examined were some of the NASA standard systems (S-band, X-band, Ka-band and Ku-band) and newer optical systems (double ND:YAG laser and CO_2 laser).

Several scenarios for completing an Earth-Mars communication link are shown in Figure 5-17. One or more of these scenarios with possible SSF and lunar base communication links would form the bases of the communications network. The Mars end of such a system with satellites placed at aerosyncronous orbits spaced 120° apart is illustrated in Figure 5-18. By coordinating the satellite orbit period with Mars rotation, uninterrupted



LINK SCENARIOS
FIGURE 5-17

contact can be maintained for ≥ 95% of the time. These types of networks will work well for S, X, Ka and Ku frequencies, but if the millimeter, infrared or optical bandwidths are used, the existing dish antennas must be replaced with high elevation telescopes or geosynchronous satellites. These systems must be able to support missions in which the free space communication distance varies from ~1.0 astronomic unit (A.U.) for an opposition mission to ~2.5 A.U. for a conjunction mission, with possible interference from the Sun when it is positioned between the vehicle and Earth. The communications data rate will change with an inverse proportionality with the link distance. The data rate requirements set forth in Reference 35, shown in Figure 5-19, were reviewed for this study. The high rate video listed is referred to as "real time" video. A minimum of 100 Mbps (megabits per second) data rate is needed for this type of video. For science communication without data storage, about 300 Mbps is required. While no data rate problems exist directly for the vehicle, indirectly the vehicle design may be affected by the choice of the communications medium used to obtain the required data rates. If X-band is used, a data compression ratio of 100:1 will be needed to obtain just the real time video.



TYPICAL COMMUNICATIONS RELAY SATELLITE CONFIGURATIONS

FIGURE 5-18

Type of Data	Data Rate (Mbps)	Comments
High Rate Video	100	High rate video - single channel, color, 512 x 512 pixels, 8 bits/pixel, 30 frames/sec
Low Rate Video	1.0	Low rate video - single channel, mono, 1024 x 1024, 8 bits/pixel, 0.1 frames/sec
Voice	0.02	Single channel links with manned vehicles, rovers, or base sites
Science	1.0 ≤ 1.0	Low rate imagery - single channel, mono, 1024 x 10214, 8 bits/pixel, 0.1 frames/sec Low duty cycle spectral scanning or SAR imaging with data storage available
	≤ 300	No data storage, spectral scanning or SAR imaging
Engineering	0.20 0.002	Per manned spacecraft Per manned spacecraft
Telerobotics	0.2 200	Command channel per rover Stereo video, color, 512 x 512 pixels, 8 bits/pixel, 30 frames/sec
Command Data Load	≤ 0.002 ≤ 1.0	Per spacecraft or science platform/site Earth to manned vehicles or manned outposts, access to Earth data bases

DATA RATE ALLOCATIONS

FIGURE 5-19

Science and telerobotics will likely need higher data compression ratios. As a comparison, Ka - band will need at least a 10:1 compression ratio for video. A further comparison of the various systems capabilities is given in Figure 5-20. To handle these data rates large antennas will be needed by current technology, which will add to the mass of the vehicle, Figure 5-21. Advanced communications schemes will require technical developments in pointing and tracking methods. The sophisticated electronics that must be used in these systems may become volume limited. Additional vehicle impacts that are expected for several types of communications systems are indicated in Figure 5-22. Not all of the possible size, mass, complexity and operations concerns have been identified for all candidate communications systems nor appropriate applications for individual system fully identified. A preliminary state-of-the-art assessment, matching frequency with application, for several systems is given in Figure 5-23. Additional work in this communication area is needed.

Parameter	S-band (Viking)	X-band	Ka-band *	Double Nd:YAG *#
Satellite at Mars				
Transmission rate to Mars (Mbps)	0.058	6/1.5	100/25	1000
Transmission rate from Mars (Mbps)	0.058	6/1.5	100/25	1000
Transmitter Power (W)	20	20	50	12
Receiver Power (W)	20	20	50	12
Transmitter size (m)	64	70/34	70/34	1.0
Receiver size (m)	1	3.6	5	0.5
Transmission distance (AU)	1.0 §	1.0	1.0	1.0
Frequency (GHz)	381	8.4	32.4	0.532 microns
Mars Surface				
Transmission rate to Mars (Mbps)	0.043	29/7.2	100/25	1000
Transmission rate from Mars (Mbps)	0.043	29/7.2	100/25	1000
Transmitter Power (W)	20	50	50	6
Receiver Power (W)	20	50	50	6
Transmitter size (m)	64	70/34	70/34	10
Receiver size (m)	1	5	5.0	1
Transmission distance (AU)	1.0	1.0	1.0	1.0
Frequency (GHz)	381	8.4	32.4	0.532 microns
Mars Rover				
Transmission rate to Mars (Mbps)		0.12/0.029	0.4/0.1	
Transmission rate from Mars (Mbps)		0.12/0.029	0.4/0.1	
Transmitter Power (W)		20	20	
Receiver Power (W)		20	20	
Transmitter size (m)		70/34	70/34	
Receiver size (m)		0.5	0.5	
Transmission distance (AU)		1.0	1.0	
Frequency (GHz)		8.4	32.4	

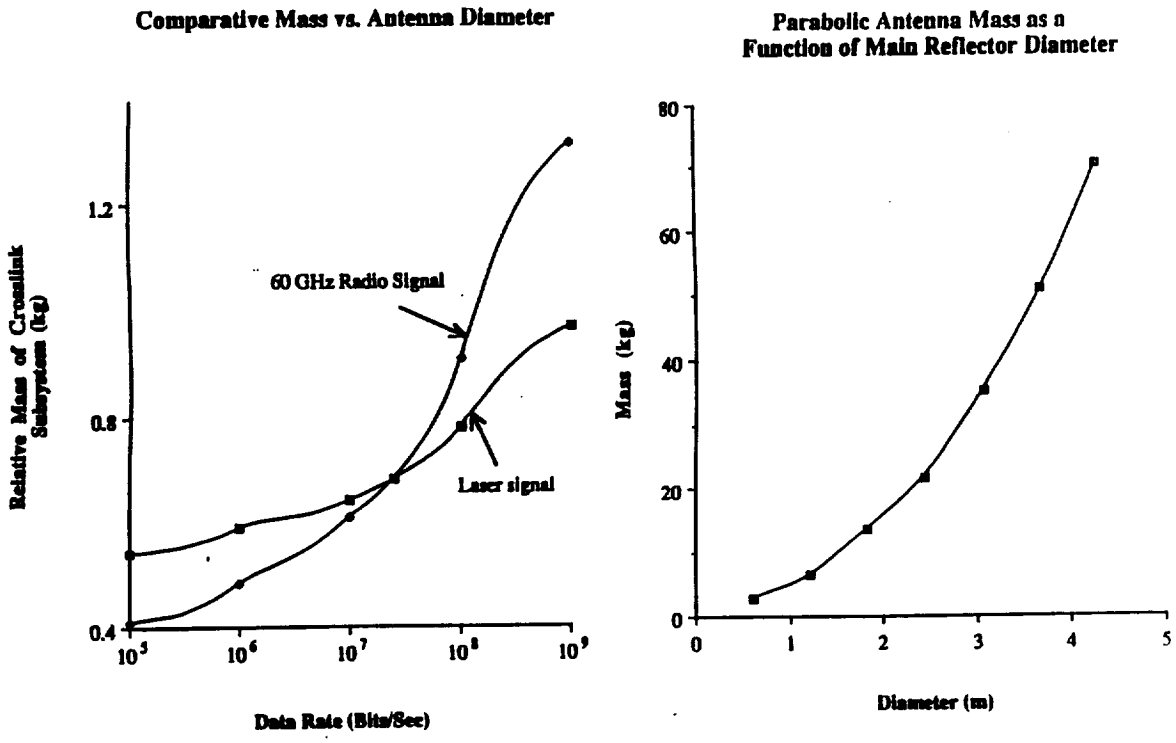
* May utilize data compression depending on current technology

§ Distances are of the order of 1.0 A.U. but vary ± 0.6.

\# Nd:YAG - Neodynium:Yttrium Aluminum Garnate

COMPARATIVE COMMUNICATION CAPABILITIES

FIGURE 5-20



ANTENNA MASS

FIGURE 5-21

	X-band	Ka-band	Millimeter Wave	Infrared	Optical
Characteristics	• Well developed technologies • Less sensitive to weather attenuation than Ka-band • Proven reliability	• Smaller antenna size than X-band • Capable of 'real time' video using 10:1 data compression	• Low interference from solar noise • Low hardware mass • High power potential • High data rates	• High data rates • Low hardware mass • Space qualified (CO_2) • High power potential	• High data rates (300 Mbps + for Mars-Earth downlink) • Smaller aperture transmitter and receiver • Low hardware mass • Power savings over RF
Vehicle Impacts	Design considerations: large antenna size and high hardware mass will affect configuration of vehicle.	While antenna size and hardware mass are less than X-band, they are still design considerations	Technical advances in high power travelling wave tube amplifiers and coherent power combining necessary. May impact vehicle design.	Lower volume and mass a plus in vehicle design. Development of more accurate pointing and tracking devices may become a design consideration.	Development in phased array technology, advanced pointing and tracking instruments may be design considerations.

COMMUNICATIONS SELECTION RATIONALE

FIGURE 5-22

Frequency	Application	Advantages	State of the Art
Optical, IR, mm - wave	• In transit MTV/MEV communication • Earth orbit - Mars orbit communication	• Large data rates • Power savings • Mass savings • Volume savings	• Optical technology rapidly developing. Optical system feasible based on 1988 technology. • IR, mm - wave technology requires development in long - life systems. 1988 technology capable of 100 GHz + bandwidth in near IR using mode-locking
mm - wave Ka - band	• Mars orbit - surface communication • Mars surface - surface communication	Less likely to suffer from atmospheric scattering and absorption than IR.	• Ka- band technology well developed for this use • mm - wave technology should be adequate by 2000
Ka - band	• MTV tracking using moderate gain	Fast, accurate location of spacecraft for rapid laser link - up to minimize pointing loss to within 1/2 of angular beam width.	• Ka - band tracking system is feasible based on 1988 technology.

COMMUNICATION CANDIDATES

FIGURE 5-23

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6. SUPPORT REQUIREMENTS

Two areas of support requirements are anlayzed for the vehicle concepts, on-orbit assembly operations and the facility requirements, with associated ground processing operations. Details are provided in the following sections.

6.1 On - Orbit Analysis

The details of the on - orbit assembly operations for the Mars mission vehicles (MMVs) are provided in this section. Each MMV concept has unique characteristics which require different assembly techniques and on-orbit equipment and facilities. This study began by conceptualizing the facilities required for on-orbit assembly.

6.1.1 On - Orbit Facilities

On-orbit assembly facilities concepts were developed during this study to evaluate the requirements for transportation, handling, storage, debris protection, and assembly operations. Results from this study showed that assembly operations were dependent upon the capability to manifest the MMV systems in a heavy lift launch vehicle (HLLV) and the launch processing time required by the HLLV. An assumption in the study was that there would be four HLLV launches per year. As a result, the average on-orbit assembly time is two years for the different MMV concepts. This long duration time on-orbit defines debris protection to be a primary requirement for the vehicle and assembly facility.

Data show that since the 1985 Space Station debris environment baseline, debris flux has increased 4-5 times in magnitude. By the year 2016, interpolation of the present data shows that the debris environment will be 20-25 times worse than the baseline environment for Space Station. The current concept for debris protection of Space Station is a buffer shield of aluminum 0.05 inches thick, with a 4.3 inch stand off from the outer pressure shell of the Space Station module. This shielding concept was used for weight comparison during the assembly facility analysis.

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The on-orbit assembly analysis began with the cryo/aerobrake MMV. A concept for an assembly facility was developed using expended National Space Transportation Systems (NSTS) external tanks (ETs), Figure 6-1. The approach was to structurally attach two rows of six ETs to form the debris shield and a basis for an assembly platform. The twelve ETs require approximately 50,000 pounds of additional propellant, modification to existing equipment and support structure. However, a debris shield constructed similar to Space Station concept and providing the same coverage area would weigh approximately 150,000 pounds. Other major advantages of this concept are: utilization of presently disposed equipment, greater debris protection provided by several buffer walls, and a facility which features internal storage space. Several studies have already been conducted to evaluate concepts for using the ETs as orbital nodes, satellite stations, and other facilities. A major concern with this concept is the external insulation (SOFI) would require containment to prevent deterioration and creation of new space debris.

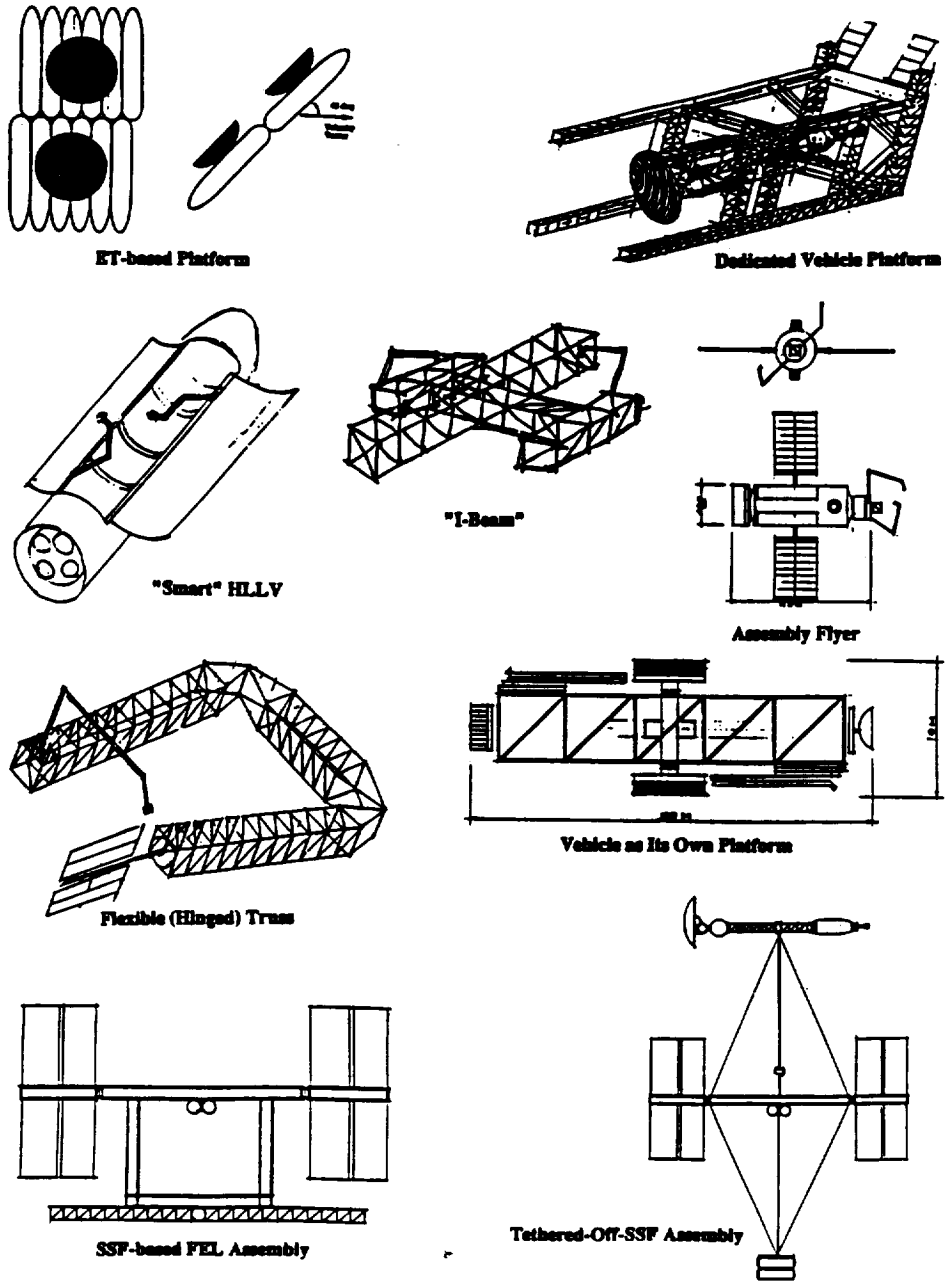
Debris protection is a passive requirement of the assembly facility, that is, the shielding is an integral part of debris sesnsitive areas of the facility structure. The active requirements of the facility are defined as follows:

- Integration of mission vehicles including assembly and launch processing
- De-integration of mission vehicles for repair
- Re-integration of mission vehicles for resupply and refurbishment

To meet these requirements the facility must provide capabilities for storage fixtures, assembly jigs, assembly robotics, power, guidance, attitude control, reboost and a communication link to the ground. Special equipment required for assembly operations will be discussed in the following sections.

The Mars mission vehicles are large in size, for instance the solar electric powered (SEP) vehicle is two hundred meters by two hundred meters square. Several assembly facilities concepts developed during this study vary in size to accommodate the various MMVs. These concepts are illustrated below in Figure 6-1. Several advantages / features and disadvantages of each on-orbit assembly facility are shown in Figure 6-2. These facility concepts range from a very large dedicated facility (that is entirely self-contained) to concepts that utilized the vehicle structure itself as an assembly platform. The key point is that none of these facilities satisfy all the vehicles' assembly requirements. A separate assembly facility can also support other uses following assembly of the MMV, such as

Scientific Orbital Experiments, communications relay station, or assembly / refurbishment facility for other mission vehicles. The conclusion of this study shows that some type of assembly facility is required for debris protection and assistance in assembly operations. The specfic configuration and capability of this facility is dependent upon the particular MMV chosen and its individual requirements.



ON-ORBIT ASSEMBLY FACILITY CONCEPTS

FIGURE 6-1

Node Concepts	Key Features/Advantages	Key Disadvantages
Assembly Flyer Platform	• Performs HLLV unloading, payload/crew transport, and assembly with one vehicle • Compatible with SSF • Capable of manned/robotic operations • Uses CTV for main p/a • Can serve as free flying platform between assemblies	• No additional storage • Requires vehicle to have additional control and reboost systems • Requires development and production of sophisticated man-rated space vehicle • Requires localized debris shielding
SSF Based Assembly of First Element	• Uses planned SSF growth concept • Provides quidk and easy crew logistics access to initial assembly operations • Allows verification and checkout of critical systems prior to independent vehicle operations • Does not disrupt SSF operations beyond first assembly mission (remainder of assembly accomplished by vehicle itself as platform)	• Impact to SSF (resources, microgravity, drag, etc.) • Eventually requires vehicle to have additional control and reboost systems • Requires localized debris shielding • No additional storage beyond first element
Tethered off-SSF Assembly Platform	• Compatible with current SSF design • Provides quick and easy crew and logistics access to entire assembly and propellant transfer operations • Microgravity and dynamic loads impacts to SSF minimized by tether • Removes hazardous operations and materials to SSF standoff distance	• Impact to SSF resources • Requries localized debris shielding • No additional storage • Requires additional reboost and control systems on SSF
Dedicated Assembly Node	• Abundant storage • Totally self-contained • Vehicle systems unused • Multiple robot arms • Sections of vehicle may be assembled simultaneously	• Larger than SSF • Will take long time to construct • Excessive reboost requirements • Mechanically complex
I-Beam Platform	• Simple • Can easily reach most parts of vehicle • Brought up in first assembly mission • Uses vehicle systems • Can serve as base for experiments	• Fuel cells, batteries required for initial deployment • Vehicle will have to provide comm., power, reboost • Limited storage area • Precursor mission required for deployment
"Smart" HLLV Platform	• No additional platform required • HLLV shroud provides limited debris shielding • HLLV provides for comm., data., rcs, GNC, etc. • Robot arms transferable to NTR	• Increased HLLV complexity • Reboost fuel has to be replenished • Limited storage • Vehicle must be detached from HLLV prior to assembly complete
Hinged Truss Platform	• Uses vehicle truss as assembly platform; no other platform needed • Reach to remote engine section of vehicle provided by flexing truss at hinges • Vehicle subsystems used; no additional systems necessary	• Requires a precursor mission to deploy truss • Batteries, fuel cells necessary for initial deployment • Rboost, comm., data, power, must be in place prio to assembly start • Limited storage • Local debris shielding required
Vehcile as its own Platform	• Reduces needed on-orbit infrastructure • Deletes additional facilities and resources needed for designing, builiding, launching, and maintaining separate assembly platform	• Requires dedicated HLLV flight for non-optimized packaged first element • Requires vehicle to have additional control, reboost • No additional storage • Requires batteries or fuel cells for initial deployment • Requires localized debris shielding

ON-ORBIT FACILITY CONCEPTS ADVANTAGES AND DISADVANTAGES

FIGURE 6-2

6.1.2 On - Orbit Operations

On - orbit operations for this study were limited to assembly of the MMV and did not consider explicitly the construction or assembly of the on-orbit facility or fuel top-off missions prior to launch processing. Assembly operations for each vehicle concept include the launch processing of the HLLV, orbit stabilizing of the HLLV at the assembly facility, transportation of payload from the HLLV to the assembly facility, assembly of the vehicle, and test and verification of vehicle systems prior to launch processing.

An in depth on-orbit assembly operation analysis was developed for the cryo/aerobrake, nuclear thermal reactor (NTR), nuclear electric power (NEP), and solar electric power (SEP) MMV (References 8 - 10). Assumptions for this analysis were defined as follows:

- Assembly completed primarily with ground controlled robots. Crew operations were limited to contingency, critical inspections, repair of robotic systems, internal system checkout, and critical assembly operations.
- Processing of the launch vehicle requires 90 days between missions.
- Robotic assembly equipment can perform alignment and mechanical fastening as required.
- Task duration times were developed to ensure consistency throughout the assembly analysis.

This analysis developed an initial top level flow of on-orbit operations. Manifesting of the systems in a 10 meter diameter payload shroud on a heavy lift launch vehicle (HLLV) was studied to determine the launch sequence as required by the on-orbit operations. For all launch missions, the HLLV would achieve a stable orbit adjacent to the assembly facility. A cargo transfer vehicle (CTV) stationed at Space Station would then transfer to the assembly node orbit. The primary task for the CTV is to transfer vehicle components and systems from the HLLV to the assembly facility. At the assembly facility, the CTV will hand-off the payload to the platform robotic manipulating system (PRMS). The PRMS performs the operations of storing the payload for future operations and/or unpacking components for assembly and placing them on the assembly fixture / jig. The platform anchor system (PAS) was derived as an assembly jig capable of transferring systems from one assembly site / location to another. The sequence described below is the initial on-orbit assembly scenario for the cryo/aerobrake vehicle.

A complete on-orbit assembly sequence was developed in detail in Reference 7. The top level sequence and rationale for decision is provided. First assembly mission operations were to launch the mars transfer vehicle (MTV) habitat module and surface payload module to the assembly facility. The habitat module would provide crew stationing, communications, data and local control of the robotic assembly systems. Sufficient "burn - in" and check out time for the habitat systems would be provided by having the module present throughout the assembly sequence. The surface payload module would not have the Mars science experiments on-board, until the fourth mission.

The next two assembly missions include operations for assembly of the Mars excursion vehicle (MEV) aerobrake. The aerobrake was manifested in five sections for each mission. The time duration for each mission is approximately 8 days, of 16 hours per day. Based upon the 90 days between missions for vehicle processing the analysis indicates that each aerobrake requires two launch missions and 180 days of on-orbit operations. This is a major point developed by this study.

The MEV systems, descent lander, ascent systems, and surface and science payload are brought up on the next mission and assembled to the completed aerobrake structure. Each system has interfaces which require verification after the assembly operations. These interfaces are verified robotically if feasible and internal interfaces / systems are verified by system self - check or by on-orbit assembly crew.

The MTV aerobrake assembly operations are identical to the MEV aerobrake and are given in the next two missions. The MTV systems, the trans earth injection system (TEI), and habitat module are then assembled to the aerobrake structure. During assembly of the habitat module to the aerobrake, crew, if present, would be stationed from the MEV ascent / descent module. Internal systems of the MTV would be verified using the same operations required for internal systems of the MEV.

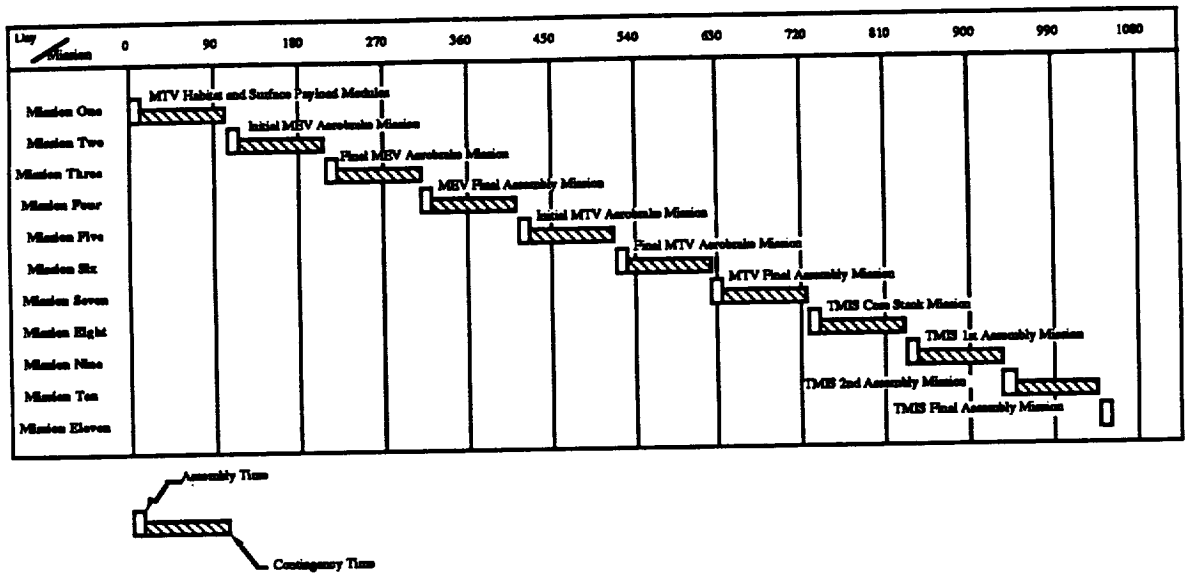
A major assembly operation for this vehicle is the task of assembling the MEV to the MTV system. This study developed a concept of assembly operations and supporting structure for the final MMV configuration. The thermal protection system (TPS) of the aerobrakes is facing the debris shielding of the assembly facility. Once assembled, the MEV and MTV systems are rotated into mating configuration and secured. The final stage of assembly operations is to mate the trans mars injection system (TMIS) to the MMV. These operations require the final four assembly missions.

The TMIS consists of a core stack assembly, including the engines, and three propellant tanks. The core stack is first assembled to the MMV and then each propellant tank is assembled during separate missions. Limitations of the HLLV used for this study require a propellant top-off mission. Once the interfaces of the TMIS are verified the MMV is ready for launch processing.

The above assembly scenario was used in developing an on-orbit assembly schedule to determine the time required for assembly using a given mission launch from orbit date and calculating the earliest start date. The duration times assigned to each task were chosen and held consistently throughout this study. This study showed that the total on-orbit assembly time is dependent on the 90-day launch processing required by the HLLV. Figure 6-3 shows the assembly time per mission for the cryo/aerobrake vehicle and the contingency time remaining until the next assembly mission begins. Further analysis of the duration times and assembly operations are required to validate the data below. However, even if the assembly duration times are a magntiude greater than those selected for this study, the total assembly time is limited by the 90-day HLLV launch intervals.

Similar analyses were performed for the NTR, NEP, and SEP vehicles. Results showed that the assembly sequence is dependent on the particular assembly facility concept used for a particular MMV. The on-orbit analyses were developed to the same level of detail as the cryo/aerobrake analysis. The ground rules and assumptions used for the on-orbit assembly analyses of the NTR, NEP, and SEP are as follows:

- Based on Mars vehicle (NEP, SEP, and NTR) configurations as of 8/15/90.
- Baseline Earth-to-Orbit (ETO) Vehicle (HLLV) has 10m x 30m shroud with with 140 mt payload capability.
- HLLV nosecone has some additional TBD volume for launch element packaging.



CRYO/AEROBRAKE ACTUAL ASSEMBLY TIME
FIGURE 6-3

- Nominal 85% payload packaging and mass factors used for HLLV manifesting
  (propellant tanks may be accepted.
- HLLV has a nominal 3 to 7 day station-keeping ability.
- HLLV unloaded piece by piece by cargo transfer vehicle (CTV).
- Crew transported to assembly location from SSF via ACRV.
- CTV will be designed to support all identified manned/unmanned operations (on-orbit refueling may be available via on-orbit depot, HLLV provisioning, the Mars vehicle itself, or SSF).
- HLLV launched on 90 day centers = time constraint for on-orbit assembly operations.
- All Mars vehicles are assumed to be launched February 2016.
- Any localized debris shielding is removed from Mars vehicle prior to departure
  from Earth (micrometeroid shielding is assumed to be needed for the mission duration.

For the NEP and SEP vehicle, the ET based facility was used to build the MEV aerobrake and to assemble the MTV system, MTV habitat, initial truss sections during the first few missions. Later missions used the facility for temporary storage and vehicle structures as

the assembly facilities from where robotics built the remaining sections of the vehicles. Debris shielding is localized to needed areas of the vehicles.

For the NTR vehicle, the dedicated facility was used for assembly. Once the NTR truss is placed at the center of the facility (as an added support to the facility) in the first mission, the vehicle is then built from the MEV down to the engine.

This analysis developed the on-orbit assembly sequence for the MMV concepts. Duration times were assigned to individual operations to quantify the sequence schedule. Results from the on-orbit operation analysis showed that assembly time is dependent upon the number of earth to orbit (ETO) launches. Comparison of the actual on-orbit assembly time for each vehicle concept shows that assembly operations do not require 90 days per mission as required for launch processing of the HLLV (Figure 6-4). During ground processing analysis it was determined that the MTV habitat module has several interfaces to other major subsystems. This implies that the MTV habitat module mission should occur later in the time frame than as developed for this initial on-orbit study. The conclusion of this study is that efficient manifesting is required to reduce the number of launches to orbit and reduce the on-orbit assembly time. Ground processing and on-orbit operations must be worked in conjunction for assembly of a vehicle of this stature to be feasible.

Days per Mission

Vehicle	#1	#2	#3	#4	#5	#6	#7	#8	#9	#10	#11	Total
NTR	7.0	8.5	19.6	18.5	5.25	5.25	9.5	10.5	16.5			101.0
NEP	8.0	10.0	19.6	60.0	13.0	49.0	78.0	16.0				193.0
SEP	8.0	10.0	19.6	22.0	25.5	39.0	25.5					150.0
Cryo/Aerobrake	12.0	12.0	12.0	12.0	12.0	12.0	12.0	12.0	12.0	12.0	8.0	128.0

ACTUAL ON-ORBIT ASSEMBLY TIME COMPARISON
FIGURE 6-4

6.1.3 Special On - Orbit Equipment

The requirements for Special On-Orbit Support Equipment evolved from a study which defined robotics equipment for assembly (Reference 36). As a ground rule, for that study on-orbit assembly is to be completed robotically with crew presence for contingency only. The current study progressed further by defining specific robotic tasks required to complete specific operations. Development of an advanced robotic systems, such as the Mobile Servicing Center is crucial to the success of the on-orbit assembly operations.

Robotics for on-orbit assembly are a major component in the overall MMV system assembly. Lessons learned from the assembly of Space Station and the assembly of the node will directly apply to construction and assembly of the Mars mission vehicle. Other equipment required for assembly include storage fixtures and smart assembly jigs. The following is a list of on-orbit operations that the different robotic equipment are required to perform.

Platform remote manipulating system (PRMS)

1. Receives assembly components from the CTV
2. Manuevers assembly component about the assembly facility
3. Transfers assembly components to other robotic systems
4. Performs coarse alignment operations during assembly
5. Monitors assembly operations

Remote aerobrake manipulating system (RAMS)

1. Receives assembly component from PRMS
2. Manuevers assembly components about assembly area
3. Performs precision alignment operations during assembly
4. Performs precision assembly operations (mechanical fastening)
5. Monitors assembly operations
6. Capable of performing repair tasks during assembly and mission phases

Platform anchor system (PAS)

1. Receives assembly components from PRMS
2. Mechanically attaches assembly components to assembly node structure

3. Manuevers assemblies about assembly facility structure
4. Transfers electrical power from assembly facility to assembly component

Assembly support fixtures (ASF)

1. Receives assembly component from PRMS
2. Fixtures assembly components to assembly facility structure

This study briefly looked at the transportation and handling of component from the launch vehicle to the assembly node. The conclusion was that a cargo transfer vehicle (CTV), a derivative of the orbital maneuvering vehicle (OMV) is required to remove the payload from the launch vehicle and hand-off to the assembly node robotics. The major modification to the OMV is to increase the payload capacity to accommodate the mass and size of the systems of the Mars mission vehicle.

For the advanced vehicles (NTR,NEP, and SEP) special on-orbit equipment comprise the following.

- Robotic welding and certification equipment
- Alkali metal heating capability for NEP
- Remote truss manipulator system (RTMS) for NEP and SEP
    - Performs assembly tasks away from RAMS and PRMS
    - Robotic system carried with the vehicle during the mission
    - Capable of both coarse and fine alignment operations

The facilities/equipment for ground and on-orbit operations are summarized in Figure 6-5.

Existing industrial robotic techniques can be applied to some of the minor assembly tasks, such as mechanical fastening and placing of components. Development of hardware capable of handling the masses of a Mars vehicle and on-orbit assembly will be required. For the advanced vehicle concepts that require operations such as welding carbon - carbon tubes, further definition of design and assembly operations are required to determine the feasibility of robotic assembly.

Facilities/Equipment	NTR	NEP	SEP
Ground			
• Reactor/engine mating and processing facility	X	X	
• Nuclear fuel loading facility	X	X	
• Contaminated materials storage and disposal facility	X	X	
• Solar array/radiator packing and storage facility	X	X	X
• Alkali metals materials and transferring facility		X	
• Radiation/hazardous materials contamination treatment facility	X	X	
• Robotics to handle radioactive fuels and hazardous chemicals/materials and components	X	X	
• Vehicle truss processing and packaging facility	X	X	X
On-Orbit			
• On-orbit robotic welding and certification equipment	X	X	
• On-orbit alkali metal heating capability		X	
• On-orbit robotic repair/maintenance equipment	X	X	X

FACILITIES AND EQUIPMENT FOR THE ADVANCED VEHICLE CONCEPTS
FIGURE 6-5

6.1.4 Node Location Study

During the course of this program, we participated in a trade study that evaluated possible locations for on-orbit assembly and refurbishment operations. This study focused on "off-station" nodes only and examined all aspects of the node location including the support infrastructure. The primary requirements of an assembly node are to provide accessibility and support services to mission vehicles. These support services include debris protection, crew stationing and assembly support.

The four orbits chosen for the node location study were low earth orbit (LEO), geosynchronous earth orbit (GEO), low lunar orbit (LLO), and Lagrangian point - 2 Orbit (L2). For this study, two cases (the second) were analyzed for GEO. The first was a direct launch from orbit to Mars and the second was a powered earth gravity assist (PEGA) type launch from orbit to Mars. A data summary for each orbit is shown in Figure 6-6. This data summary was developed to evaluate each orbit with respect to the orbital

mechanics and other characteristics relating to the MMV mission scenario. The lunar transfer vehicle was used for transportation of vehicle components from the HLLV, once stable orbit at LEO is obtained, to the upper altitudes.

Node / Data to Mars	LEO MASE REF.	GEO Case 1	GEO* Case 3	LLO*	L2*
Δ V to Mars (m/sec)	4281	5402	2423	1995	1315
TMI S mass	470.3 t	751.5 t	191.8 t	147.7 t	87.9 t
Debris Env.(total time)		0	< 1 hr	< 1 hr	< 1 hr
Radiation Env. total time in Van Allen belts	~ 2hrs	0	< 2 hrs	< 2 hrs	< 2 hrs
Radiation Environment	Trapped, SAA	Trapped, GCR, SPE	Trapped, GCR, SPE	GCR, SPE	GCR, SPE
Crew to Node ΔV /t	SSF*** ΔV=10-100m/sec hrs-days	SSF - 6 hrs ΔV= 4200m/sec	SSF - 6 hrs ΔV=4200m/sec	SSF- 3-13days ΔV= 4000 m/s Moon- 2hr ΔV= 2100m/s	SSF -8-18 days ΔV= 3374m/s Moon -3 days ΔV= 2900m/s
Launch window timing	e.s o.** ~ 10 min orbit align ~ 5d opps/day ~15-16 rec = 30-60 days	e.s.o.~ 1/day orbit align ~ planetary position opp. ops/day = continous		e.s.o. @ 10 min alignment =12hr rec.= 27 days retrograde opp.	alignment =12hr rec.= 27 days retrograde opp.
Logistics wndow timing	anytime	5 hr transit, 2 opportunities/day		every 10 days	transit from Moon = 3 day

* denotes PEGA (powered Earth gravity assist)
** engine start opportunity
*** co- orbiting with SSF costs ΔV to maintain poisition relative to SSF(not continous thrust)

ORBITAL DATA SUMMARY
FIGURE 6-6

Upper altitude nodes require a significant increase in earth to orbit (ETO) and lunar transfer vehicle (LTV) flights. A LEO node requires debris shielding, however the penalty is much less than the increased flights for upper altitudes. Further definition of the mission architecture is required prior to final selection of the assembly node location.

6.2 Ground Support

Facility requirements and ground processing operations have been developed for the different vehicle concepts. Ground processing operations are dependent on on-orbit

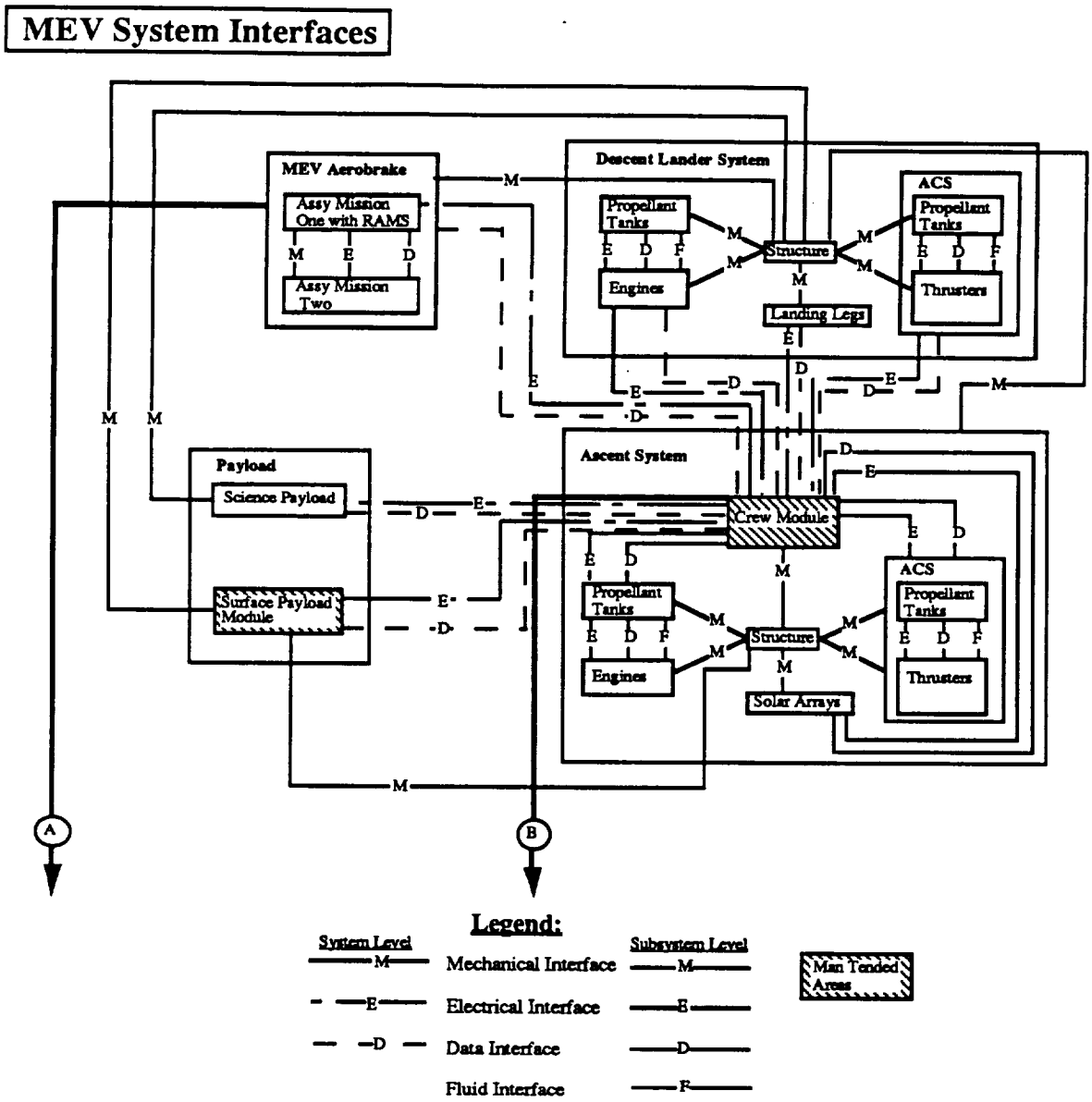
assembly sequence. The following sections discuss the facility requirements and ground operations.

6.2.1 Ground Operations

This study developed basic ground rules and assumptions that apply to all Mars mission vehicle concepts (Reference 7). The approach taken for this analysis was "Sequential Interface Verification", that is the process of verifying the interfaces of the Mars mission vehicle elements without complete assembly. This analysis assumed that tests performed and certified by a contractor would not be repeated and that interfaces would be verified by flight hardware or by match mate devices / prototypes when necessary. A general test philosophy was developed for Mars vehicle assembly and processing.

Ground processing functional flows were developed to analyze the integration of interfaces between systems and subsystems of each of the vehicles. An example of the primary system interfaces considered during this analysis for interface and verification is shown in Figure 6-7. After review of the on-orbit assembly sequence, ground processing sequencing analysis was started, which determined that the initial on-orbit sequence needed to be modified. For instance, the MTV habitat module, for the cryo/aerobrake vehicle, needed to be launched as late in the sequence as possible due to the numerous interfaces that it has with the other systems of the Vehicle. This places a requirement on the assembly node to provide crew stationing. The alternative is to begin the ground processing tasks earlier and launch as planned for in the orginal on-orbit sequence. Larger storage volumes would be required and systems, as well as subsystems, would be stored for a greater period of time. A typical ground processing schedule derived during this task is illustrated in Figure 6-8.

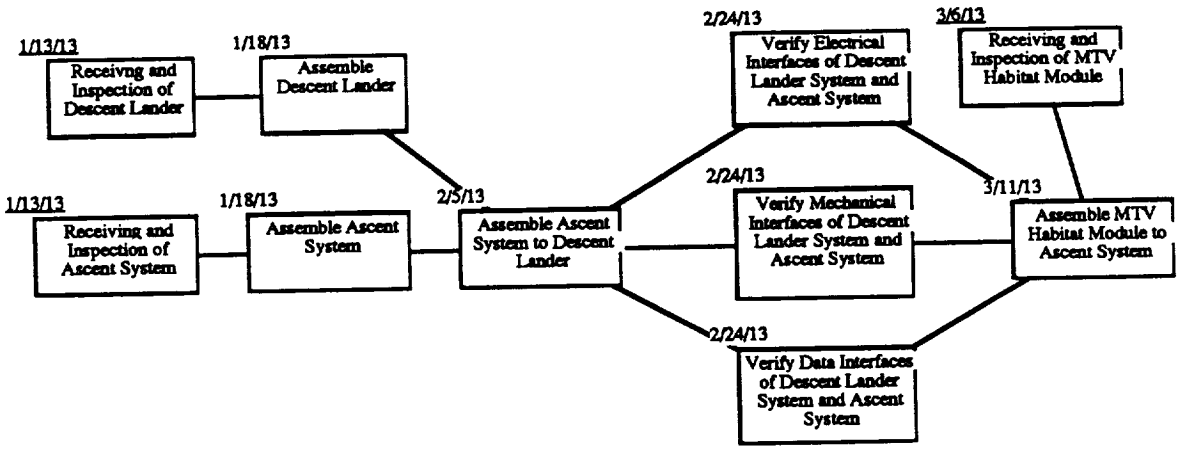
This study defined a system as a group of components and supporting structure that is integrated by a contractor and delivered as a unit to the processing facility. Systems interfaces are defined as those which transmit data, power, or fluids across the system's boundaries and mechanically secure one system to another. Subsystems interfaces are defined as those which are internal to a system and are verified by the manufacturer prior to system integration. These definitions were used in the development of the ground operations analysis.



SYSTEM INTERFACES
FIGURE 6-7

Durations and time scales used in the ground processing analysis were developed as a baseline for future analyses. The analyses determined the basic process flows and established the sequential interface verification sequence. Further definition of the vehicle concepts at the component level will allow more detailed analysis of ground processing. This study has established that on-orbit assembly sequence and ground processing sequence are dependent upon another.

16 Hour Work Days with 8 Hours Schedule time



GROUND PROCESSING SCHEDULE (TYPICAL)
FIGURE 6-8

6.2.2 Ground Facilities

A guideline used for this analysis stated that flight hardware or match mate type devices would be used for test and verification of all interfaces during ground processing. For structures, such as the aerobrake, requirements are driven by the physical size, shape, volume and method of transporting and handling during assembly. Assembly, ground support equipment (GSE) and safety access area requirements were derived from standard KSC operations and Space Station ground processing requirements.

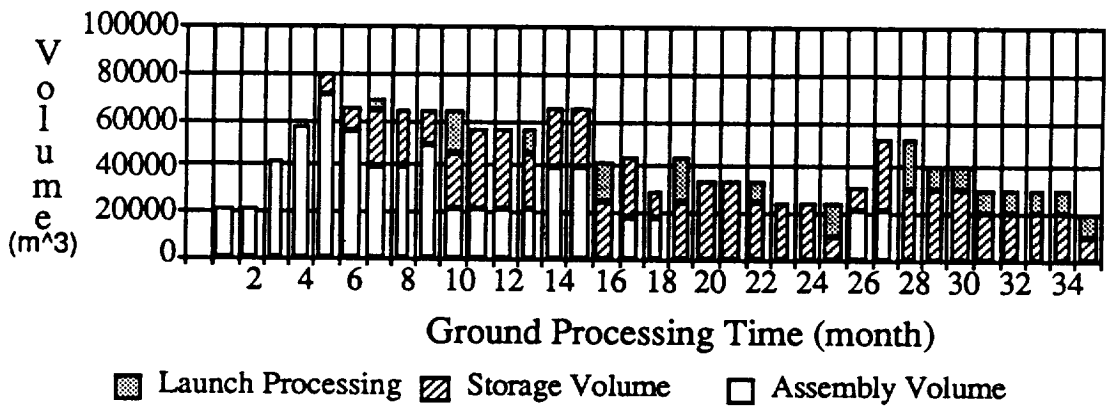
Generic ground rules developed for the cryo/aerobrake ground processing analysis are shown below.

- Processing facility ground rules:
  - Utilize standard services: cranes, power, communications, clean rooms.
  - Make unique hardware portable: special test equipment, work stations.

- Provide large volume work space that can be readily adapted to system block changes, multiple systems in flow.

- Provide for hazardous system processing.
- Minimize transportation and handling between facilities.
- Minimize stacking of subsystems: use components for interface test.

A study was completed to determine the required assembly volume for the cryo/aerobrake ground processing, Figure 6-9. Storage volume is required throughout the ground processing because each system is an integral part of the over all vehicle. From the data illustrated in Figure 6-7, the interface diagrams show that each system is dependent on other systems, therefore, large volume facilities are required to assemble, test, verify, and store the flight hardware and airborne support equipment. Efficient utilization of ground facilities is required to meet the delivery schedule for on-orbit assembly.



CRYO/AEROBRAKE GROUND FACILITIES VOLUME
FIGURE 6-9

The special ground facilities and equipment required to process components of the NTR,NEP, and SEP vehicles prior to launch were provided in Figure 6-5. Detailed facility and volume requirements are yet to be derived for these vehicles.

6.3 Summary Comments

This study began the analysis of the on-orbit and ground operations for the Mars mission vehicle concepts. The results showed that on-orbit assembly time is dependent on the facility available and on the launch vehicle capabilities. Also, ground processing was shown to be dependent on the vehicle design and the on-orbit assembly sequence. Through further definition of the vehicles the operation analysis can determine specfic on-orbit assembly times and facilities required, and the required ground processing facilities and operations. This type of analysis can only be finalized after the vehicle design is completed.

7. TECHNOLOGY

7.1 Introduction

Technology task efforts during the period of the study were concentrated on both the identification and prioritization of the technology requirements relating to, and the identification of high leverage technologies for the major identified Lunar and Mars program areas. An important part of this identification and prioritization process was our participation in many of the important technology working groups held at the various NASA centers. Our involvement in these various working groups (e.g., NEP, NTP workshops, SEI reviews, etc.) benefitted the technology task significantly, particularly in the areas of technology need and projected availability. The other tasks derived benefits of the technical design data, projected performance levels, proposed development schedules, and required funding levels derived from these various working groups. The working groups also presented the opportunity for us to communicate the current overall system level needs of the SEI to those individuals working in the specific technology areas. These efforts will continue throughout the preliminary SEI program phases, in order to target technology areas needing development to meet future space needs, as well as track the progress of past and current development efforts, to ensure that the results are and utilized to the fullest extent possible in the SEI program.

The activities undertaken to meet these requirements were divided into five main subtask areas. These areas include both completed and continuing technology assessment efforts, requirements derivations for the reference vehicles, technology performance level support for various other tasks, technology and advanced development schedule derivation for the major technologies, and a prioritization process centering around a return on investment (ROI) analysis based on life cycle cost models (LCCM) developed for each reference scenario. The current task was tied closely to the programmatics task during the ROI analysis work, as well as the scheduling, and overall cost/benefits analysis efforts. The assessment results included readiness versus needs analyses for enabling/enhancing SEI related technologies, assessment of performance level enhancements for evolutionary missions, and a preliminary identified set of high leverage technologies relating to Lunar and Mars missions. The ROI analysis, along with the scheduling and assessment activities, aided in the identification of the high leverage technologies, and in the prioritization of these

technologies across the various architectures. Three SEI program activity levels were developed in the study which encompass seven transportation architectures, comprised of four hardware/technology concepts, and three operational modes. These three activity levels were intended to cover a range of implementations (low, medium, and high). Prioritization results derived from the transportation architectures can be utilized to obtain a relative technology cost to benefits comparison for the major technology areas. The requirements and potential enhancements assessments were carried out during the completion of all of the other engineering tasks, particularly in the area of trades and sensitivities analysis. Technology development schedules were utilized as a necessary criteria for the technology prioritization process, in that the time-frame for the availability of a technology for a given mission architecture can drive the life-cycle costs, and program planning significantly.

7.2 Assessment Activities

The technology assessment consisted of three separate efforts. The 90-Day Study kicked off the technology task work with a brief two-day assessment in support of a MSFC Technology Requirements and Advanced Development activity for Lunar/Mars outpost initiative. Our assessment was supplemented during the 90-Day Study by SRS Technologies, in support of NASA MSFC. The results of this assessment were published by NASA (Reference 37). During the 90-Day Study, assessments were also carried out to identify those technologies considered "critical", and those considered "significantly enhancing" to our reference concepts (cryo/aerobraked, cryogenic all-propulsive, NEP, NTR, and SEP). An example of the assessment results are shown in Figure 7-1. This effort was a preliminary, top level assessment, utilized mainly to pinpoint the major areas of concern for our later state-of-the-art technology assessment.

The second assessment activity was an in-house survey carried out to obtain technology readiness vs. needs information on all of the identified Mars technology areas. The assessment included the SOTA assessment, technology needs levels and dates, and preliminary predicted technology availability levels. Examples of the assessment forms are presented in Figures 7-2a and 7-2b, in the areas of cryogenic storage and fluid management. Similar forms were produced for each of the technology functional categories shown in Figure 7-3. The forms contained our assessments of technology need levels and dates, as well as related technology issues, including both qualitative and quantitative performance level requirements. The assessment results, as a whole, present our overview

of technology needs and projected readiness levels. This information, along with other assessments completed in support of the NASA SEI efforts, can become the basis for a technology "database." This database would be used in conjunction with a systems requirements database, to aid in connecting NASA related technology development efforts more closely to future program needs, in order to avoid data loss, duplication, or technology need disconnects. A portion of this assessment was the assembly of a matrix of technology issues vs. mission architectures, completed for each technology area. A matrix was assembled to provide a means of representing the applicability of the relevant technology issues to the seven identified Lunar/Mars architecture options, which are outlined in the Executive Summary (see Section 1.0). Each column represents a major technology concern for the particular technology area, and the rows include information on their applicability to the Lunar program (L), Mars program (M), both Lunar and Mars (L/M), or Lunar or Mars evolution (LE or ME). An example of the portion of the matrix for radiation exposure technology issues is shown in Figure 7-4.

- **Large scale solid core reactor**
- **Fuel rod material**
  - Fabrication, maximum operating temperature, and materials (graphite matrix, composite, or carbide)
  - Fuel coating effectiveness, compatability, and process development
  - Thermal stress and corrosion resistance
- **High temperature heat exchanger materials development**
- **Lightweight shielding configuration**
- **Low chamber pressure engine development (higher Isp-1050s vs. 925s)**
- **Testing**
  - Ground test containment facility design and construction
  - Full scale in-space tests
- **Safety Issues**
  - Subcriticality issues under full core immersion and compaction
  - Intact reentry issues
  - Shutdown capability & control

PRELIMINARY HIGH LEVEL TECHNOLOGY ISSUES FOR NTR
FIGURE 7-1

BOEING

Functional Category: Propulsion: Propellant handling and storage

Cryogenic Storage

MISSION	Need Level/Date Phase B	Need Level/Date Phase C/D	Accommodation Level/Date
1. Mars Reference	6	7	6/95
2. Lunar Reference	5	6	6/95
3. Evolutionary Architectures	6	7	6/95

SYSTEM REQUIREMENTS DRIVER(s):

High Isp cryogenic propulsion systems for Lunar and Mars space transfer vehicles. Long duration (up to 3 yrs.) low propellant loss systems for space transfer vehicles and cryogenic fluid storage facilities. Low boiloff ground hold and launch system (5 - 20 min.), and Mars surface storage system (30 - 600 days).

System Description and Urgency Assessment:

Consists of a thermodynamic vent system coupled to vapor cooled shields for boiloff reduction, LH2 Para/Ortho conversion for H2 boiloff reduction/elimination and/or O2 reduction/elimination, fluid gaging for health monitoring, passive orbital disconnect struts or low conduction straps/supports for in-space heat leak reduction, and subcooled/slush H2 & O2 for boiloff reduction/elimination during assembly. Weight is critical to objective performance. Mars ascent systems: Double walled tanks to maximize MLI effectiveness, passive and active cooling systems to minimize Mars surface boiloff. Flexible vacuum jacketing, He/N2 purge systems, or foam/MLI combinations for reduction of ground hold heat leak.

System performance validation is essential for space based OTV and Lunar vehicle operations, and especially critical for the longer duration Mars reference and Lunar/Mars evolutionary mission architectures.

Future Studies Office

Sheet Number: 1 of 1

SYSTEM PERFORMANCE NEEDS

DATE: 3/12/90

Integration Area: Space Transfer

System: Cryogenic Storage

Functional Area: Propulsion: Propellant Handling & Storage

System performance requirements	*Performance Level* Mars Reference	Lunar Reference	Evolutionary Arch.
Passive			
Thermodynamic Vent Systems (TVS)	Single phase vent	Single phase vent	Single phase vent
Vapor cooled shields (VCS)	> 50% reduction in heat leak	> 50% reduction in heat leak	60-70% reduction in heat leak
Support system (thermal disconn.)	90% reduction of support heat leak in low - g. →		
Para/Ortho Conversion	Significant reduction in O2 mission boiloff →		
Subcooled/slush H2 and O2	Reduction/elimination of LEO boiloff losses →		
Overall system boiloff rate	<0.25% / month	< 2% / month	< 0.1% / month
Liquid acquisition for restart	100% liquid	100% liquid	100% liquid
Active			
Refrigeration	> 20 W capacity @ 20K	Not Needed	> 40W capacity @ 20K
Efficiency	> 1%	NA	2 - 10%
Overall system boiloff rate	0%	NA	0%
Tank design			
Advanced MLI design	$<1 \times 10^{5}$ °K Thermal cond.		
Advanced MLI installation	<1.5 MLI degradation factor	<1.5 MLI degradation factor	<1.5 MLI degradation factor
Al-Li Composite tank material	<10% overall thermal protection system fraction →		

Additional Notes

Long term cryogenic storage system capable of either greatly reducing LOX/LH2 boiloff or eliminating it altogether (refrigeration, vapor cooled shields, and Para/Ortho conversion for LOX & LH2). Performance should be reliable enough. These technologies are categorized by system type and prioritized by relative importance. Need for a program in addition to National Aerospace Plane (NASP).

EXAMPLE INFORMATION SHEET FOR INTERNAL ASSESSMENT ACTIVITY

FIGURE 7-2

BOEING

Functional Category: Propulsion; Handling & Storage

Cryogenic Fluid Management

MISSION	Need Level/Date Phase B	Need Level/Date Phase C/D	Accommodation Level/Date
1. Mars Reference	6	7	6/95
2. Lunar Reference	5	6	6/95
3. Evolutionary Architectures	6	7	6/95

SYSTEM REQUIREMENTS DRIVER(s):

Health Monitoring system which continuously checks propellant and tank integrity. Mission architectures which require fluid transfer in low - g. Propellant boiloff during long vehicle assembly time in LEO.

System Description and Urgency Assessment:

System consists of equipment and procedures for low - g fluid management. Low - g fluid transfer needed primarily for filling and topping off of cryo tanks. Fluid acquisition needed for tank emptying, venting, or engine start/restart. Continuous quantity gaging is needed for health monitoring, and thermal mixing for ullage pressure control. Detection of leaks and assessment of tank integrity following meteroid impacts (Ascent systems : Fluid transfer for filling/topping off of cryo tanks before descent.)

Acquisition, pressure control, and gaging system performance validation is necessary for space based OTV and Lunar operations, as well as the long term Mars missions, where high reliability is critical.

Future Studies Office

Sheet Number: 1 of 1 | SYSTEM PERFORMANCE NEEDS | DATE: 3/26/90

Integration Area: Space Transfer

Functional Area: Propulsion: Propellant Handling & Storage

System: Cryogenic Fluid Management

System performance requirements	*Performance Level*		
	Mars Reference	Lunar Reference	Evolutionary Arch.
Fluid Transfer	Top-off before Earth dep.	Top-off before Earth dep.	Zero vent transfer
Acquisition - start	μ - g Acquisition for start (100% fluid or 2 phase restart cap.)		μ - g Acquisition for start
- transfer	>90% tank fill prior to Earth dep.		>95% low-g or Art.-g tank emptying
Fluid gaging	Pressure, Temp. & Quantity (<1-2% error) →		
Thermal mixing	Autonomous, reliable, tank ullage pressure control (no - "hot - spots")		

Additional Notes

Advanced systems for safe cryogenic fluid transfer, accurate gaging , and reliable in-flight operation.

Need for a program in addition to Air Force National Aerospace Plane (NASP), to address unique needs of exploration missions.
COLD -SAT or alternative is critical technology development tool. May need additional orbital testing (possibly during flight test phase).

EXAMPLE INFORMATION SHEET FOR INTERNAL ASSESSMENT ACTIVITY (CONTINUED)

FIGURE 7-2

Propulsion: Propellant Handling & Storage

- Cryogenic Storage
  - Low heat leak cryogenic fluid TPS, max. performance storage system
  - Cryogenic tank materials
- Cryogenic Fluid Management
  - Propellant quantity gaging, tank integrity monitor
  - Cryo fluid rematable umbilical: seals
- In-Situ Resource Utilization

Propulsion: Engines

- Small Cryogenic Engine Development
  - Isp, gimballing, etc.
  - Throttlability
  - Restart cap.
  - Nozzle extension/retraction
- Large Cryogenic Engine Development
  - Isp, gimballing, etc.
  - Restart cap.
- H_2 - O_2 ACS/RCS

Propulsion: Advanced

- Electric
  - SEP
  - NEP
- Nuclear
  - NTR
  - GCR
- Non - Rocket Surface to Orbit System

Propulsion: Alternate Architectures

- Lunar
- Mars

Assembly & Checkout

- In-space automated rendezvous & docking (AR&D)
  - Sensing & control
- In-space assembly
  - Welding
- Refurbishment for Reuse
  - Thermal control coatings
  - Non-destructive testing techniques
- Flight readiness firing surrogate, automatic checkout

Data Management System

- DMS/On-board System Diagnostics
  - Automated self-check
  - Man - machine interface (workstations)

Communications

- Deep Space Communications
  - High data rate communications or high performance compression

Power

- Space Transfer Power Systems
- Lunar & Planetary Surface Power Systems

Artificial - g System

- Tether system
- De-spin Systems

Aerobraking

- Environments Definition
  - Mars Aeroheating Prediction
  - Earth Aeroheating Prediction
  - Mars atmosphere uncertainty effects on GN&C
- GN & C
  - Sensing & control architectures
  - Real time trajectory generation
  - Fault detection, isolation / control reconfiguration
  - Uncertainty (risk) analysis
- Structural Design
  - Passive TPS
  - Advanced structure
  - Smart structure
  - MTV design
  - MEV design
  - LTV design
- TT & C

EVA

- Low mass, high pressure EVA suit/PLSS for transit and surface use
  - CELLS interface
  - serviceability
  - mobility unit
- Airlock

Crew Systems

- Crew Habitats: structure
  - Long duration
  - Fabrication technologies
  - Short duration
- Thermal Control Systems
  - Thermal control coatings
  - Space radiators
- Optimized Radiation Protection
  - Requirements & materials
  - System to inert & can waste
- Meteor / Debris Protection

On - Board Maintenance

- Hull Repair
- Surface Contamination Control
- Automation & Robotics Systems
  - Voice activated command & control

Mission Analysis

- Flight Mechanics
  - Low thrust maneuvering (code validation)
  - High thrust maneuvering (code validation)

TECHNOLOGY FUNCTIONAL CATEGORIES FOR THE IN-HOUSE ASSESSMENT ACTIVITIES

FIGURE 7-3

Technology Issues vs. Mission Architecture Matrix **Technology Area:** Radiation Protection **Date:** 03/26/90

	Technology Issues							
	Effective rad. protection of electronics	Microbiotic effects	Effects to living systems from HZE particles	Advanced shielding methods	Determination of nuclear cross sections	Nuclear reactor shielding	Orbiting nuclear reactor disposal	
1. Cryogenic/Aerobraking	M/L	M/L	M/L	M/L	M/L	—	—	
2. NEP	M/L	M/L	M/L	M/L	M/L	M/L	M/L	
3. SEP	M/L	M/L	M/L	M/L	M/L	M/L	M/L	
4. NTR (Nuclear Rocket)	M/L	M/L	M/L	M/L	M/L	—	—	
5. L2 - Based Cryogenic/ Aerobraking	M/L	M/L	M/L	M/L	M/L	L	L	
6. Direct Cryogenic/ Aerobraking	M/L	M/L	M/L	M/L	M/L	M/L	M/L	
7. Cycler Orbit	M/L	M/L	M/L	M/L	M/L	L	—	

Technology Issues vs. Mission Architecture Matrix **Technology Area:** Radiation Protection **Date:** 03/26/90

	Technology Issues							
	Reevaluation of risk assessments	SPE & dosimetry warning systems	Advanced shield materials	Reassessment of rad. effects in 0-g	Chemical inhibitors	Development of materials data base	Reduction in uncertainties of environments	Code development
1. Cryogenic/Aerobraking	M/L	M/L	M/L	M	M/L	M/L	M	M/L
2. NEP	M/L	M/L	M/L	M	M/L	M/L	M	M/L
3. SEP	M/L	M/L	M/L	M	M/L	M/L	M	M/L
4. NTR (Nuclear Rocket)	M/L	M/L	M/L	M	M/L	M/L	M	M/L
5. L2 - Based Cryogenic/ Aerobraking	M/L	M/L	M/L	M	M/L	M/L	M	M/L
6. Direct Cryogenic/ Aerobraking	M/L	M/L	M/L	M	M/L	M/L	M	M/L
7. Cycler Orbit	M/L	M/L	M/L	M	M/L	M/L	M	M/L

TECHNOLOGY ASSESSMENT MATRIX EXAMPLE
FIGURE 7-4

The final technology assessment activity was a preliminary assessment of the major technology thrust areas as identified in NASA POP '91 support activity (Reference 38). This assessment centered around the areas of aerobraking, cryogenic propulsion, cryogenic fluid systems, and vehicle avionics and software. This assessment consisted of a SOTA assessment and a benefits vs. cost assessment, consisting of both qualitative analysis and quantitative material (which mainly consisted of the ROI data.) Additonally an expanded set of technology development categories was produced in support of this activity. The major categories were drawn from the original NASA technology "thrust" areas, and expanded where necessary, based on inputs from various NASA representatives, and STV and

STCAEM contractor representatives. The final breakdown of the technology functional categories is shown in Figure 7-5. The POP support work completed as a portion of this initial contract, was the identification of the "top 10" technology areas of concern relating to the Lunar/Mars program, along with backup assessment and justification information for each area. More detailed assessments are yet to be conducted. The ten areas identified during this assessment are as follows:

a. Cryogenic systems (cryogenic fluid storage, management, and health mgt.) - these technologies are significantly enhancing to enabling for all vehicle. options.

b. Cryogenic propulsion - also enabling for all options.

c. Advanced ECLSS - enabling for all long term missions and flight operations.

d. High and low energy aerobraking - significantly enhancing to enabling.

e. Crew modules and crew systems (including radiation protection) - significantly enhancing to enabling for long term missions.

f.. Nuclear thermal propulsion - significantly enhancing for all Mars missions.

g. Vehicle avionics and software (including vehicle power) - high reliability system enabling; common avionics "suite" enhancing.

h. Vehicle assembly and processing (ground and in-space) - significantly enhancing to enabling for large vehicle; includes M/D protection issues.

i. Electric propulsion (solar and nuclear) - more capable, flexible for evolutionary use.

j. Advanced Structures and Materials - enhancing across most mission elements.

1 - Aerobraking
- Thermal Protection System Materials Development
  - Ablative materials
  - Reradiative materials
  - Sealants
  - Space environmental effects
- Structural Design Trades and High Temperature Struct. Materials Development
  - Rigid aerobrakes
  - Flexible aerobrakes
- Adaptive GN & C
  - Real time trajectory generation
  - Equipment technology and advanced development (optical, space based tracker, laser gyro, advanced computer & high altitude GPS receiver)
  - Mars atmosphere uncertainty effects on GN&C
  - GN&C analyses and trades (architecture selection)
- TT & C (Interplanetary)
  - Optical systems
  - Advanced RF systems
- Health Management
- Subsystem Intergration
  - Assembly technology & TPS/structure interfaces
    - latching & "smart" structures
- Aerophysics Codes and Analysis
  - Mars atmosphere characteristics
  - Mars aeroheating prediction
  - Earth aeroheating prediction

2 - Cryogenic Engines / Propulsion
- Main Engine Design and Development – RL-10/ASE
  - Engine configuration/characteristics trades and selection: (man rating; space basing; performance; throttling; margins; operational characteristics and health monitoring)
  - Component development : (combustion chambers, electromechanical valves and actuators, high perf. turbopumps)
  - Modularity
    - integrated system
    - vehicle interfaces
  - Engine advanced development / system integration
- Auxiliary propulsion system
  - Throttling
- Propulsion System
  - System architecture (centralized vs. distributed)
  - Main / Auxiliary system interfaces
  - Component & system life testing / long term environmental effects
- Health Management
- Advanced Codes and Analysis

3 - Cryogenic Systems (Vehicle)
- Long Term Storage
  - Low heat leak cryogenic TPS/advanced MLI application procedures
    - MLI degradation / long term environmental effects
  - Vapor cooled shields
  - Para-to-ortho Hydrogen conversion
  - Subcooled / slush H2 and O2
  - Refrigeration / reliquification
- Fluid Management
  - Propellant quantity gaging
  - Health management / tank integrity monitor
  - Low-g fluid transfer
    -Cryo fluid rematable umbilical: seals
  - Low-g fluid acquisition
  - Ullage pressure control (tank destratification)
- Health Management / Tank Integrity Monitor

4 - Vehicle Avionics and Software
- Vehicle Avionics
  - Artificial intelligence / expert systems (e.g. neural software)
  - Advanced computer systems / data management
  - Architecture development trades and analyses
  - Overall architectures; redundancy management
  - Sensing & control
  - Uncertainty (risk) analysis
  - Advanced communication systems (high data rate or high perf. compression)
  - Space environmental effects
  - Autonomous landing
- Space Transfer Power Systems
  - Fuel cells
  - Batteries
  - Solar arrays
- Advanced Software
  - Automated code generation
  - Software verification and validation
  - Multi-redundant and real-time operation
  - Advanced analytical code development
- Health Management
  - Health and Status management including high confidence fault detection, isolation & control, and reconfiguration

5 - Vehicle Structures and Propellant Tanks
- Advanced Structural Concept Development and Advanced Development
  - High performance structural materials & processes
  - Vehicle frame and shell structures
  - Mechanisms (e.g. landing gear, deployment systems)
  - Meteoroid/debris shields
  - Large solar array structures (e.g. SEP)
  - Welding technology (in-space assembly)
  - Thermal control systems and coatings
  - Meteoroid / debris protection materials and structure
  - Space environmental effects / surface contamination control
- Propellant Tanks
  - High performance tank materials & processes
    - Al/Li
    - Composites

TECHNOLOGY FUNCTIONAL CATEGORIES

FIGURE 7-5

6 - Crew Modules and Crew Systems
- Crew Modules
  - Crew module pressure vessels and interior structures
    - advanced materials and structural concepts
    - fabrication technology
  - Excursion Vehicle Small Airlock Development
  - Integrated system tests
- Crew Systems
  - Optimized Radiation Protection
    - requirements & materials / analytical studies
    - innovative radiation protection system concepts
  - Advanced Health Maintenance and Onboard Medical Systems
  - Advanced Controls and Displays
    - adv. man-machine interfaces; interfaces to veh. avionics
    - controls and displays
  - Advanced Fire Protection and Control
- Health Management

7 - Environmental Control and Life Support
- Life Support (atmosphere management; water recovery & management; food supply & galley equipment; waste management)
  - Advanced analysis and simulation techniques
  - Advanced component and subsystems technology and advanced development
  - Equipment durability, life testing and redundancy management
  - Integrated system testing
- Manageable 1/3 and 1/6 g Zero Pre-BreatheEVA Suit/PLSS for Transit and Surface Use
  - CELSS interface
  - Serviceability
  - Mobility aids
- Health Management

8 - Vehicle Assembly, Servicing, & Processing
- Ground Assembly, Servicing, and Operations
  - Sim. and test facility technology (e.g. lg. air bearing test facility)
  - Robotics (ground assembly and processing)
  - Planetary ISRU interface operations
  - Vehicle assembly and processing facilities
- In-Space Assembly and Operations
  - In-space automated rendezvous & docking (AR&D)
    - sensing & control
    - docking and berthing methods and devices
  - In-space assembly and checkout, servicing process
    - automation & robotics systems
    - joining, attachment, deployment, and fasteners
    - welding
    - verification techniques
    - artificial intelligence
    - health and status management
- Ground Assembly, Servicing, and Operations (Planetary)

9 - Orbit Launch and Checkout
- Automated Rendezvous &Docking
- Robotics
- Refurbishment for Reuse
  - Thermal control coatings
  - Non-destructive testing techniques
  - Pressure vessel repair
- Health Management
  - Automatic checkout
    - flight readiness firing surrogate,
  - Artificial intelligence / expert systems
  - On-board built-in test and system diagnostics
    - Automated self-check
    - Man - machine interface (workstations)

10 - Vehicle Flight Operations
- Flight Mechanics
  - Trajectory / Orbit Determination and Data Fusion
  - Low thrust trajectory design codes & validation
  - High thrust trajectory design codes & validation)
- Mission Management
  - Artificial intelligence / expert systems
  - Ground, orbital, transit, and landing control
- Health Management

11 - Artificial Gravity
- Concept Development
  - Tether systems
  - Spin/de-spin systems
- Techniques to Counter Space Physiological Effects

12 - Nuclear Propulsion
- Nuclear Thermal Propulsion
  - Reactor design, analyses and trades
  - Fuel development
  - Shielding and control systems
  - High performance turbopumps
  - Pressure vessels and nozzles
  - Test facility design and technology demonstration
  - Integrated engine advanced development / tests
- Nuclear Electric Propulsion
  - Reactor design, analyses and trades
  - Fuel development
  - Shielding and control systems
  - Test facility design and technology demonstration
  - Power conversion system development
  - Power conditioning, control and distribution
  - Thruster technology (ion; MPD; arcjet)
    - performance improvement and demonstration
    - life improvement and demonstration
  - Integrated System Advanced Development / Life Tests
- Health Management

13 - Solar Electric Propulsion
- Power & Propulsion
  - Power system
    - Cell and array manufacturing technology
    - Array installation/deployment
  - Power Conditioning, Control and Distribution
  - Thruster Technology (ion; MPD; arcjet)
    - Performance improvement and demonstration
    - Life improvement and demonstration
- Structure & Assembly
- Integrated System Advanced Development / Tests
- Health Management

TECHNOLOGY FUNCTIONAL CATEGORIES (CONTINUED)

FIGURE 7-5

7.3 Technology Requirements

The issue of technology performance level requirements relating to the Mars reference vehicles is an important area of concern. This task was accomplished during the technology assessment exercises described in the previous section, and by the derivation of a preliminary set of technology performance level requirements for each reference vehicle. These requirements are included in the technology sections of each of the Implementation Plan & Element Description Documents (Reference 7-11). An example set of these technology requirements is shown for the reference cryo/aerobraked (CAB) vehicle in Figure 7-6. These technology requirements were derived, in part, to highlight the main areas of technology development concern for the CAB vehicle, and also to illustrate that technology levels required to perform the reference, or "zero-base" mission, are far beyond the current state-of-the-art in many areas.The requirements were broken down by stage, and then further broken down into the technology functional categories such as propulsion, cryogenic storage/fluid management, avionics, etc. A closely related task was a brief assessment to identify the required level of technology development associated with each vehicle element. These are shown in Figure 7-7. This identification of critical/enhancing technologies is meant to serve as a base for continuing efforts to prioritize NASA sponsored technology development efforts. These technologies were selected based on qualitative and technical analysis only, therefore a final assessment cannot be made in this area without including scheduling and cost concerns.

7.4 Task Interrelationships

Technology performance level analysis provided a base for the vehicle systems concept developments. The trades and sensitivities task required technology performance estimates, both current and projected, in order to provide rationale to perform system and subsystem level trades. Performance impacts to the system had to be assessed in order to arrive at adequate criteria for the various trades. These system impacts, or technology leverages, are very important in determining sensitivities of certain systems to performance level or mission requirements changes. The results of the trades and sensitivities analysis were then fed back into the technology task, along with programmatics inputs, to form the basis for the technology leverage assessment and prioritization (see Section 7.6) A comparison of the major Mars propulsion options was presented, along with a trade network analysis for Mars basing (Section 2). These trade networks were used to

I. TMIS

A. Cryogenic storage system
1. Thermal protection system - MLI over foam. (1" foam; ~ 1" MLI)
2. Tanks launched wet.
3. Thermodynamic vent coupled to a single vapor cooled shield.
4. Topoff before Earth departure.
5. ~ 6 months in LEO before use.
6. Negligible boiloff loss after topoff.

B. Propulsion
1. Isp = 475 s
2. Thrust = 200 klb/engine
3. Advanced space engine.
4. Nozzle area ratio = 400
5. No throttling requirements.
6. Gimbal angle (nominal) = 10°
7. Up to 3 burns for departure maneuver (2 restarts).
8. Engine out capability (crossfeed propellant lines).
9. No specified engine cycle.
10. In-space changeout capability.
11. Off vehicle preflight checks.
12. No retraction / extension required.

C. Structure
1. Material - metal matrix composites, advanced alloys, and organic matrix composites.
2. Meteor/debris protection provided for tanks and plumbing.

D. Avionics
Piggybacked on MTV.

E. Power
1. Level : < 1 kW
2. System: Auxiliary power units on engine pod; piggybacked on MTV for back-up.

F. Assembly
1. Off station assembly.
2. Degree of assembly: Separate tanksets / propulsion modules connected in LEO to form propulsion stage.

II. MTV

A. Cryogenic storage system
1. Thermal protection system - MLI; 100 layers on H_2 & O_2 tanks (2").
2. Tanks launched wet - no transfer other than topoff before Earth departure.
3. Thermodynamic vent coupled to a series of vapor cooled shields on the H_2 tank, and one on the O_2 tank.
4. Topoff in LEO before Earth departure.
5. ~9 months in LEO before Earth departure.
6. Boiloff loss of < 10% before Mars departure.

B. Propulsion
1. Isp = 475 s.
2. Thrust = 30 klb/engine.
3. Nozzle area ratio = 400.
4. No throttling requirements.
5. Gimbal angle (nominal) = 10°
6. M/D shield for plumbing & tanks.
7. 3 burns @ 4 - 6 month intervals - minimal degradation.
8. 2 restart capability.
9. Engine out capability (crossfeed propellant lines).
10. Expander cycle.
11. In-space change out capability.
12. Off vehicle preflight checks.
13. No retraction/extension required.

C. Structure
1. Vehicle
a. Metal matrix composites / advanced alloys / organic matrix composites.
b. Micrometeoroid protection for habitat structure (shell and insulation).

EXAMPLE SET OF TECHNOLOGY LEVEL REQUIREMENTS FOR CAB REFERENCE VEHICLE

FIGURE 7-6

2. Aerobrake
   a. L/D = 0.5
   b. Crossrange: NA
   c. Vhp = 7.07 km/s.^{2}
   d. Max-g loading = 6.
   e. Max. temperature = 4000° F.
   f. Structure material: Carbon Magnesium ribs (σ_{ult} = 200 ksi) bonded to titanium honeycomb shell.
   g. TPS material: Advanced reradiative tiles.
   h. Relative wind angle (reference) = 20°.

D. Avionics
1. Planetary vicinity -
   a. Relative velocity error = 100 m/s.
   b. Relative position error = 25 km.

2. System -
   a. Relative velocity error = 100 m/s.
   b. Relative angle error = 0.5°.

E. Power
1. Level - 15 kW.
2. System: Solar arrays with battery storage (NiCad).
3. Back up system: NA

F. Assembly
1. Off station assembly.
2. Assembly level (complexity): TBD

G. Habitat
1. ECLSS: Space Station Freedom derived system with similar degree of closure; potable H_2O from cabin condensate; CO_2 reduction/regeneration; Hygiene H_2O from urine processing. CELSS to be evaluated.

2. Structure
   a. 2219 - T8 aluminum pressure vessel.
   b. Pressurized to 20 psig on launch for structural integrity.
   c. Insulation & M/D shield external to pressure shell.
   d. No penetrations in end domes.
   e. Radiation storm shelter provided, and configured to utilize equipment & supplies as partial shielding.
   f. External space radiator integral with M/D shield.

3. Cabin repressurizations: 2+ (outbound emergency could use propellant for repress.)
4. Spares: 15% of active equipment - component level.
5. Redundancy: Two complete and separate systems for life critical systems + spares. Component changeout capability.
6. Residence time = 535 days.
7. Science: Transit science as allowed by individual mission.
8. EVA capability: EVA suits provided for all crew; EVA waste fluid recovery for ECLSS.

H. ECCV
1. Apollo size & style.
2. Open ECLSS (LiOH, no H_2O recovery).
3. Residence time: 2 - 3 days.
4. Propulsion: RCS only.

III. MEV

A. Cryogenic storage system
1. Thermal protection system: 100 layers of MLI for H2 and O2 tanks (2").
2. Tanks: double wall tanks with vacuum annulus; low thermal conductivity support system for inner tank.
3. Thermodynamic vent: Simple design for gravity field.
4. Tanks launched dry and filled prior to descent, from MTV tanks, or refrigerated. (no boiloff prior to descent)
5. Stay time from 30 - 600 days on Mars surface.
6. Boiloff level < 20% for surface stay.

EXAMPLE SET OF TECHNOLOGY LEVEL REQUIREMENTS FOR CAB REFERENCE VEHICLE (CONTINUED)

FIGURE 7-6

B. Propulsion
1. Isp = 460 sec.
2. Thrust = 30 klb / engine.
3. Nozzle area ratio = 200.
4. Throttleability = 15:1.
5. Gimbal angle (nominal) = 10°.
6. No restart capability necessary for nominal case.
7. Space storage time between burns : NA.
8. Engine out capability (crossfeed propellant lines).
9. Expander cycle.
10. In-space changeout capability.
11. Off vehicle preflight checks.
12. Retraction / extension capability.

C. Structure
1. Vehicle
   a. metal matrix composites / advanced alloys / organic matrix composites.
   b. Micrometeoroid protection for tanks and plumbing.

2. Aerobrake
   a. L/D = 0.5 to 1.0
   b. Crossrange: 1000 km.
   c. Vhp = 7.07 km/sec.^{2}
   d. Maximum g loading: 6.
   e. Maximum temp: TBD (estimated 3100° F).
   f. Structure material: Carbon Magnesium ribs (σ_{ult} = 200 ksi) bonded to titanium honeycomb shell.
   g. TPS material: Advanced reradiative tiles.
   h. Relative wind angle (reference) = 20°.

D. Avionics
1. Error without beacon = 1 km.
2. Touchdown error = 1 m/s.
3. Obstacle avoidance capability.

E. Power
1. Level: ~ 2.5 kW.
2. System: fuel cells (regenerable).
3. Back-up system: abort to orbit.

F. Assembly
1. Off station assembly.
2. Assembly level (complexity): TBD

G. Habitat
1. ECLSS: open system; stored potable H_2O; LiOH CO_2 adsorption.
2. Structure
   a. Aluminum (2219 - T8) pressure vessel.
   b. Overpressurized on launch for structural integrity.
   c. Insulation and micrometeoroid protection external to pressure vessel.
   d. No penetrations in end domes.
   e. No radiation shelter provided in MEV.
   f. External space radiator integral with micrometeoroid shield.
3. Repressurizations: 2.
4. Spares: 15% of active equipment mass; component level.
5. Redundancy: EVA suits as backup to cabin repressurization.; no system level ECLSS redundancy required due to low complexity open system.
6. Residence time: ~3 days (surface systems support surface stay).
7. Science: none.
8. EVA capability: provided for all crew; transferred from MTV.

EXAMPLE SET OF TECHNOLOGY LEVEL REQUIREMENTS FOR CAB REFERENCE VEHICLE (CONCLUDED)

FIGURE 7-6

COMPONENT	Technology Development Activity			
	Comp valid in lab	Comp 1-g sim tests	System 1-g sim tests	Flight Test
Cargo Carrier				
Boick 1 Ops				
HLLV Booster Stage	●	●	●	
STME	●	●	●	
Adv Space Engine	●	●	●	
ET				
Propulsion Boattail				
Recov. P/A Module	●	●	●	
Cryo All Propulsive Veh				
TMIS tank/core	●	●	●	
MOC tank/core				
NTR Vehicle				
NTR engine	●	●	●	●
Truss	●	●		
Tanks	●	●	●	
NEP Vehicle				
power system	●	●	●	
PMAD equipment	●	●	●	
turboalternator	●	●	●	
Standard Avionics Sta.	●	●	●	
LEO Tanker	●	●	●	●
Lunar Vehicle				
LTV	●	●	●	●
transfer hab	●	●	●	
LEV	●	●	●	●
LEV crew cab	●	●	●	
Aerobrake	●	●	●	●
Mars Vehicle				
MTV	●	●	●	●
MEV	●	●	●	●
MEV crewcab	●	●	●	
Mars transit hab	●	●	●	
MCRV	●	●	●	●
Aerobrake	●	●	●	●
LEO Node	●	●		
CELSS	●	●	●	
Solar/RFC Power Units	●	●	●	
Space/Surface TCS	●	●	●	
SP-100 Reactor PSPS	●	●	●	●
Lunar ISRU Equip	●	●	●	
Mars ISRU Equip	●	●	●	
Rover				
small chassis	●	●	●	
small wh/dr system	●	●	●	
large chassis	●	●	●	
large wh/dr system	●	●	●	
Straddler Chassis	●	●	●	

REQUIRED TECHNOLOGY ADVANCEMENT ACTIVITIES VS. MISSION HARDWARE MATRIX

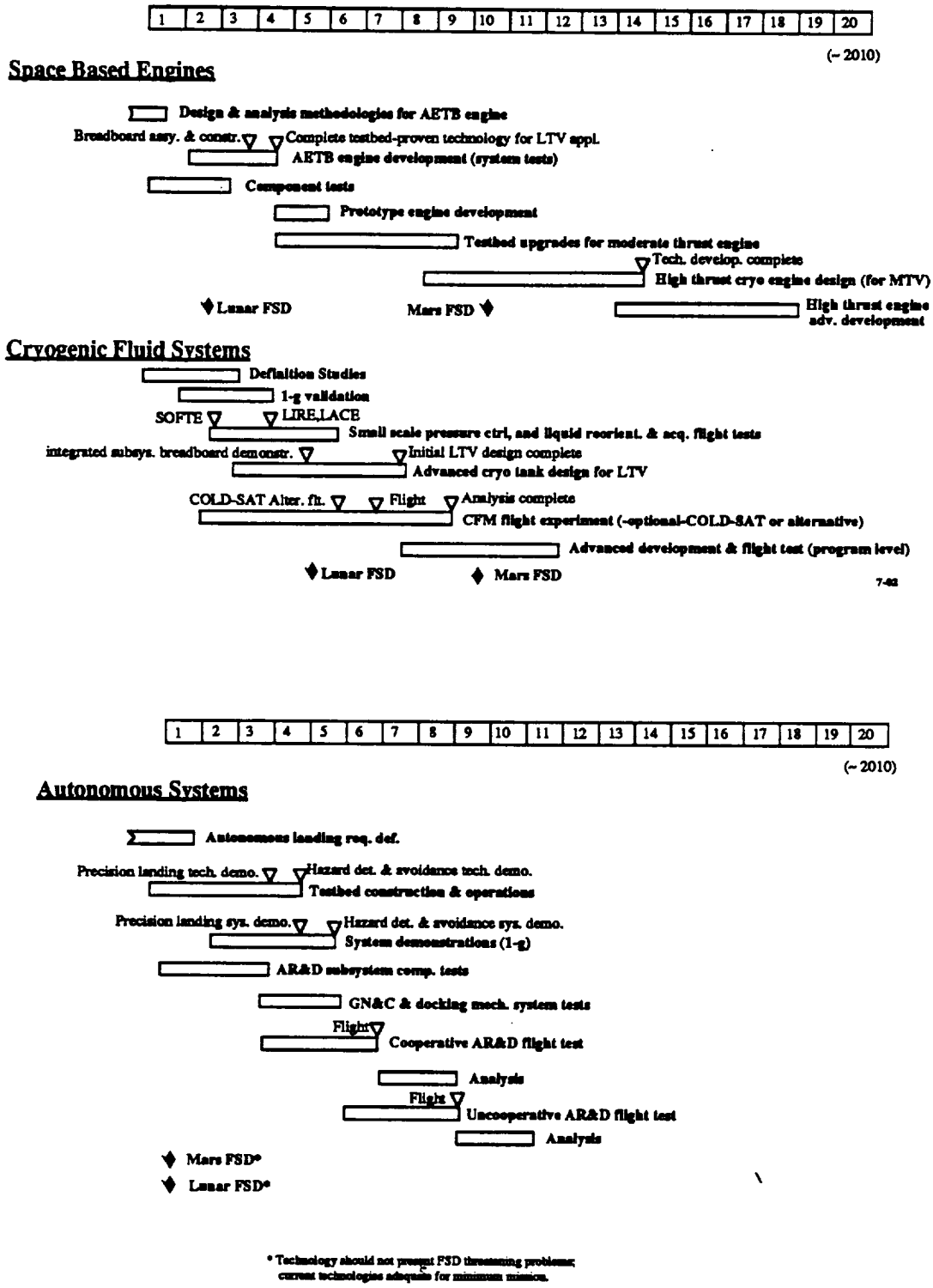
FIGURE 7-7

illustrate interrelationship of program goals and requirements, to mission design and vehicle type. The task required assessment of the capability, flexibility, reusability, and other issues relating to the various mission mode and vehicle options.

7.5 Technology Development Schedules

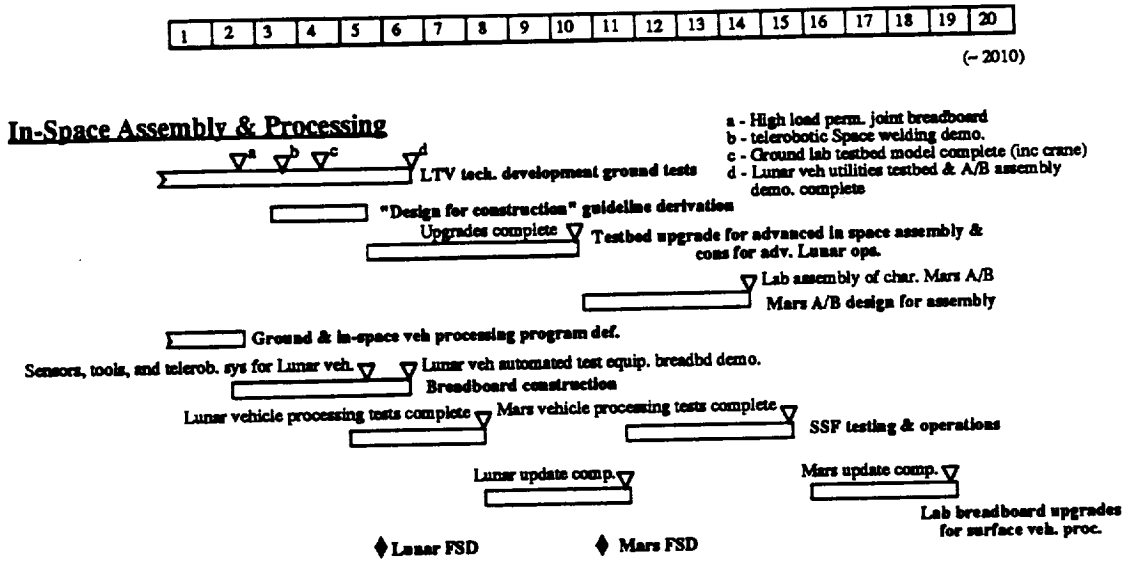
In order to completely assess the technology options for a given application, reasonable development schedules for these options must be derived and weighed against the needs schedules for the program. Any disconnects in need vs. availability can be identified at that point, and suggestions can be given on how these disconnects can be dealt with. Development schedules are a primary element in the prioritization process, along with costs, performance, safety, and reliability.

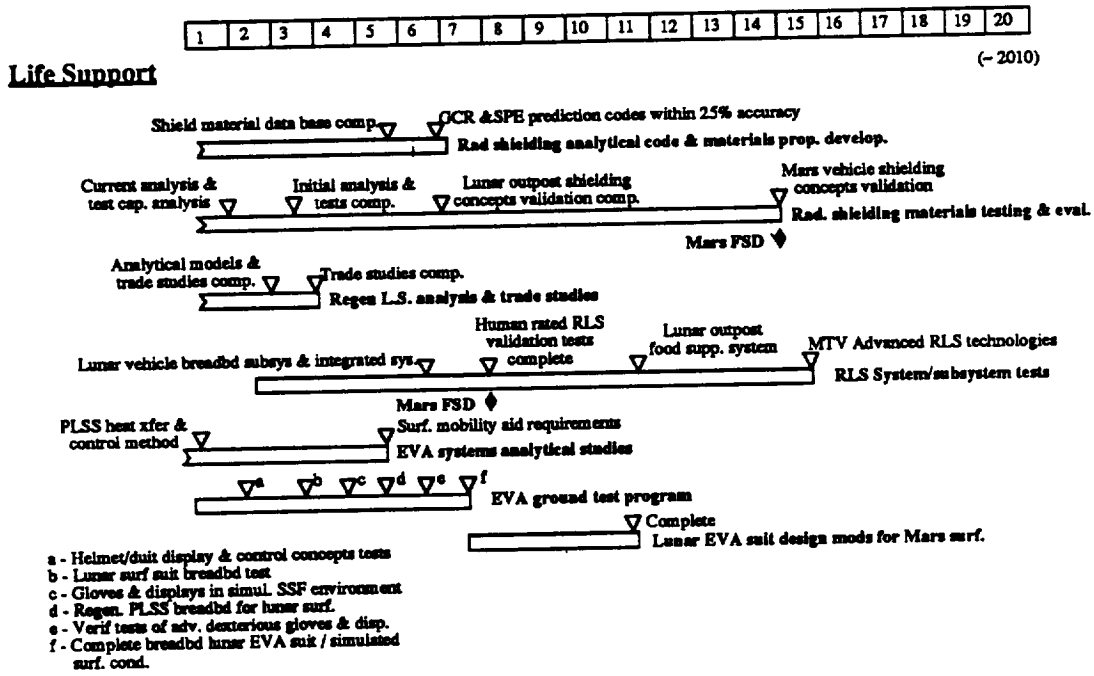
Preliminary development schedules were derived in the areas of aerobraking, cryogenic propulsion and fluid management, autonomous systems, in-space assembly and processing, high rate communications, life support systems, SEP, NEP, and NTP. These schedules were used in the buildup of the overall program architecture schedules, and in the return-on-investment analysis. Technology development costs depend on the required funding profiles for the development timeframe, as well as the scope of each of the identified elements of the program. The schedules, shown Figure 7-8 were derived based on planned NASA development, along with in-house estimates of additional required development elements. These include additional validation tests (ground and space) required to increase system reliabilities to an adequate level for long term Mars system utilization, major facility design and development, and additional component level development detail, where available. None of the schedules are attached to a particular start date, so they can be integrated in other architectures to derive the various required technology development start dates, and any needs/availability disconnects could be identified. All of the schedules assume relatively level funding during the technology and advanced development timeframe, so funding discontinuities could push the program significantly to the right. All of the development schedules include points where a full scale development (FSD) decision can be made. This point is defined as where the technology is at an adequate maturity level to move from the preliminary design and analysis phase, into the hardware development phase of the development program (phase C/D). The FSD phases of a development program may include the construction /refurbishment of



PRELIMINARY TECHNOLOGY DEVELOPMENT SCHEDULES

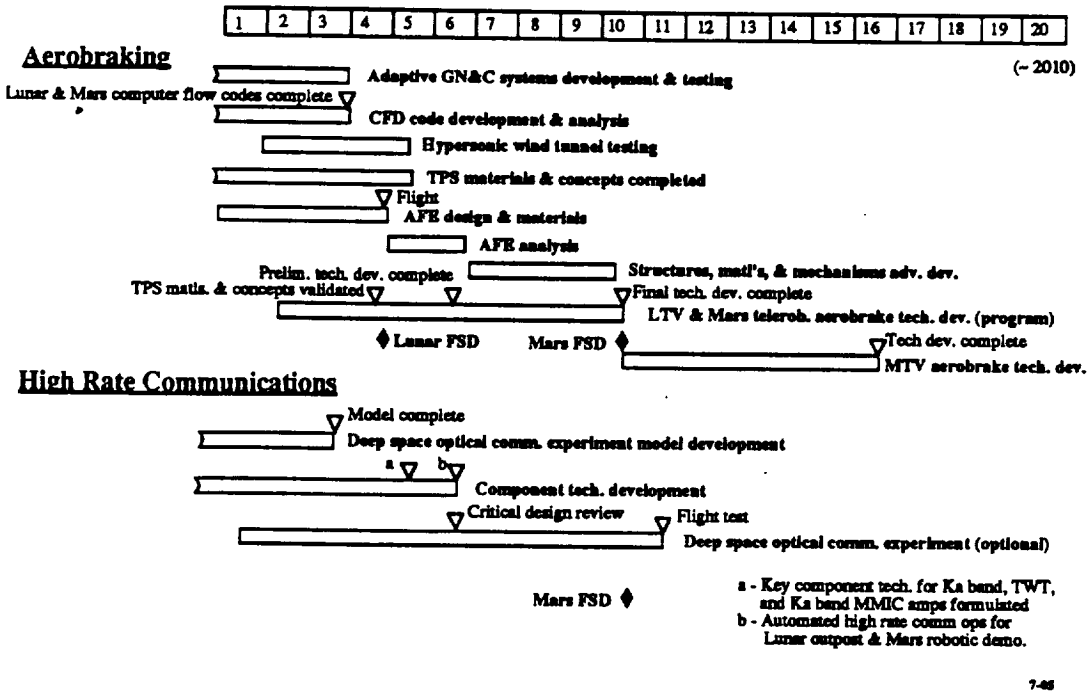
FIGURE 7-8



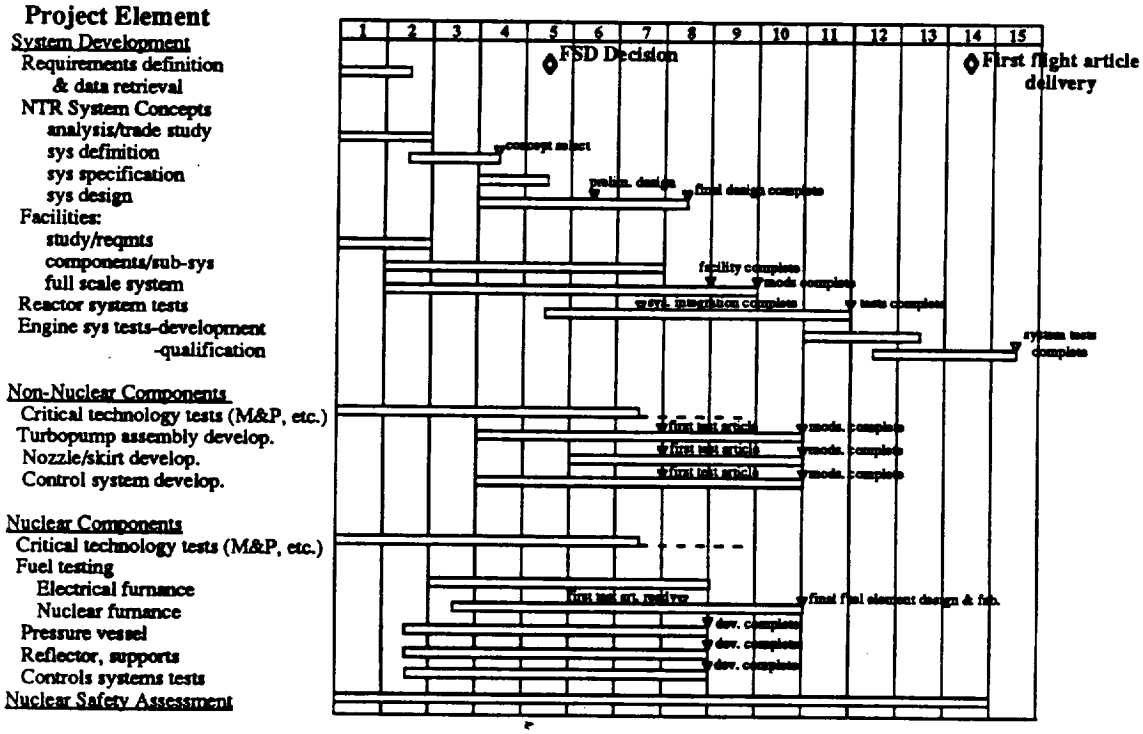


PRELIMINARY TECHNOLOGY DEVELOPMENT SCHEDULES (CONTINUED)

FIGURE 7-8



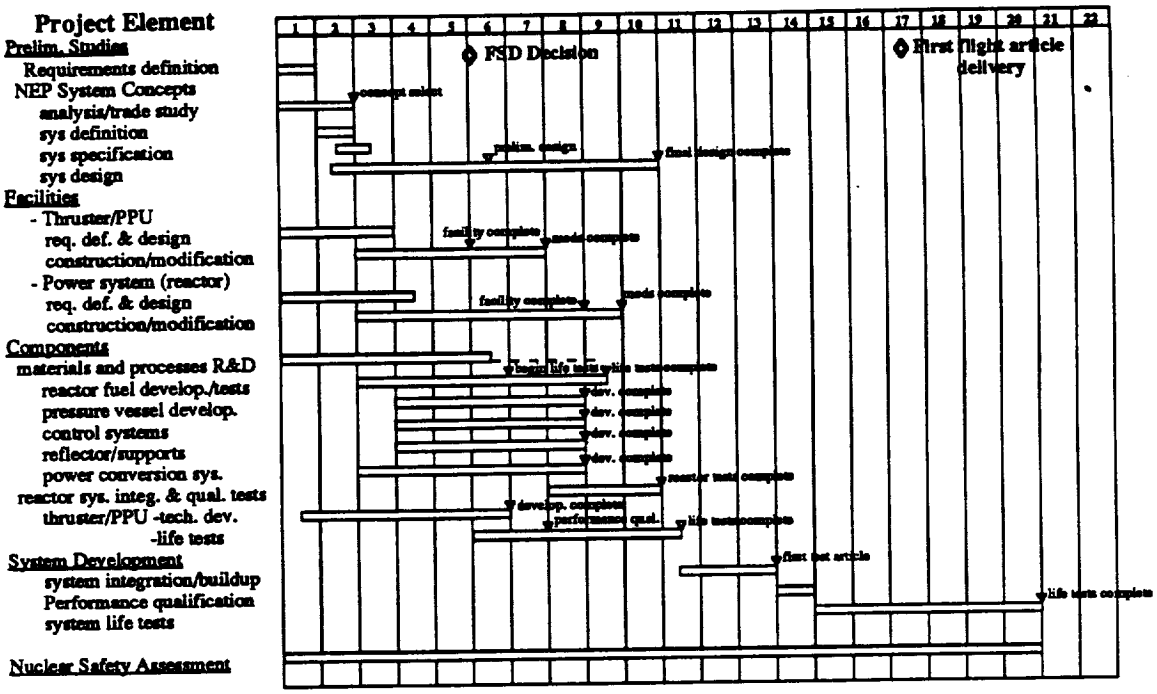
NTR Engine Development Program



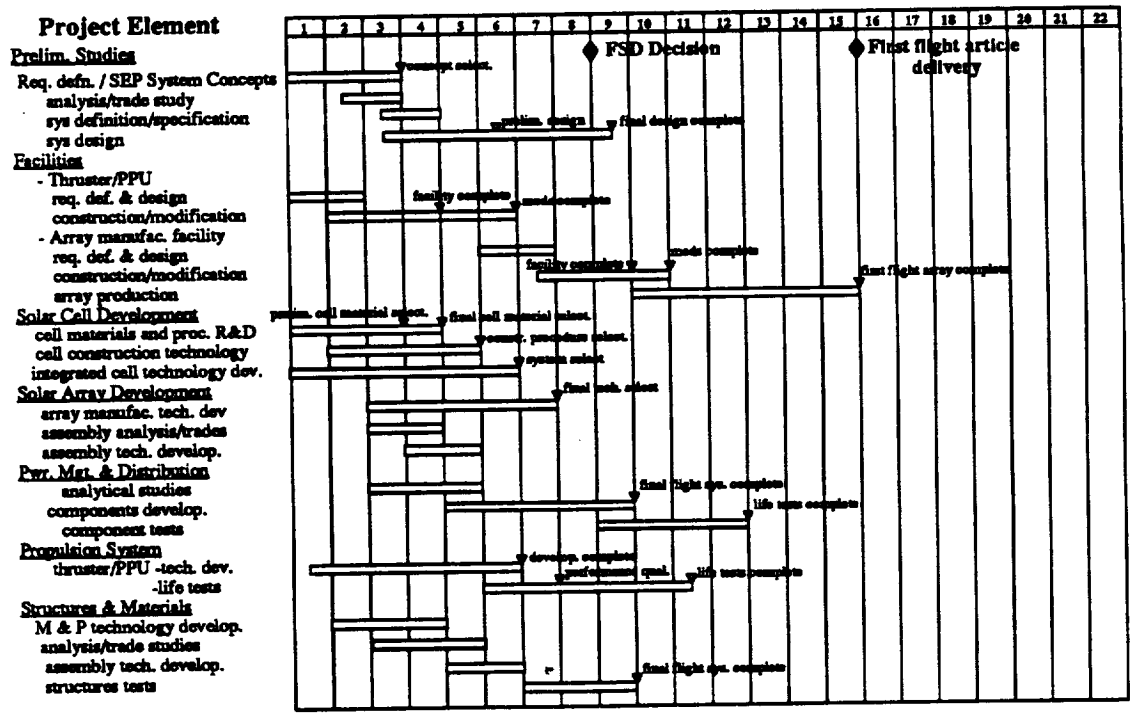
PRELIMINARY TECHNOLOGY DEVELOPMENT SCHEDULES (CONTINUED)

FIGURE 7-8

NEP Technology Development Program



SEP Technology Development Program



PRELIMINARY TECHNOLOGY DEVELOPMENT SCHEDULES (CONCLUDED)

FIGURE 7-8

necessary major test facilities, flight hardware development (first unit/ backup), and subsystem/system level life tests. The FSD decision can be pushed back in a program, however (e.g. autonomous systems), if the development risk is adequately low for a given technology. A low development risk could mean that a technology is either well in hand, or that the development is not as closely tied to NASA funding level throughout the development program (e.g., data management systems or robotics).

A preliminary set of identified test facilities is presented in Figure 7-9. The development of test facilities is included in the technology schedules where an adequate facility does not currently exist. The major facilities noted in this study are containment facilities necessary for the nuclear propulsion (NTP and NEP) systems and life tests. Existing facilities are assumed to be available, with modifications and refurbishment, where needed. The availability of these test facilities may be limited, however, depending on the level of need of other future NASA and DoD programs. A major area of concern for existing test facility utilization is that the physical size of the available test facilities may not be adequate for SEI technology program needs.

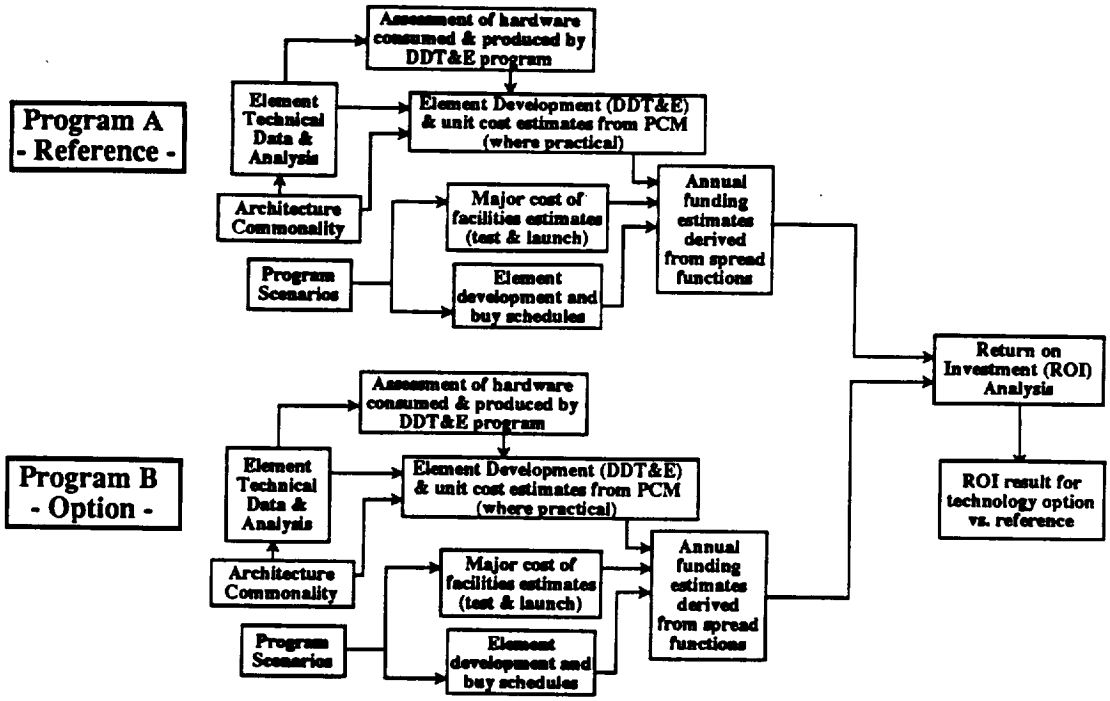
Facility	Applications
1. Thermal Vacuum Chamber	Cryogenic storage, H2-O2 RCS, radiation shielding, aerobrake, cryo fluid transfer, ion thrusters.
2. Hypersonic Wind Tunnel	Aerobrake, ETO vehicles.
3. Engine Test Vacuum Chamber	Cryogenic space engines, ion thrusters.
4. Large Air Bearing Facility	AR&D, on-orbit assembly, avionics.
5. Materials Laboratory	Vehicle structures, radiation shielding, aeroshell
6. Sensor Test Lab	Avionics, AR&D.
7. Engine Component Test Facility	Cryogenic space engines.
8. Nuclear Test Facility - NTR	NEP power system.
9. Nuclear Test Facility - NEP	NTR propulsion system tests/containment.
10. Solar Cell Test/Manufacturing Facility	Vehicle & surface power systems, SEP power system.
11. Assembly & Operations Facility	Aeroshell, vehicle structure.
12. Mock-up/Training Facility	All manned vehicle systems.
13. Acoustic/Vibration Test Facility	All vehicle components.
14. Solar Observatory	Radiation shielding.
15. Computer Sim. Facility/A&R	Avionics, AR&D, ECLSS, on-orbit assembly, flight operations.

PRELIMINARY TECHNOLOGY DEVELOPMENT TEST FACILITY REQUIREMENTS
FIGURE 7-9

7.6 Return on Investment Assessment

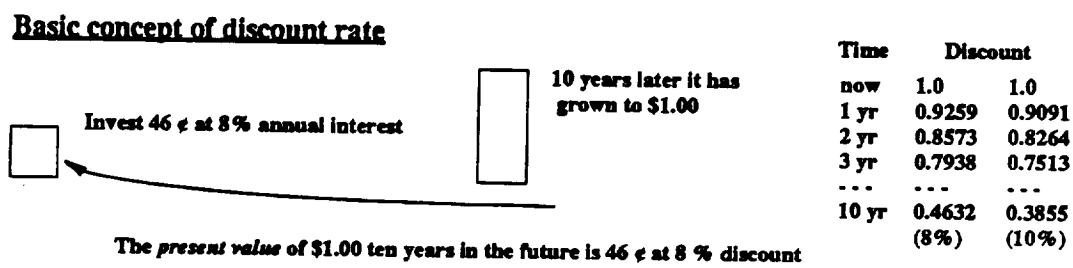
An important criteria in the technology costs vs. benefits assessment is the return-on-investment (ROI) analysis. The ROI analysis consists of three main parts: technology development costs, mission and program performance impacts, and life cycle costs. The technology and advanced development costs by year are entered into the ROI program, and represent program "penalties" for choosing technologies which are beyond state-of-the-art. A preliminary set of technology and advanced development costs were derived for the life cycle cost model (LCCM), in order to carry out the various ROI trades (see Section 1). Sources for these estimates include various NASA technology/AD planning reports, contractor estimates from the working group meetings held during the past year, and in-house estimates based on available knowledge of the planned and past related technology development programs. Where flight tests are required, reasonable cost estimates for a single Shuttle or unmanned flight test are included assuming current launch costs. All cost estimates are based on FY '90 dollars, with launch costs of approximately $1300 / lb for the ALS.

The next major input to the system is the performance and quantifiable impacts to the system if the technology is not utilized. These include additional ETO flights, increased production and operational costs, and decreased reliability (increased spares, flight units). The actual ROI model inputs are derived from two separate life cycle cost streams that are built and executed independently. An illustration of the costing methodology flow is shown in Figure 7-10. The model calculates a resultant cost stream from the difference of the two LCC cost streams. The ROI is then calculated by iterating the discount rate in order to obtain a net present value of zero for this resultant cost stream. This is therefore the return on investment (ROI) for the program employing the technology option, relative to the reference scenario. This return can be negative if the option does not result in a net savings over the reference scenario, or undefined if a crossover point does not exist for the two cost streams. In this case, one of the options is simply cheaper than the other. A graphic representation of the ROI analysis procedure is shown in Figure 7-11.

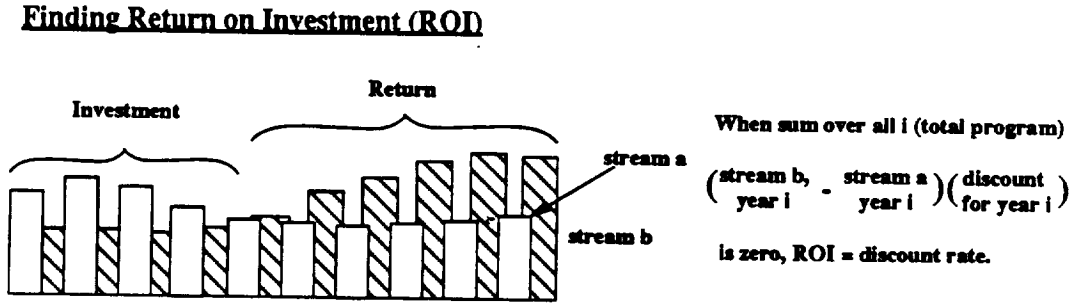


COSTING METHODOLOGY FLOW

FIGURE 7-10



Time	Discount	
now	1.0	1.0
1 yr	0.9259	0.9091
2 yr	0.8573	0.8264
3 yr	0.7938	0.7513
- - -	- - -	- - -
10 yr	0.4632	0.3855
	(8%)	(10%)



RETURN ON INVESTMENT METHODOLOGY

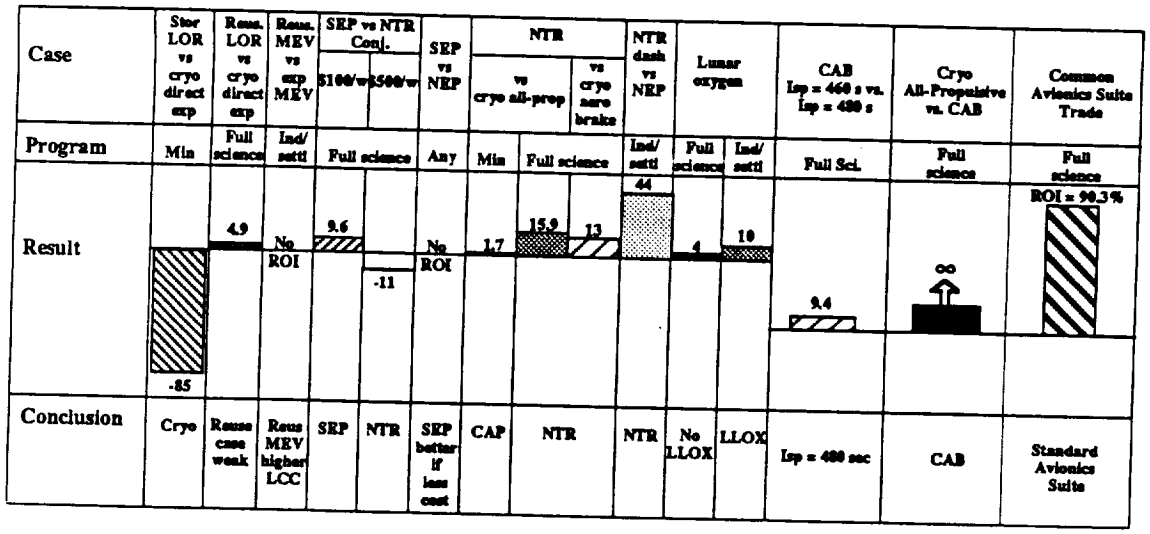
FIGURE 7-11

The return on investment analysis is not intended to provide the actual rate of return to be expected for a single technology development concern, but to give a *relative* return of one technology option verses another. The accuracy of the cost analysis is undoubtedly affected by the uniqueness of the hardware components for the Mars vehicles, due to the absence of a significant development and manufacturing cost database for related past focused development programs. Wherever possible, related past programs, or experience derived from past programs, are utilized in the estimate derivations. Expert opinions and trade or industry assessment results are used for items which could not be adequately costed using the existing database (e.g., aerobrake, advanced propulsion system). The cost estimating procedures used throughout the assessment were kept as constant as possible to increase the confidence level in the relative technology rankings. The spread in years of the reference architectures (34-36), along with the relatively low number of flights (3 for the minimum case to 9 for the settlement / industrialization case), combine to decrease the scale of all but the most beneficial technology options. For this reason, any ROI in excess of 5% is considered significant. Additionally, technologies with small ROI's (<5%) that increase capability for later missions significantly, are considered notable.

An overall summary of the ROI assessment results is shown in Figure 7-12. The results of these trades displayed some interesting trends.

a. The use of cryogenic propulsion systems (vs. storable) resulted in one of the highest ROI's. This was due to the additional design, development, and testing (DDT&E) required for the added vehicle elements needed to fly the required LOR mode for the storable LTV (storable direct vehicle becomes too large), as compared to the cryogenic direct vehicle (i.e. LEV, LEVCM, service module). Additional ETO flights also affected this trade in favor of the cryogenic system, while the additional cryogenic systems technology investment (~ $300M) was not a significant factor.

b. The cryogenic all-propulsive vehicle vs. cryogenic / aerobraked vehicle trade resulted in a very high ROI (program would not converge). The tremendous size of the CAP vehicle for the fast transfer (opposition class) missions results in very large operational costs (including ETO costs), which is much greater than the additional technology and advanced development investment required for the CAB vehicle.



ROI RESULTS SUMMARY
FIGURE 7-12

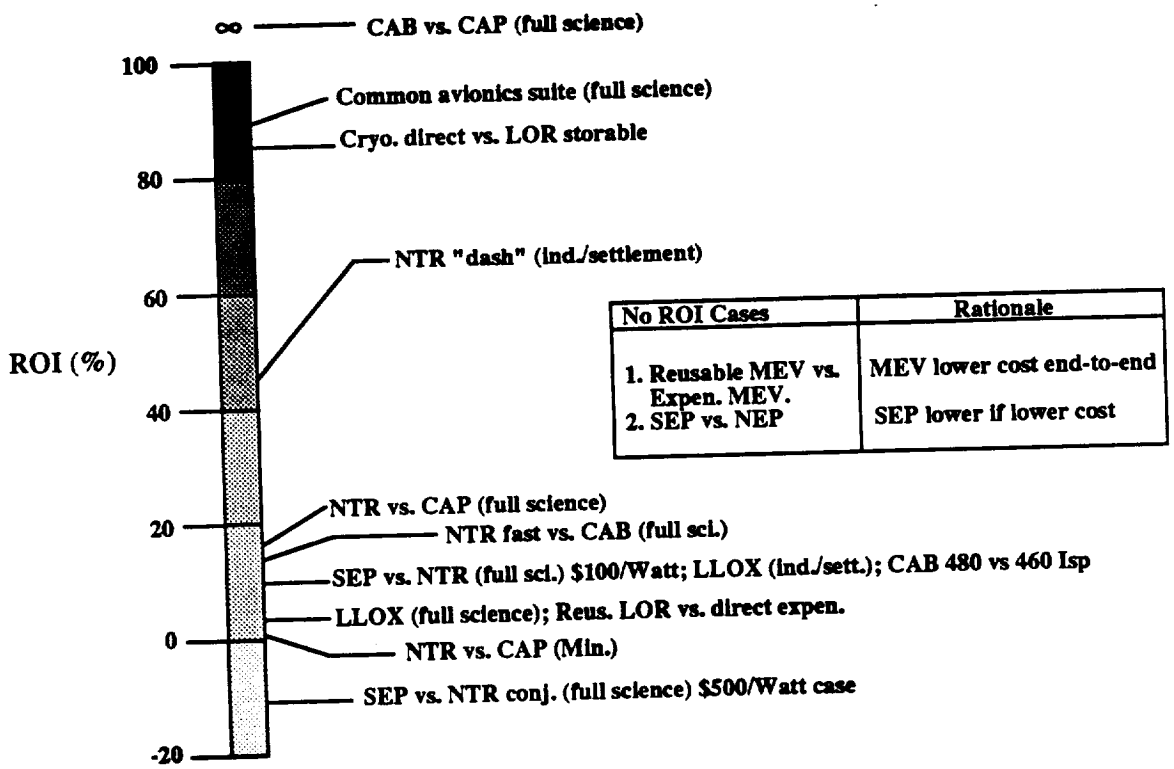
c. A 90.4% ROI resulted from an avionics trade of a common, adaptable system, vs. a unique development effort for each mission element. The high ROI resulted from a significant payoff relatively early in the program (~ $600M), compared to a relatively small short term investment (~ $140M), although the overall LCC for the full science program was not affected greatly.

d. The reusable MEV trade for the industrialization / settlement case resulted in no advantage for the higher technology option (reusable MEV). Additional HLV flight rate assessments may be required.

e. Solar electric propulsion can become a significant technology leverage issue if the manufacturing cost of flight rated arrays can be reduced below approximately $200/watt.

f. The NTR vehicle traded favorably across all of the scenarios, with the exception of the minimum scenario (ROI = 1.7%). It should be noted that the positive ROI is encouraging considering the low number of flights (3), and the significant additional development costs for the NTR propulsion system.

A relative ranking of these ROI results obtained for the various technology options are shown in Figure 7-13.



No ROI Cases	Rationale
1. Reusable MEV vs. Expen. MEV.	MEV lower cost end-to-end
2. SEP vs. NEP	SEP lower if lower cost

COMPARISON OF ROI RESULTS FOR TECHNOLOGY TRADES
FIGURE 7-13

8. REQUIREMENTS AND DERIVED REQUIREMENTS

8.1 Introduction

In order to perform the STCAEM study, it was necessary to start with the top level guidelines and requirements provided by NASA for the Lunar and Mars programs. These conditions are the basis for the derived requirements and the detailed studies conducted in the course of this project.

8.2 NASA Guidelines for Study

The NASA guidelines for space exploration to Mars relating to our study activity are provided below.

- Multi-impulse TEI and TMI is permitted (engine restart).
- Cryogenic propulsion as a reference concept for Earth/Mars departures and Mars descent and cryogenic/aerobrake for Earth and Mars.
- Advanced propulsion technology options include nuclear thermal rocket, nuclear electric propulsion, and solar electric propulsion.
- MTV expendable on "difficult" opposition missions; return to Earth via ECCV.
- TMIS always expendable.
- Three-burn departures acceptable for TMI to ease launch declination window problems.
- Maximum size surface payloads on piloted MEV: 6 m diameter and 13 m length.

8.3 Level I, II, and III Requirements

8.3.1 Mars Initiative Level I Requirements Employed

- The Mars initiative is an evolutionary program leading toward the establishment of a Mars base.
- First manned mission to Mars departs Earth between 2010 and 2020.

- Mission definition and programmatics for the Mars initiative shall be developed through 2025.
- Mission design for piloted mission shall minimize outbound/inbound trip times consistent with reasonable mission performance.
- Launch vehicles shall be sized to reduce the number of launches for Mars mission to six or less (goal).
- Crew size per mission in the four to twelve range.

8.3.2 Mars Initiative Level II Requirements Used in This Study

- Provisions to be made for direct and/or indirect observations of operations.
  - In-space proximity operations require direct operator viewing on manned missions.
- Repairable common system elements to be extensively used.
- Landings shall be made during daylight hours.
  - Determination of proper trajectory to insure daylight approach to the Martian surface.
- Isolation and rapid egress from any habitable element in any emergency.
  - All habitable elements shall have redundant escape paths.
  - Use of pressure hatches for separation between pressurized sections.
- Allow for high leverage technology extensions.
- EVA's not to exceed eight hours/day, two crew minimum EVA, all crew members to have pressure suits.
- Emergency operation of all systems by EVA suited crew member.
  - Crawl hatches, internal and external, sized to allow passage of EVA suited crew member.
- No single point failure in subsystems of safety-critical systems (not applicable to rigid structures).
- Trajectory design for piloted missions shall enable Mars flyby abort to Earth (Note: this requirement is interpreted to mean propulsive flyby for fast-trip modes).
- Orbit selected for piloted missions shall enable early Mars departure for return to Earth.
- Manned flights to deliver crew and 25t payload to surface.

8.3.3 Mars Initiative Level III Requirements Employed in This Study

- All hazardous materials are to be stored outside of pressurized elements.
- Surface radiation protection, if needed, will be provided using Martian regolith/shielding and/or supply storage techniques.
  - Investigate the need for additional equipment that would be necessary to bury habitat modules or place regolith in shielding containers.
  - Determination if the degree of radiation passing through the Martian atmosphere warrants burial as a protective means.
- Provide radiation shielding on MTV to be accessible in 30 minutes.
  - Determination of areal density (g/sq. cm) of radiation protection required.
- Crew transfer module concepts (reusable) to be sized for crew size range of four to twelve.
  - Transit ECLS system will be closed to maximum extent as possible, closed on H_20 and 0_2, MEV ECLS optimized.
- Vehicles sized to meet a standard mission phase duration and delta V budget with operational and performance margin such that a fixed vehicle design can fly all Mars opportunities of interest.
- Crew and one ton payload return to LEO for each manned mission.
- Quarantine and medical provision provided.
- Factors of safety set for metallic/nonmetallic/pressure structure.
- Failure tolerance/maintainability requirements identified.
- Mars Transfer Vehicle (MTV):
  - Transfer crew, cargo and propellants between LEO (SSF) and Mars orbit -
  Piloted vehicle (MTV) reusable without major maintenance operations for 5 complete missions
  - MTV shall transport crew, the fueled MEV (Mars Excursion Vehicle), its payload, and the ECCV (Earth Crew Capture Vehicle) from LEO to Mars orbit
  - MTV (optionally using the ECCV for crew capture at LEO) shall transport crew from Mars to LEO
  - Cargo MTV shall transport MEV and 100 ton total surface payload
  - EVA capability provided
  - Zero-g transit phase, determine penalties and design changes for artificial g
  - In-transit science to be performed on outbound and inbound legs
- Mars Excursion Vehicle (MEV):

- Vehicle shall transfer crew and cargo between Mars orbit and Mars surface
- Chemical propulson (LOX/LH2)
- Piloted version (MEV-P) expended at Mars
- MEV-P shall transport crew and 25 tons of payload from Mars orbit to Mars surface
- MEV-P shall transport crew and 1 ton gross science payload from Mars surface Mars orbit
- MEV-P shall have a contingency capability to house a crew up to 30 days on Mars surface, while a separate habitat is prepared
- Cargo MEV shall provide for landing 100 tons of cargo on Mars, using 2 MEVs at 50 t payload each

• Earth launch vehicle requirements imposed by MTV and MEV:
  - Payload of 140 tons to inclination of 28.5°, 500 km reference
  - Shroud size: 10 m diameter and 30 m length

8.3.4 Lunar Transportation Level I Requirements That Supported The Current Study

• Minimize transportation impact on science at SSF (goal).

8.3.5 Lunar Transportation Family Level II Requirements

• Earth-to-Orbit Transportation
  - All launches out of ETR
• Earth-to-Orbit Support Facilities
  - Minimal vehicle assembly activities in LEO desireable
• Space Transportation Vehicles
  - Chemical propulsion (LO2/LH2)
  - Up to 45 t delivered to the surface in a cargo mode
  - 4 crew + cargo to the surface in crew mode
  - LTV/LEV independent of lunar LOX availability assumptions
  - In-space refueling capability
  - Autonomous rendezvous and docking
  - Minimum vehicle activities required in LEO
• Crew Transfer Module
  - Supports crews of 4, 6, and 8 - designed nominally for 4.

- Lunar Surface System (LSS) Interface
  - Surface systems remove/load payload - unloads within 48 hours of landing
  - Solar flare protection provided by LSS

8.3.6 Lunar Transportation Family Level III Requirements

- Space Transportation Vehicles
  - LO2/LH2 propellant
  - Aerobrake option
  - Boiloff of cryogenic propellant
  - Design contingency on dry mass:
    - Existing hardware - 5%
    - Modified hardware (existing/moderate tech.) = 10%
    - New hardware (advanced tech.) = 20%
    - Performance flight reserves = 2%
  - Propellant transfer capability

8.4 Derived Requirements

In the current study, the derived requirements are grouped into six categories: Mars Transfer System Derived Requirements, MTV Derived Requirements, MEV Derived Requirement, Lunar System Derived Requirements, MRV-TMIS Derived Requirements, MTV-ECCV Derived Requirements, and Earth Orbit Support Facility Derived Requirements.

The derived requirements developed during the course of the current study have been documented in detail in Reference 39. A summary of these requirements are provided below.

In addition, the instructions on the use of this data base is provided in Reference 40.

8.4.1 Mars Transfer System-Derived Requirements

- Design Integration
  - Wake closure cone behind all aerobrakes is 44° wide. The total wake closure angle is centered on the velocity vector

- Equipment design life must account for mission duration plus one year
- All components designed for 5 missions with refurbishment (except aerobrake)
- Design for range of crew sizes, from 4 to 12
- L/D range from 0.5 to 1.0 for aerobrake vehicles at Mars

• GN&C
- 8500 m/s maximum entry velocity at Mars
- 150 m/s error-correction (post aerocapture)

• Propulsion
- Engine out capabilities in all mission phases. NTR engine out capabilities TBD
- Engine must continuously track C.G. of vehicle from beginning to end of all burns
- Maximum gimbal angle of engines ±12°
- All passive cryogenic thermal control system
- No MTV - TMIS fluid transfer before earth departure. (MEV tanks refrigerated or filled after MOC)

• Man Systems
- Solar Proton Event (SPE) protection to be provided
- Allow for direct viewing of all docking, berthing and landing procedures

• Structure and Mechanics
- All critical function lines and redundant systems shall be ranked such that a single accident does not incapacitate more than one functional path.
- All systems shall function up to 2 years in a dormant state having been subjected to the space environment.
- The airborne support equipment mass for launch to Earth orbit is assumed to be 15% for all hardware except the aerobrake, until better defined.
- Airborne support equipment mass assumption for the aerobrake shall be 20% of the aerobrake mass, until better defined.
- Aerobrake will be launched to Earth orbit in sections for on-orbit assembly as the reference case.
- MTV and MEV aerobrakes have common layout of attach points.
- Vehicle elements will have removable debris shield panel cladding for protection during LEO operations. These panels will be removed and saved in LEO to be used for the next mission-opportunity. The panels shall not add to the LEO debris environment.

- Mission vehicles will carry a robotic manipulation capability to inspect and maintain all exterior areas and systems.
- Structure optimized to minimize weight, operations, complexity and development effort.
- Greater than 30cm separation between all major vehicle exterior systems (i.e., tanks, modules).

• C&DH
  - Connectability between links maintained 90% of the time. Availability when scheduled - 98% connectability.

8.4.2 MTV Derived Requirements

• Design Integration
  - Crew module must accommodate alternative advanced propulsion options

• GN&C
  - Capture trajectory entry interface for aerocapture options not to exceed 6-g limit and to preclude an uncontrolled skip-out
  - Aerocapture exit errors not to exceed 0.25° inclination, RAAN (Right Ascension of the Ascending Node), and ARGP (ARGument of Periapsis), and a 10% period
  - GN&C requirement for advanced propulsion:
    ~ NTR - capture into planned orbit
    ~ EP (electric propulsion options)

• Electrical Power
  - Solar power to be used for transfer phase, batteries or fuel cells to be utilized for sun occultation time while in Mars orbit except for NEP.
  - NEP power derived from existing power system with a backup energy supply via fuel cells

• Man Systems
  - Added protection to crew from Solar Proton Events (SPE) will incorporate use of a "storm shelter".
  - Consumables stored will suffice for crew residence time from 443-1018 days (includes abort), assumes 100% ECLSS closure of water and oxygen, 0% closure on food and .25kg leakage per day
  - Two (2) astronauts able to pass through major circulation paths while wearing EVA suits.

- Crew quarters shall provide sufficient volume for casual conversation between at least two (2) crew members
- Volume per crew guidelines extrapolated from historical data
  ~ Transfer hab = 112m^3/crew
- Two independent pressurized volumes for safety
- Gravity condition emphasized to accommodate 0-g and 1-g and for surface commonality
- 2.3 m standard ceiling height for psychological and locomotion
- Crew visibility during all maneuvers (docking/rendezvous)
- There shall be 2 means of egress from each module for emergency escape
- Crew module to accommodate 0-g and induced g environments

• Structure and Mechanisms
  - Airborne support equipment for aerobrake shall be 20% of aerobrake mass.
  - All penetrations occur in barrel section to minimize mass.

8.4.3 MEV Derived Requirements

• Design Integration
  - Provide 15% of active weight for spares
  - MAV must be able to abort-to-orbit during powered descent phase
  - Twenty-five ton down payload on manned vehicles
  - Protective covers provided for all mission critical systems
  - "Down" delivered payload on manned vehicles
    ~ 25 mt down payload for reference MEV (include habitat module)
    ~ 0.7 mt down payload for the 'Mini-MEV' (crew habitat is provided by the ascent/descent cab)

• GN&C
  - L/D range from 0.5 to 1.0
  - Deorbit from 1 sol x 250 km periapsis orbit (nominal)
  - Engine start before aerobrake drop
  - Capture trajectory entry interface for MEV aerocapture at Mars not to exceed 6-g limit on crew members and equipment and to preclude an uncontrolled skipout of the Mars atmosphere
  - Landing accuracy after aerobrake jettison: unaided by landing beacons, 1km cep, with beacon 30m cep
  - Aerobrake jettisoned in controlled manner during powered descent phase

- Currently, cross range = 1500 km for high L/D aerobrake
- Landing approach descent path angle = 15°

- Propulsion
  - Pre-descent checkout of engines to be provided (checkout extent TBD)
  - One (1) meter clearance established between engine bells and surface
  - Engine out capabilities for ascent/descent stages
  - Passive cryogenic storage system: MLI with vapor cooled shields
  - Gravity field environment eliminates need for zero-g acquisition and venting
  - Vacuum jacketed ascent tanks for Mars boiloff reduction
  - MEV propellant transferred from MTV prior to descent
- Electrical Power
  - Solar arrays to supply power following separation from MTV for ~ 50 day approach to Mars. Arrays to be retracted 12 hours prior to Mars encounter, power shall be provided by batteries or other internal source
  - Batteries or fuel cells to provide power for ascent and descent phases
- ECLSS
  - Capability of two (2) crew cab repressurizations
- Man Systems
  - Consumables will suffice for a crew residence time of 30 - 600 days dependent on mission stay time and abort scenarios, assumes 100% ECLSS closure of water and oxygen, 0% closure of the food and .15 kg leakage per day
  - The maximum surface stay time is 600 days
- Structure and Mechanisms
  - Shall be at least two (2) functionally independently pressurized areas for emergency conditions. There shall be two (2) EVA suits stored in these areas
  - Establish 30cm clearance between all elements to allow for movement during high-stress maneuvers
  - Crew cab to have SSF diameter (4.4m), width (1.4m), and penetrations and attachments occur at rings
  - Surface hab system to be removable later by surface construction transport vehicle and protected from damage by MAV blast during ascent start

8.4.4 Lunar Transportation Family Derived Requirements

- Mission Level
  - Vehicles to accommodate the following Missions:
    - Tandem Direct initially
    - Evolve to Lunar Orbit Rendezvous (LOR)
    - L2 missions accommodated with tandem direct vehicles
    - Stand alone "sortie" mission to a site of interest
      - -- 4 crew
      - -- Daytime only (14 days)
  - Space based vehicles
  - On-orbit assembly minimized
  - Vehicles designed to be integrated, intact, into a 10m x 30m by 100t class launch vehicle
  - Minimum of 5 reuses with up to 30 year servicing life
  - System evolution to Mars rather than entire vehicle commonality
  - System flexibility for changing mission modes:
    - Changing crew size
    - Changing payloads
    - Increasing propellant loads
- Habitation Modules
  - All habitation modules sized volumetrically from historical spacecraft volumetric data
  - Direct visibility of all landing and docking procedures
  - Size hab modules for crew of 6 or 7 days for evolution scenarios
- Propulsion System
  - Maximum engine gimbal = 18.5° to limit steering loss to 5%
  - 1 engine-out capability
  - Advanced RL-10 30 klbf. engines
  - Throttling or gimballing capabilities for steering in engine-out mode
- Transfer Vehicle
  - Capability to rendezvous with Space Station Freedom
  - Accommodate 4 crew initially for up to 28 days
  - Single up/down orientation during both landing and aerobraking
  - Contiguous transfer to excursion vehicle
- Excursion Vehicle

- Return crew of 4 and 500 kg of samples to LLO
- Deliver 25 t of cargo to LS in non-reusable mode
- Deliver crew of 4 -10 t payload in crew reusable mode
- Designed to accommodate varying, nonsymmetrical payloads
- Payload integration optimized for surface access
- Crew module placement optimized for surface access and visibility
- Contiguous transfer from LTV
- Sustain 4 crew for 3 to 5 days
- An airlock is mandatory for lunar surface residences beyond 3 days
- Landing gear requirements:
    - Designed for 12° slope
    - Max. surface undulation between footpads = 1 m

• Aerobrake
- L/D = 0.25
- Sized to accommodate a reusable system
- Derivative of the Mars L/D = 0.5 shape

8.4.5 MTV - TMIS Derived Requirements

• Design Integration
- On-orbit assembly to be minimized to extent practical.
- Flexible to support reference missions (interconnect design to support reference mission requirements.
- Fully modularized to utilize ETO capacity, the amount of modularization shall be a function of the ETO vehicle chosen.
- Assembly to be accomplished on-orbit, remotely and robotically.

• Propulsion
- Passive thermal control system including zero-g thermodynamic vent system coupled to multiple vapor cooled shields.
- TMIS insulating system is a continuously purged MLI over foam design optimized for minimum ground-hold, launch, and orbital boiloff. Includes vapor cooled shield (coupled to TVS) outside of foam.
- TMIS tanks launched late in assembly sequence to minimize orbital stay time before TMI burn (6 months).
- MTV tank insulation system is 2-4 inch thick MLI blankets. Multiple vapor cooled shields placed at optimum points in the MLI.

- Reference vehicle is launched "wet" with top-off (dry/wet issue to be traded).

- Structure and Mechanisms
  - Thrust structure - tanks - intertanks used as primary structure for cryo/aerobrake vehicles only.
  - The airborne support equipment mass for launch to Earth is assumed to be 7% for all hardware sets.

8.4.6 MTV - ECCV Derived Requirements

- GN&C
  - Capture trajectory entry interface for ECCV aerocapture or aeroentry into Earth atmosphere not to exceed 6-g limit on crew and personnel, and to preclude an uncontrolled skip out of Earth atmosphere.
  - L/D = 0.25.
- Structure and Mechanisms
  - Interior materials must conform to NASA standards for outgassing, fire hazards, etc.

8.4.7 Earth Orbit Support Facility Derived Equipment

- Groundrules:
  - Multiple (ETO) flights will be used in assembly
  - Line of sight communications are to be used
  - Extensive use of robotic and telepresent systems will be made
  - Minimal EVA activities
- Assumptions:
  - On-orbit propellant fueling or launched wet propellant tanks may be used
  - Two RMS systems will be used in assembly
  - On-orbit spares will be 15% of vehicle weight
  - Robotic software and sensors will allow supervisory human control
  - Proximity operations will be viewed directly or by video
  - Assembly schedule will be four years or less

8.5 STCAEM Requirements Data Base

The Space Transfer Concepts and Analysis for Exploration Mission Requirements Database is a user-friendly, restricted access system that contains all the requirements identified in the STCAEM study.

The Macintosch SE computer and the 4th Dimension data base program were used to develop the requirements data base. A customized user interface has been created to enable anyone to take advantage of the information contained in the data base. As a result of this customization, a user does not need to be familiar with the workings of 4th Dimension; simple, easy to follow menus and help screens allow anyone to benefit from the data base. The system was designed and developed to run on a Macintosh SE with a standard nine inch monochrome monitor.

In this initial development of the requirements data base, a simple "flat file "structure was selected to store each requirement. This file structure allows for efficient entry of data. At the same time it provides for rapid search and retrieval of the data base contents; this design provides a reliable structure which can be easily modified. The complex structures available in the 4th Dimension data base software are not needed by the requirements data base in its present form. In the future when additional, more detailed requirements have been defined, this data base can easily be modified to make use of any specific 4th Dimension features.

The information for each requirement included in the data base is shown in Figure 8-1, with details provided in References 39 and 40.

ITEM NAME	DESCRIPTION	EXAMPLES/COMMENTS
Date	Requirement date	11/02/89
Type	Requirement classification	Mission, Technology, Systems
Origin	Who generated requirement	BOEING, NASA, MASE
System	Requirement identification	MEV, MTV, etc.
Subsystem	Additional system identification	Ascent Propulsion
Component	Additiona subsystem identification	Propellent Gauging System
Requirement	Actual requirement	
Rationale	Reason for requirement	
Reference	Requirement reference	Document, Person

DATA BASE CONTENT
FIGURE 8-1

9. PROGRAMMATICS

The objectives of the Programmatics task during the current phase of the study were: (1) realistic initial schedules that include initial critical path program elements; (2) initial descriptions of new or unique facilities requirements; (3) development of a stable, clear, responsive work breakdown structure (WBS) and WBS dictionary; (4) initial realistic estimates of program costs, cost uncertainties, and funding profile requirements; (5) initial risk analysis, and (6) early and continuing infusion of programmatics data into other study tasks to drive requirements/design/trade decisions.

The issues addressed during the study included: (1) capturing all potential long-lead program items such as precursor missions, technology advancement and advanced development, related infrastructure development, support systems and new or modified facility constuction, since these are as important ascost and funding in assessing goal achievability; (2)incorporating sufficient operating margin in schedules to obtain high probability of making the relatively brief Mars launch windows; (3) the work breakdown structure must support key study goals such as commonality and (4) cost estimating accuracy and uncertainty are recurring issues in concept definition studies.

9.1 Introduction

The study flow, as required by MSFC's statement of work, began with a set of strawman concepts, introduced others as appropriate, conducted "neckdowns", and concluded with a resulting set of concepts and associated recommendations.

As the study progressed, much discussion among the SEI community centered on "architectures". In this study, architectures were more or less synonymous with concepts, since the statement of work required that each concept be fully developed including operations, support, technology, and so forth.

We started with ten concepts as shown in Figure 1-3 (in the Executive Summary). After the "neckdown" was completed, significant effort was put into programmatics.

As was indicated earlier, we established three levels of activity to evaluate in-space transportation options. The minimum was just enough to meet the President's objectives; in fact "return to the Moon to stay" was interpreted as permanent facilities but not permanent human presence. The minimum program had only three missions to Mars. The median (full science) program aimed at satisfying most of the published science objectives for Lunar and Mars exploration. The maximum program aimed for industrialization of the Moon, for return of practical benefits to Earth, and for the beginnings of colonization of Mars. The range of activity levels, as measured by people and materiel delivered to planetary surfaces, was about a factor of 10. The range of Earth-to-orbit launch rates was less, since we adopted results of preliminary trade studies, selecting more advanced in space transportation technologies as baselines for greater activity levels. The high level schedules developed for these three levels of activity were shown in Figures 1-16 to 1-18 and a comparison of them for both Lunar and Mars was shown in Figures 1-19 and 1-20.

9.2 Schedule/Network Development Methodology

A PC system called Open Plan by WST Corporation was used, which allows direct control and lower cost over a larger (mainframe) system. The network was purposely kept simple. Summary activities were used in development of the networks. When detailed to a lower level, some activities will require a different calendar than we used. One calendar with a five day work week - no holiday was used. Utilizing multicalendars on a summary network could confuse the development. The Preliminary WBS Structure Level 7 was followed for selection of work to be detailed. An example of Level 7 is: MEV Ascent Vehicle Structure/Mechanisms. We then developed a generic logic string of activities with standard durations for like activities. This logic was then applied against each WBS Level 7 element. To establish interface ties between logic strings and determination of major events, we used the Upper Level Summary Schedule and Summary Level Technology Schedule.

9.2.1 Goals/Purpose

There were two goals for the schedule/network development. These were:

a. Guidelines for Future Development. The schedules are a preliminary road map to follow in the development program.

b. Layout Basis Framework for Network. The networks can be used for future detail network development. This development can be in phases retaining unattended logic for areas which can be be detailed.

9.2.2 Status

Six preliminary networks have been developed. They are:

- Lunar minimum
- Lunar full science
- Lunar industrialization
- Mars missions
- Mars full science
- Mars settlement

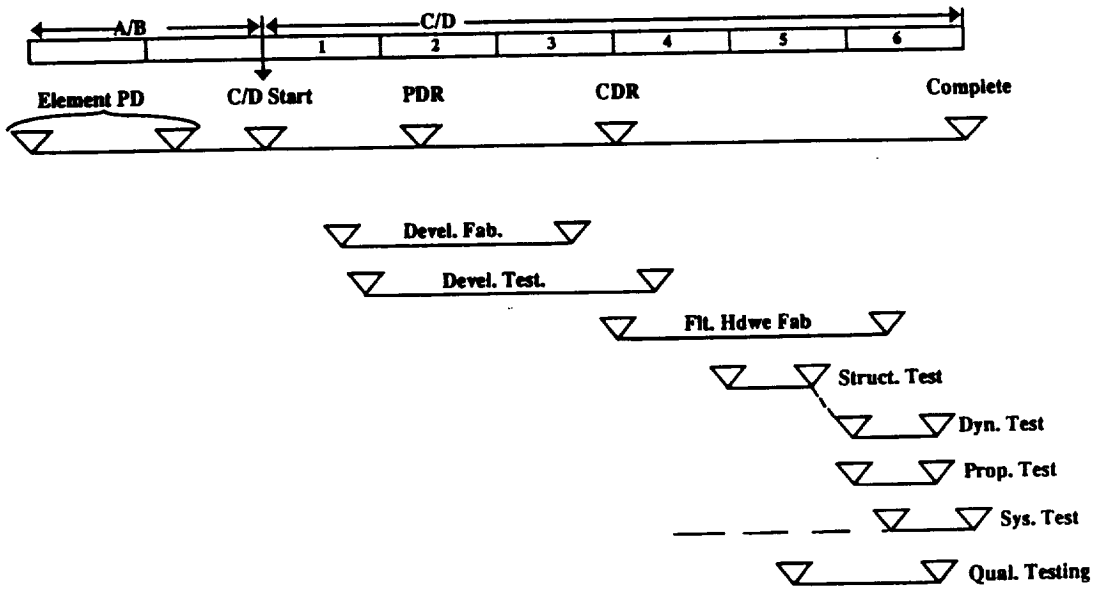
These networks will be further developed as information becomes available The technology development plan schedules are shown in Figure 7-8; an example of the standard 6 year program phase C/D schedule is shown in Figure 9-1. The network schedules developed during the study are available in Reference 12.

9.3 Facilities

The facility requirements and approaches are discussed in Section 6.

9.4 Work Breakdown Structure

The approach to developing a WBS tree and dictionary was to use the Space Station Freedom Work Package One WBS as a point of depature to capture commonality, modularity and evolution potentials. We worked with MSFC to evolve the WBS illustrated in Figure 9-2. The WBS dictionary details are provided in Reference 41.



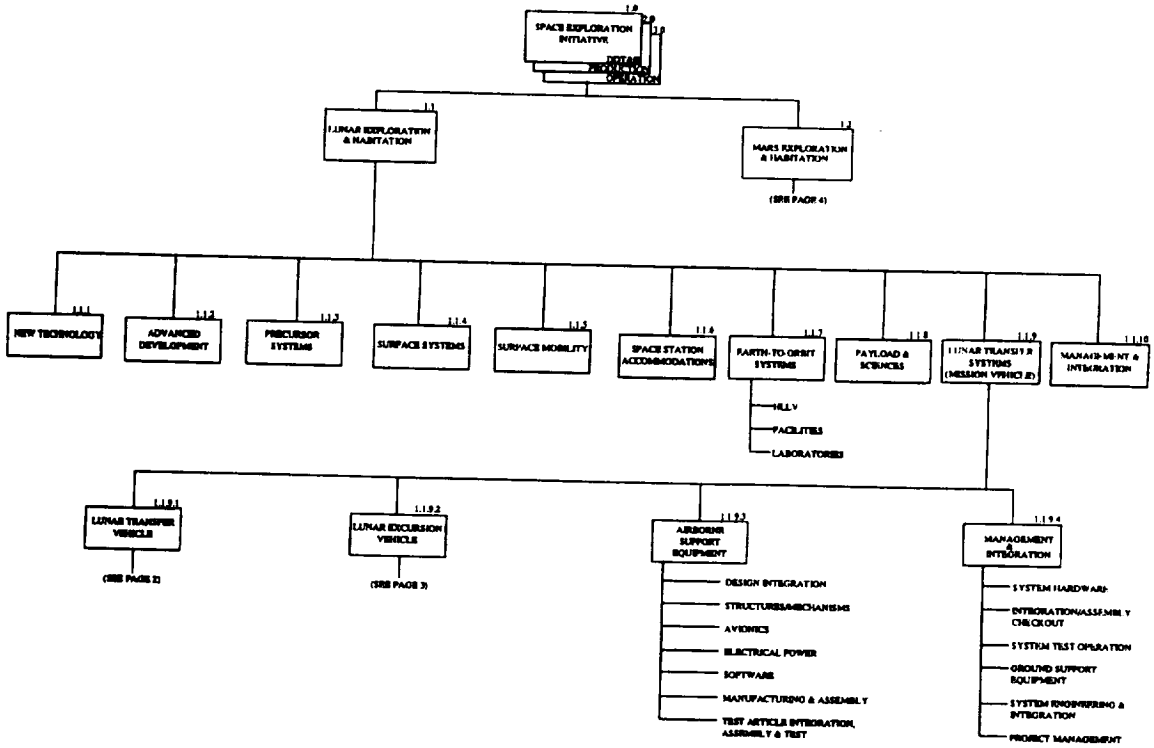
STANDARD SIX YEAR PROGRAM PHASE C/D DEVELOPMENT SCHEDULE
FIGURE 9-1

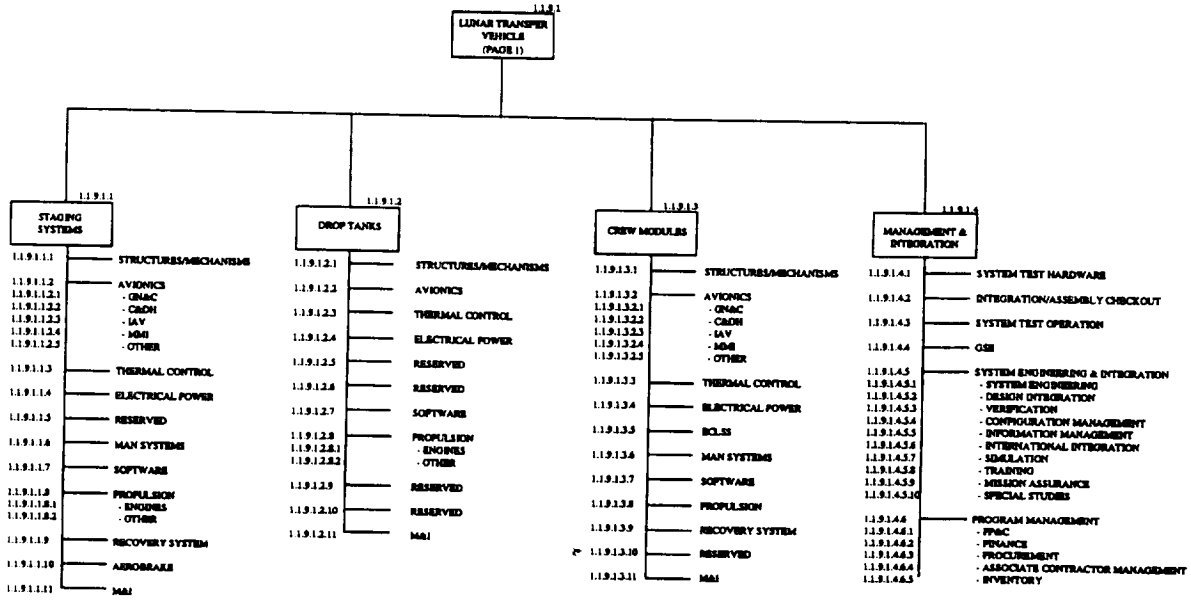
9.5 Cost Data

9.5.1 Overall Approach

Space transfer concept cost estimates were developed through parametric and detail estimating techniques using program/scenario plans and hardware and software descriptions combined with NASA and subcontractor data. Our estimating approach simulates the aerospace development and production environment. It also reflects program options not typical of aerospace programs. This flexibility allows assessment of innovative program planning concepts.

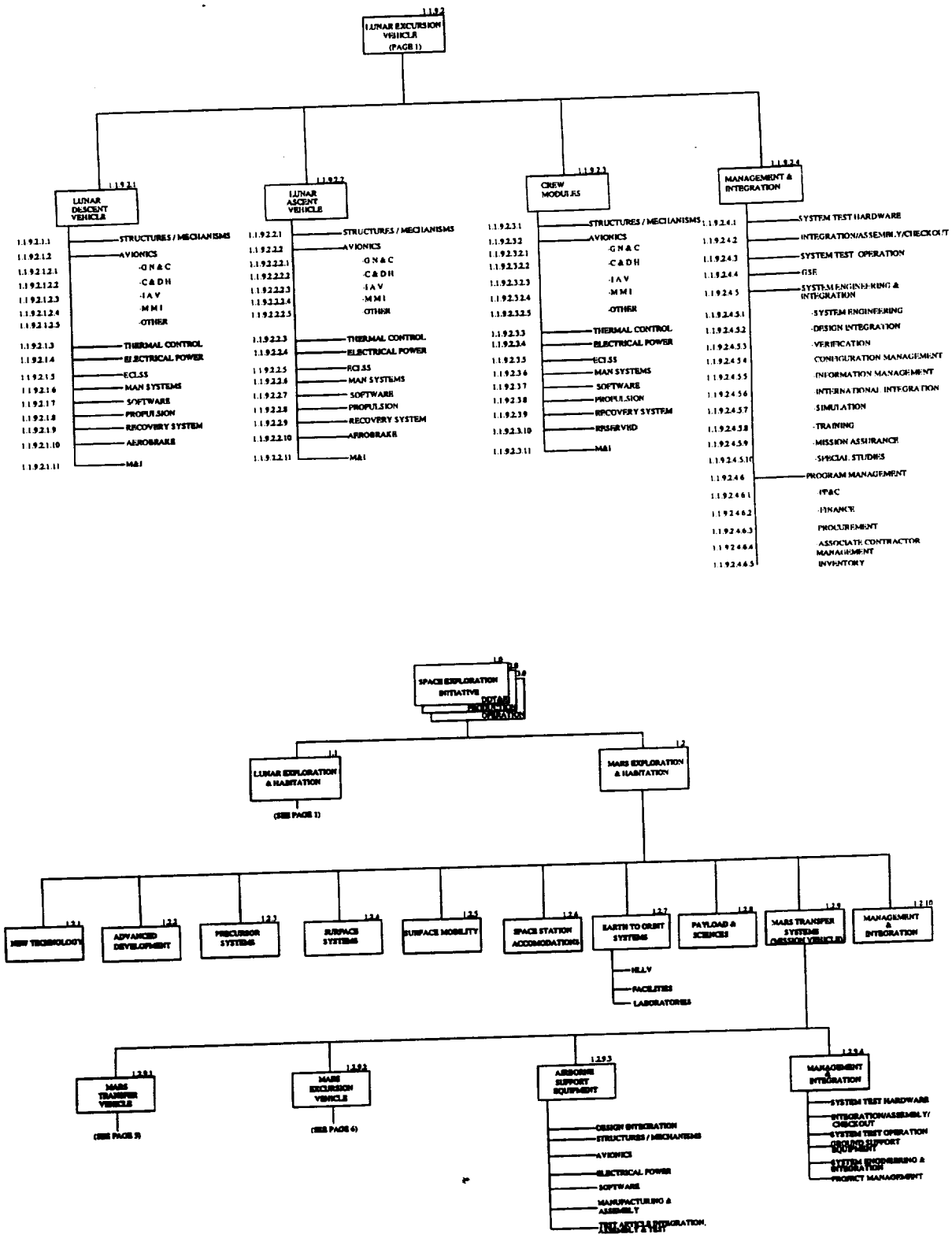
Several tools were employed in this analysis. For developing estimates the Boeing Parametric Cost Model (PCM) designed specifically for advanced system estimating was used. It utilizes a company-wide, uniform computerized data base containing historical data compiled since 1969. The second major tool is a Boeing developed Life Cycle Cost Model. The third tool is the Boeing developed Return on Investment (ROI) Analyses.





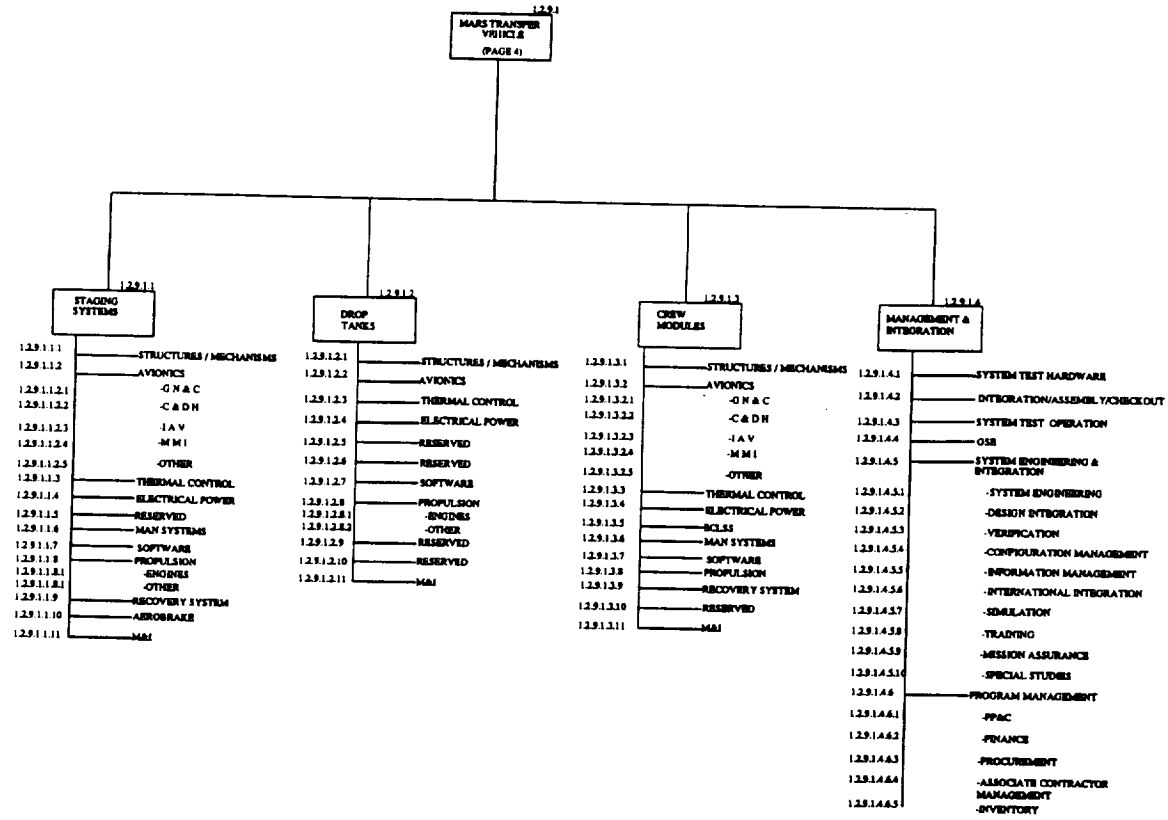
SEI WORK BREAKDOWN STRUCTURE TREE

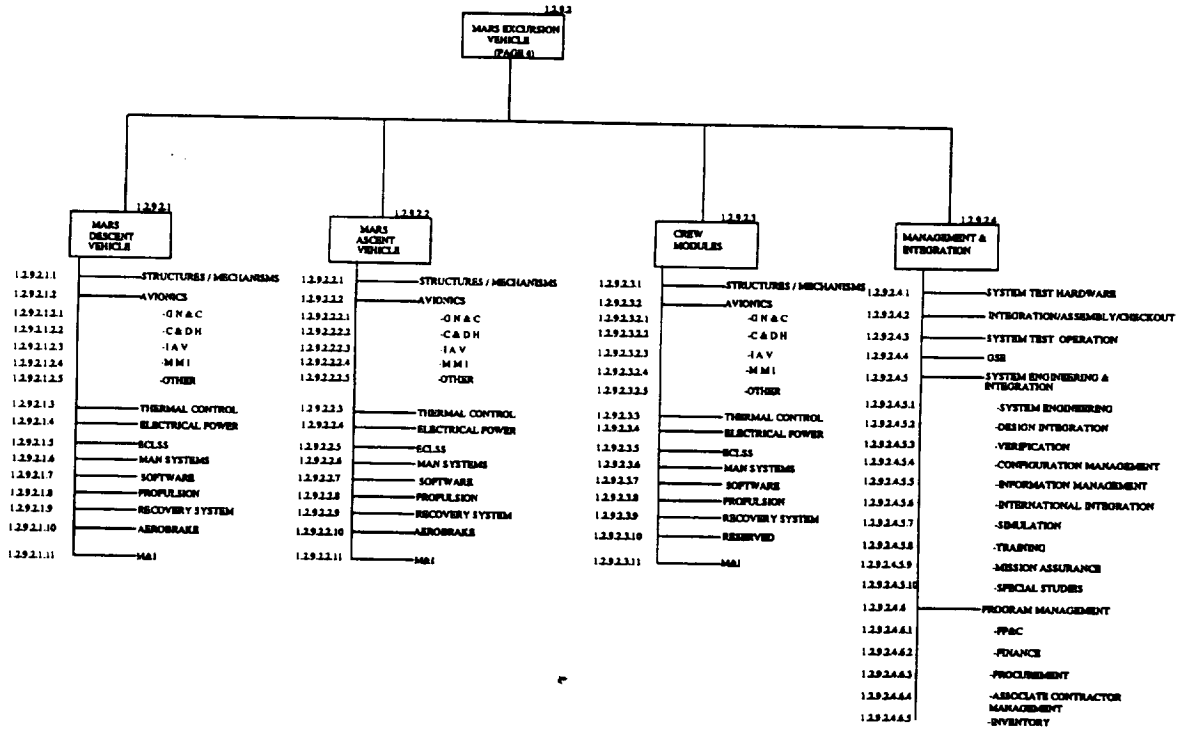
FIGURE 9-2



SEI WORK BREAKDOWN STRUCTURE TREE (CONTINUED)

FIGURE 9-2





SEI WORK BREAKDOWN STRUCTURE TREE (CONCLUDED)

FIGURE 9-2

The approach to cost estimating was to use the PCM to establish DDT&E and manufacturing cost of major hardware components or to use other estimates, (e.g. Nuclear Working Group estimator) if they were considered superior and then feed them to LCC model. Variations on equipment hardware or mission alternatives can be run through the LCC and then compared for a return on investment. This flow is illustrated in Figure 1-29. We were able to investigate alternative concepts quickly, giving system designers more data for evolving scenario/mission responsive concepts. Transportation concepts, trade studies, and "neckdown" efforts were supported by this approach.

9.5.2 Parametric Cost Model

PCM develops cost from the subsystem level and builds upward to obtain total program cost. Costs are estimated from physical hardware descriptions (e.g., weights and complexities) and program parameters (e.g., quantities, learning curves, and integration levels). Known costs are input directly into the estimate when available; the model assesses the necessary system engineering and system test efforts needed for integration into the program. The PCM working unit is man-hours, which allows relationships that tie physical hardware descriptions first to design engineering or basic factory labor, and then through the organizational structure to pick up functional areas such as systems engineering, test, and development shop. Using man-hours instead of dollars for estimating relationships enables more reliable estimates. The PCM features, main inputs, and results are shown in Figure 9-3. The applicable PCM results are then put into the Life Cycle Cost Model to obtain cost spreads for the various missions/programs. The various hardware components costed for the three different missions/programs are shown in Figure 9-4

9.5.3 Life Cycle Cost Model

The LCCM cost data is a composite of HLLV costs, launch base facilities cost estimate based on $/sq. ft. and parametric estimates derived from the Parametric Cost Model. The principal source of information is from the PCM. All hardware cost estimates, with the exception of HLLV, have been developed with this model.

Features

- Designed specifically for advanced system estimating
- Uses company-wide, uniform computerized data base
- Contains historical data compiled since 1969
- Allows direct input of known costs into the estimate

Main Inputs	Results
• Hardware Characteristics - Category (e.g., primary structure, power conditioning, etc.) - Weight (or Thrust) - Complexity - % Off-the-Shelf - Maturity - Quantity - Manufacturing Learning Curve • Support Cost Factors - Systems Engineering - Management - Operations - Spares	• DDT&E and Manufacturing Estimates - Based on previous Boeing programs - Provides first flight unit costs - Excludes test hardware - Excludes fees • New hardware must be relatable to PCM database to produce reasonable estimate • PCM estimates improve with increasing hardware detail.

PARAMETRIC COST MODEL TOP LEVEL FEATURES

FIGURE 9-3

Components		*Lunar/Mars*		
		Minimum	Full Science	Settle/Ind
HLLV	Cargo Carrier & Core	X	X	X
	STME	X	X	X
	Recov PA Mod	X	X	X
Propulsion	Std Avionics Suite	X	X	X
	Adv Space Engine	X	X	X
	NTR Tanks		X	
	MOC Tank	X		X
	MOC Core	X		X
	NTR Stage		X	
	NTR Engine		X	
	NEP Stage			X
	NEP Engine			X
	TMIS Engine	X		X
	TMIS Tank	X		X
	TMIS Core	X		X
Modules	LEO Tanker	X	X	X
	LTV Hab	X	X	X
	LTV	X	X	X
	LEV	X	X	X
	LEV Crew Module	X	X	X
	MTV	X		X
	MTV Crew Module	X	X	X
	MEV	X	X	X
	RMEV			X
	mini-MEV		X	
	MEV Crew Module	X	X	X
	Lunar Aerobrake	X		
	MTV Aerobrake			
	MEV Aeroshell	X	X	X
	MCRV	X	X	X

MISSION HARDWARE COMPONENTS

FIGURE 9-4

9.5.4 Return On Investment

One of the principal uses of the LCCM is to develop trades and return on investment for technology options. As shown in Figure 1-29, two separate life cycle cost models (which include DDT&E and production cost data derived from the parametric cost models) must be developed for each ROI case; a reference, and a case utilizing a technology option. The two life cycle cost streams are separately entered, and the ROI model is executed. The flow also illustrates that not all of the data entered into the life cycle cost model is derived from available costing software. Technical analysis must accompany this data. For example, the number of units which must be produced for the DDT&E program must be determined. This is done at the subsystem level based on knowledge of past programs, and proposed system/subsystem tests.

9.5.5 Results

A summary of the cost data produced by the PCM for the NTR reference vehicle are given in Figure 9-5. The PCM program was used to produce DDT&E and production cost estimates for each of our reference Mars and lunar vehicles to the subsystem level. The DDT&E costs generated by the PCM did not include all of the necessary hardware for the first mission vehicle. Hence all necessary additional units (prototypes,test units, lab units, etc.) were added into the vehicle cost buildups. An example of a cost build up derived for each of the reference vehicles for entry into the LCC model is shown in Figure 9-6. Again this example is for the reference NTR vehicle utilized in the full science Mars scenario. As shown the total DDT&E includes additional costs(eg. additional units in the DDT&E program), contractor fees and the engineering wrap factor. The total DDT&E from the cost buildup and the unit cost from the PCM are the primary vehicle cost inputs to the LCC model

The unit costs developed from the LCC model is shown in Figure 9-7. Cost estimates for the three proposed mission programs are given in Figures 9-8 to 9-10 and illustrated graphically in Figures 1-42 to 1-45. Additional detail on cost and schedule developments is provided in Reference 12.

Item	Engineering ($Millions)	Manufacturing ($Millions)	Total ($Millions)
Trans Mars Injection Stage	473.385	532.611	1005.996
Remote Manipulator	19.701	44.303	64.004
Mars Transfer Crew Module	1138.947	925.055	2064.003
Science	100.651	62.517	163.167
Mars Excursion Stage	58.783	134.310	193.093
Aeroshell	99.473	51.556	151.030
Mars Excursion Vehicle Crew Cab	315.766	110.413	426.178
Modified Crew Return Vehicle	273.312	199.326	472.637
Hardware Final Ass'y and C/O	---------	309.013	309.013
Spares	---------	6.180	6.180
Hardware Total Costs	2480.019	2375.283	4855.301
System Engineering & Integration	451.488	---------	451.488
Software Engineering	317.146	---------	317.146
Systems Ground Test Conduct	1771.157	---------	1771.157
Systems Flight Test Conduct	---------	---------	---------
Peculiar Support Equipment	802.895	118.455	921.350
Tooling & Special Test Equipment	---------	715.107	715.107
Task Direct Quality Assurance	---------	228.593	228.593
Logistics	135.055	---------	135.055
Liaison Engineering	205.851	---------	205.851
Data	53.665	---------	53.665
Training	O/H	---------	---------
Facilities Engineering	O/H	---------	---------
Safety	O/H	---------	---------
Graphics	O/H	---------	---------
Outplant	O/H	---------	---------
Program Management	O/H	---------	---------
Support Effort Total	3737.247	1062.155	4799.402
Total Estimate	6217.266	3437.438	9654.703

PCM NTR COSTS (TYPICAL)

FIGURE 9-5

	A	B	C	D	E	F	G	H
1		Eng'r Cost	Cost Δ's	Wrap Factor	Total D&D	Unit Cost	# Units in DDT&E	
2	TMIS Core	439.2	0	2.79	1225.368	225	3.5	
3	TMI Tank	32		2.79	89.28	48.9	2	
4	TMIS Tank	17	0	2.79	47.43	8.63	2	
5	TMIS Nuc Eng	468	0	2.79	1305.72	150	2	
6	Test Facil	0	2255	1	2255	0	1	
7	Crew Module	1113	0	2.79	3105.27	1281	3.1	
8	MEV Stg	66.8		2.79	186.372	107.5	3.5	
9	MEV Engine	0	0	2.79	0	8	5	
10	MEV Aeroshell	112.1		2.79	312.759	64.7	2	
11	Mev CM	142		2.79	396.18	126.5	3.5	
12	EDCV	280		2.79	781.2	214	3.5	
13								

	I	J	K	L	M	N	O
1	DDT&E no Fe	Fee Factor, %	Total DDT&E	Units/Msn	Unit $/Msn	Msn Cost w/ Fee	
2	2012.868	8	2173.89744	0.3	67.5	72.9	
3	187.08	8	202.0464	2	97.8	105.624	
4	64.69	8	69.8652	2	17.26	18.6408	
5	1605.72	8	1734.1776	2	300	324	
6	2255	8	2435.4	0	0	0	
7	7076.37	8	7642.4796	0.25	320.25	345.87	
8	562.622	8	607.63176	1	107.5	116.1	
9	40	8	43.2	7	56	60.48	
10	442.159	8	477.53172	1	64.7	69.876	
11	838.93	8	906.0444	1	126.5	136.62	
12	1530.2	8	1652.616	1	214	231.12	
13		Grand Total	17742.8437				1375.6068

NTR COST BUILDUP FROM PCM

FIGURE 9-6

C-5

Components		DDT&E Cost	Unit Cost
HLLV	Cargo Carrier & Core	1993.0	160.0
	STME	3000.0	5.0
	Recov PA Mod	1500.0	150.0
Propulsion	Std Avionics Suite	150.0	27.0
	Adv Space Engine	564.0	8.0
	NTR Tanks	202.1	48.9
	MOC Tank	69.9	8.6
	MOC Core	162.0	204.5
	NTR Stage	2173.9	225.0
	NTR Engine	1734.2	150.0
	NEP Stage	12256.1	3290.0
	NEP Engine	4997.2	800.0
	TMIS Engine	2423.8	26.8
	TMIS Tank	161.1	42.5
	TMIS Core	1103.2	225.0
Modules	LEO Tanker	350.0	50.0
	Cargo Transfer Veh.	0.0	0.0
	LTV Hab	3057.2	438.0
	LTV	1413.3	231.0
	LEV	1022.0	143.0
	LEV Crew Module	945.8	153.0
	MTV	951.9	165.8
	MTV Crew Module	7642.5	1281.0
	MEV	607.6	107.5
	RMEV	7285.4	1336.0
	mini-MEV	2531.8	400.0
	MEV Crew Module	906.0	126.3
	CELSS	0.0	0.0
	Lunar Aerobrake	331.8	38.0
	MTV Aerobrake	626.4	80.5
	MEV Aeroshell	477.5	64.7
	MCRV	1652.6	214.0
Power Systems	Solar/RFC Power Units	0.0	0.0
	Space/Surface TCS	0.0	0.0
	SP-100 Reactor PSPS	0.0	0.0
	MultiMeg Reactor	0.0	0.0
	TurboGenerator	0.0	0.0
	PwrPlt Heat Rej Sys	0.0	0.0
	PMAD Equip	0.0	0.0
	Lunar ISRU Equip	0.0	0.0
	Mars ISRU Equip	0.0	0.0
Rover	Small Rover Chassis	0.0	0.0
	Small Wh/Dr System	0.0	0.0
	Large Rover Chassis	0.0	0.0
	Straddler Chassis	0.0	0.0
	Large Wh/Dr Sys	0.0	0.0
Facilities	LEO Node	2000.0	150.0
	HLLV Launch Fac	5500.0	
	Cape Fac	89.6	
	NTR Test	2255.0	
	NEP Test	1600.0	
Other	Science Equip	0.0	0.0

Factors	
Hdwe Integ Factor	0.250
L. C. Cost Exponent	-0.152
Learning Factor	0.900
Refurbishment	0.250
Crew Flt Cost to LEO	150.000
Node in-space IA&CO, % of node/hdwe DDT&E & mfr.	5.000
Node DDT&E	2000.000
Node Delta #1	2000.000
Node Delta #2	0.000
Trans Factor	0.0022046
Node Turnaround Cost	125.000
Spread-2	
"(1)"	0.360
"(2)"	0.640
Spread-4	
"(1)"	0.100
"(2)"	0.240
"(3)	0.340
"(4)"	0.320
Spread-6	
"(1)"	0.050
"(2)	0.110
"(3)	0.180
"(4)	0.230
"(5)	0.240
"(6)	0.190
Spread-8	
"(1)"	0.025
"(2)	0.064
"(3)	0.106
"(4)	0.143
"(5)	0.171
"(6)	0.184
"(7)	0.175
"(8)	0.132

DDT AND E AND UNIT COST FOR SEI HARDWARE

FIGURE 9-7

Program Year	Total Development	Total Production	Operation & Support	Program SE&I	Total Program	Cumulative Program
90	0.0	0.0	0.0	0.0	0.0	0.0
91	0.0	0.0	0.0	0.0	0.0	0.0
92	0.0	10.0	2.0	1.2	13.2	13.2
93	0.0	10.0	2.0	1.2	13.2	26.4
94	0.0	20.0	4.0	2.4	26.4	52.8
95	303.6	50.0	70.7	42.4	466.7	519.5
96	868.5	50.0	183.7	110.2	1212.4	1732.0
97	1521.0	50.0	314.2	188.5	2073.8	3805.7
98	2123.8	110.0	446.8	268.1	2948.6	6754.3
99	2599.0	0.0	519.8	311.9	3430.7	10185.0
00	2859.2	0.0	571.8	343.1	3774.1	13959.2
01	2797.4	0.0	559.5	335.7	3692.6	17651.8
02	2242.4	0.0	448.5	269.1	2959.9	20611.7
03	484.6	324.4	161.8	97.1	1067.9	21679.6
04	5.7	1453.3	291.8	175.1	1925.9	23605.4
05	334.2	1777.6	422.4	253.4	2787.6	26393.0
06	1026.8	1453.3	496.0	297.6	3273.7	29666.7
07	1860.0	804.5	532.9	319.7	3517.2	33183.9
08	2648.9	1777.6	885.3	531.2	5843.0	39027.0
09	3282.3	1104.5	877.4	526.4	5790.6	44817.6
10	3652.1	780.1	886.4	531.9	5850.5	50668.1
11	3618.4	1104.5	944.6	566.8	6234.3	56902.4
12	3994.2	455.8	690.0	414.0	4553.9	61456.3
13	965.3	2831.7	759.4	455.6	5012.1	66468.3
14	0.0	3697.9	739.6	443.8	4881.3	71349.6
15	0.0	4138.3	827.7	496.6	5462.6	76812.2
16	0.0	1732.3	346.5	207.9	2286.7	79098.8
17	0.0	3223.6	644.7	386.8	4255.1	83354.0
18	0.0	3799.3	759.9	455.9	5015.1	88369.0
19	0.0	3693.2	738.6	443.2	4875.0	93244.1
20	0.0	1601.7	320.3	192.2	2114.2	95358.3
21	0.0	3799.3	759.9	455.9	5015.1	100373.4
22	0.0	2817.5	563.5	338.1	3719.1	104092.5
23	0.0	2634.0	526.8	316.1	3476.9	107569.4
24	0.0	348.8	69.8	41.9	460.4	108029.7
25	0.0	997.5	199.5	119.7	1316.7	109346.4

LIFE CYCLE COST FOR MINIMUM PROGRAM

FIGURE 9-8

Program Year	Total Development	Total Production	Operation & Support	Program SE&I	Total Program	Cumulative Program
90	0.0	0.0	0.0	0.0	0.0	0.0
91	137.5	0.0	27.5	16.5	181.5	181.5
92	518.1	10.0	105.6	63.4	697.1	878.6
93	1008.2	10.0	203.6	122.2	1344.0	2222.5
94	1858.6	20.0	375.7	225.4	2479.8	4702.3
95	2751.8	50.0	560.4	336.2	3698.3	8400.7
96	3664.7	50.0	742.9	445.8	4903.4	13304.1
97	4258.6	50.0	861.7	517.0	5687.3	18991.4
98	4376.3	110.0	897.3	538.4	5921.9	24913.3
99	3480.1	1342.5	964.5	578.7	6365.8	31279.1
00	1925.4	3278.4	1040.8	624.5	6869.0	38148.0
01	2096.8	2688.8	957.1	574.3	6316.9	44465.0
02	1619.0	3133.5	950.5	570.3	6273.4	50738.4
03	2256.8	2610.4	973.5	584.1	6424.8	57163.2
04	3148.8	2328.0	1095.4	657.2	7229.4	64392.6
05	2955.3	2521.6	1095.4	657.2	7229.5	71622.1
06	2661.0	2914.9	1113.2	669.1	7360.2	78982.3
07	2236.5	3354.8	1118.3	671.0	7380.6	86362.8
08	1687.0	4289.1	1195.2	717.1	7888.4	94251.3
09	0.0	5366.1	1073.2	643.9	7083.2	101334.5
10	0.0	2937.0	587.4	352.4	3876.8	105211.2
11	0.0	3363.7	672.7	403.6	4440.1	109651.3
12	0.0	3947.2	789.4	473.7	5210.4	114861.7
13	0.0	3911.1	782.2	469.3	5162.6	120024.3
14	0.0	4213.2	842.6	505.6	5561.4	125585.7
15	0.0	4557.7	911.5	546.9	6016.2	131601.9
16	0.0	2809.9	562.0	337.2	3709.1	135311.0
17	0.0	3886.1	777.2	466.3	5129.6	140440.6
18	0.0	3888.4	777.7	466.6	5132.7	145573.4
19	0.0	4012.7	802.5	481.5	5296.8	150870.1
20	0.0	3999.1	799.8	479.9	5278.8	156149.0
21	0.0	3842.3	768.5	461.1	5071.9	161220.9
22	0.0	3716.7	743.3	446.0	4906.1	166126.9
23	0.0	3885.9	777.2	466.3	5129.4	171256.3
24	0.0	1436.4	287.3	172.4	1896.0	173152.3
25	0.0	1760.2	352.0	211.2	2323.5	175475.8

LIFE CYCLE COST FOR FULL SCIENCE PROGRAM

FIGURE 9-9

9.6 Risk Analyses

Risk analyses were conducted to develop an initial risk assessment for the various architectures. This presentation of risk analysis results considers development risk, man-rating requirements, and several aspects of mission and operations risk.

9.6.1 Development Risk

All of the architectures and technologies investgated in this study incur some degree of development risk; none are comprised entirely of fully developed technology. Development risks are correlated directly with technological uncertainties. We identified the following principal risks:

Cryogenics - High-performance insulation systems involve a great many layers of multi-layer insulation (MLI), and one or more vapor-cooled shields. Analyses and experiments have indicated the efficacy of these, but demonstration that such insulation systems can be fabricated at light weight, capable of surviving launch g and acoustics loads, remains to be accomplished. In addition, there are issues associated with propellant transfer and zero-g gauging. These, however, can be avoided for early lunar systems by proper choice of configuration and operations, e.g. the tandem-direct system recommended elsewhere in this report. This presents the opportunity to evolve these technologies with operations of initial flight systems.

Engines - There is little risk of being able to provide some sort of cryogenic engine for lunar and Mars missions. The RL- 10 could be modified to serve with little risk; deep throttling of this engine has already been demontrated on the test stand. The risk of developing more advanced engines is also minimal. An advanced development program in this area serves mainly to reduce development cost by pioneering thecritical features prior to full-scale development.

Aerocapture and aerobraking - There are six potential functions, given here in approximate ascending order of development risk: aero-descent and landing of crew capsules returning from the Moon, aerocapture to low Earth orbit of returning reusable lunar vehicles, landing of Mars excursion vehicles from Mars orbit, aero descent and landing of crew capsules returning from Mars, aerocapture to low Earth orbit of returning Mars vehicles, and

Program Year	Total Development	Total Production	Operation & Support	Program SE&I	Total Program	Cumulative Program
90	0.0	0.0	0.0	0.0	0.0	0.0
91	137.5	0.0	27.5	16.5	181.5	181.5
92	518.1	10.0	105.6	63.4	697.1	878.6
93	1145.0	10.0	231.0	138.6	1524.6	2403.2
94	1923.6	20.0	388.7	233.2	2565.6	4968.7
95	2652.4	50.0	540.5	324.3	3567.2	8535.9
96	3537.1	50.0	717.4	430.5	4735.0	13271.0
97	3962.3	50.0	802.5	481.5	5296.2	18567.2
98	3943.3	110.0	810.7	486.4	5350.4	23917.6
99	3255.6	0.0	651.1	390.7	4297.4	28214.9
00	2783.6	3405.6	1237.9	742.7	8169.8	36384.7
01	3036.6	3843.6	1376.0	825.6	9081.9	45466.6
02	4396.1	3943.6	1667.9	1000.8	11008.4	56475.0
03	5611.2	4724.6	2067.2	1240.3	13643.3	70118.3
04	5992.6	4319.8	2062.5	1237.5	13612.4	83730.7
05	6398.5	4316.5	2143.0	1285.8	14143.9	97874.5
06	4957.1	6435.9	2278.6	1367.2	15038.8	112913.4
07	3968.2	8996.0	2592.9	1555.7	17112.8	130026.2
08	3956.3	8880.8	2567.4	1540.4	16944.9	146971.1
09	3313.0	10910.2	2844.7	1706.8	18774.7	165745.8
10	772.2	9597.1	2073.9	1244.3	13687.5	179433.3
11	1041.8	9153.4	2039.0	1223.4	13457.7	192891.0
12	1245.8	9152.5	2079.7	1247.8	13725.8	206616.8
13	1340.5	9151.7	2098.4	1259.1	13849.7	220466.5
14	1274.9	6966.7	1648.3	989.0	10879.0	231345.4
15	961.7	10355.3	2263.4	1358.0	14938.3	246283.8
16	0.0	10354.6	2070.9	1242.6	13668.1	259951.9
17	0.0	9632.5	1926.5	1155.9	12715.0	272666.9
18	0.0	10605.1	2121.0	1272.6	13998.8	286665.6
19	0.0	10604.6	2120.9	1272.6	13998.1	300663.7
20	0.0	10604.2	2120.8	1272.5	13997.5	314661.2
21	0.0	8990.1	1798.0	1078.8	11866.9	326528.1
22	0.0	8989.7	1797.9	1078.8	11866.3	338394.5
23	0.0	8989.3	1797.9	1078.7	11865.8	350260.3
24	0.0	8988.9	1797.8	1078.7	11865.3	362125.6
25	0.0	5096.0	1019.2	611.5	6726.7	368852.4

LIFE CYCLE COST FOR INDUSTRIALIZATION/SETTLEMENTPROGRAM

FIGURE 9-10

MISSION FUNCTION	BRAKE SIZE	ATMOSPHERE KNOWLEDGE & UNCERTAINTY	TARGET FOR ENTRY: GN&C PRECISION	HEATING/TPS	AERO PASS GN&C PRECISION REQUIRED
Lunar return Earth landing	Small, no ass'y required	Accurate knowledge, low uncert. effect	Very high	State-of-the-Art	State-of-the-Art
Lunar return Earth landing	Moderate requires assembly	Accurate knowledge, high uncert. effect	Very high	State-of-the-Art	Believed State-of-the-Art
Mars landing from orbit	Large, requires assembly	Poor knowledge, low uncert. effect	Can be high, e.g. done from Mars orbit	State-of-the-Art	Believed State-of-the-Art
Mars return Earth landing	Small, no ass'y required	Accurate knowledge, moderate uncertainty effect	Very high	Very high heating rates, TPS advancement needed	Believed State-of-the-Art
Mars return aerocapture	Large, requires assembly	Accurate knowledge, high uncert. effect	Very high	Very high heating rates, TPS advancement needed	Believed State-of-the-Art
Mars return aerocapture	Large, requires assembly	Poor knowledge, high uncert. effect	Poor, unless nav-aids in Mars orbit	High heating rates, some TPS advancement needed	Advancements required

DEVELOPMENT RISK ASSESSMENT FOR AEROBRAKING BY FUNCTION

FIGURE 9-11

aerocapture to Mars orbit of Mars excursion and Mars transfer vehicles. Figure 9-11 provides a qualitative development risk comparison for these six functions.

Aerocapture of vehicles requires large aerobrakes. For these to be efficient, low mass per unit area is required, demanding efficient structures made from very high performance materials as well as efficient, low mass thermal protection materials. By comparison, the crew capsules benefit much less from high performance structures and TPS.

Launch packaging and on-orbit assembly of large aerobrakes presents a significant development risk that has not yet been solved even in a conceptual design sense. Existing concepts package poorly or are difficult to assemble or both. While the design challenge can probably be met, aerobrake assembly is a difficult design and development challenge, representing an important area of risk.

Nuclear thermal rockets - The basic technology of nuclear thermal rockets was developed and demonstrated during the 1960s and early 1970s. The development risk to reproduce this technology is minimal, except in testing as described below. Current studies are recommending advances in engine performance, both in specific impulse (higher reactor temperature) and in thrust-to-weight ratio (higher reactor power density). The risks in achieving these are modest inasmuch as performance targets can be adjusted to technology performance.

Reactor and engine tests during the 1960s jetted hot, slightly radioactive hydrogen directly into the atmosphere. Stricter environmental controls since that time prohibit discharge of nuclear engine effluent into the atmosphere. Design and development of full containment test facilities presents a greater development risk than obtaining the needed performance from nuclear reactors and engines. Full- containment facilities will be required to contain all the hydrogen effluent, presumably oxidize it to water, and remove the radioactivity.

Electric Propulsion Power Management and Thrusters - Power management and thrusters are common to any electric propulsion power source (nuclear, solar, or beamed power). Unique power management development needs for electric propulsion are (1) minimum mass and long life, (2) high power compared to space experience, i.e. megawatts instead of kilowatts, (3) fast arc suppression for protection of thrusters. Minimizing mass of power distribution leads to high distribution voltage and potential problems with plasma losses, arcing, and EMI. Thus while power management is a mature technology, the unique

requirements of electric propulsion introduce a number of development risks beyond those usually experienced in space power systems.

Electric thruster technology has been under development since the beginning of the space program. Small thrusters are now operational, such as the resistance-heat-augmented hydrazine thrusters on certain communications spacecraft. Small arc and ion thrusters are nearing operational use for satellite stationkeeping.

Space transfer demands on electric propulsion performance place a premium on high power in the jet per unit mass of electric propulsion system. This in turn places a premium on thruster efficiency; power in the jet, not electrical power, propels spaceships. Space transfer electric propulsion also requires specific impulse in the range 5000 to 10,000 seconds. Only ion thrusters and magnetoplasmadynamic (MPD) arc thrusters can deliver this performance. Ion thrusters have acceptable efficiency but relatively low power per unit of ion beam emitting area. MPD thruster technology can deliver the needed Isp with high power per thruster, but has not yet reached efficiencies of interest. Circular ion thrusters have been built up to 50 cm diameter, with spherical segment ion beam grids. These can absorb on the order of 50 kWe each. A 10 MWe system would need 200 operating thrusters. The development alternatives all have significant risk: (1) Advance the state of the art of MPD thrusters to achieve high efficiency; (2) Develop propulsion systems with large numbers of thrusters and control systems; or (3) Advance the state of the art of ion thrusters to much larger size per thruster.

Nuclear power for electric propulsion - Space power reactor technology now under development (SP-100) may be adequate; needed advances are modest. Advanced power conversion systems are required to obtain power-to-mass ratios of interest. The SP-100 baseline is thermoelectric, which has no hope of meeting propulsion system performance needs. The most likely candidates are the closed Brayton (gas) cycle and the potassium Rankine (liquid/vapor) cycle. (Potassium provides the best match of liquid/vapor fluid properties to desired cycle temperatures.) Stirling cycle, thermionics, and a high-temeperature thermally-driven fuel cell are possibilities. The basic technology for Brayton and Rankine cycles are mature; both are in widespread industrial use. Prototype space power Brayton and Rankine turbines have run successfully for thousands of hours in laboratories. The development risk here is that these are very complex systems; there is no experience base for coupling a space power reactor to a dynamic power conversion cycle; there is no space power experience base at the power levels needed; and these systems, at

power levels of interest for SEI space transfer application, are large enough to require in-space assembly and checkout. Space welding will be required for fluid systems assembly.

Solar power for space transfer propulsion - Solar power systems for space propulsion must attain much higher power-to-mass ratios than heretofore achieved. This implies a combination of advanced solar cells, probably multi-band-gap, and lightweight structural support systems. Required array areas are very large. Low-cost arrays, e.g. $100/watt, are necessary for affordable system costs, and automated construction of the large area structures, arrays, and power distribution systems appears also necessary. Where the nuclear electric systems are high development risk because of complexity and the lack of experience base at relevant power levels and with the space power conversion technologies, most of the solar power risk appears as technology advancement risk. If the technology advancements can be demonstrated, development risk appears moderate.

Avionics and software - Avionics and software requirements for space transfer systems are generally within the state of the art. New capability needs are mainly in the area of vehicle and subsystem health monitoring. This is in part an integration problem, but new technoques such as expert and neural systems are likely to play an important role.

An important factor in avionics and software development is that several vehicle elements having similar requirements will be developed, some concurrently. A major reduction in cost and integration risk for avoinics can be achieved by advanced development of a "standard" avionics and software suite, from which all vehicle elements would depart.

Further significant cost savings are expected from advancements in software development methods and environments.

Environmental Control and Life Support (ECLS) - The main development risk in ECLS is for the Mars transfer habitat system. Other SEI space transfer systems have short enough operating durations that shuttle and Space Station Freedom ECLS system derivatives will be adequate. The Mars transfer requirement is for a highly closed physico-chemical system capable of 3 years' safe and dependable operation without resupply from Earth. The development risk arises from the necessity to demonstrate long life operation with high confidence; this may be expensive in cost and development schedule.

9.6.2 Man-Rating Approach

Man-rating includes three elements: (1) Design of systems to manned flight failure tolerance standards, (2) Qualification of subsystems according to normal man-rating requirements, and (3) Flight demonstration of critical performance capabilities and functions prior to placing crews at risk. Several briefing charts follow: the first summarizes a recommended approach and lists the subsystems and elements for which man-rating is needed; subsequent charts present recommended man-rating plans.

9.6.3 Mission and Operations Risk

These risk categories include Earth launch, space assembly and orbitasl launch, launch windows, mission risk, and mitigation of ionizing radiation and zero-g risks.

Earth launch - The Earth launch risk to in-space transportation is the risk of losing a payload because of a launch failure. Assembly sequences are arranged to minimize the impact of a loss, and schedules include allowances for one make-up launch each mission opportunity.

Assembly and Orbital Launch Operations - Four sub-areas are covered: assembly, test and on-orbit checkout, debris, and inadvertent re-entry.

Assembly operations risk is reduced by verifying interfaces on the ground prior to launch of elements. Assembly operations equipment such as robot arms and manipulators will undergo space testing at the node to qualify critical capabilities and performance prior to initiating assembly operations on an actual vehicle.

Assembly risk varies widely with space transfer technology. Nuclear thermal rocket vehicles appear to pose minimum assembly risk; cryo/aerobraking are intermediate, and nuclear and solar electric systems pose the highest risk.

Test and on-orbit checkout must deal with consequences of test failures and equipment failures. This risk is difficult to quantify with the present state of knowledge. Indications are: (1) large space transfer systems will experience several failures or anomalies per day. Dealing with failures and anomalies must be a routine, not exceptional, part of the operations or the operations will not be able to launch space transfer systems from orbit; (2)

vehicles must have highly capable self-test systems and must be designed for repair, remove and replace by robotics where possible and for ease of repair by people where robotics cannot do the job; (3) test and on-orbit checkout will run concurrently with propellant loading and launch countdowns. These cannot take place on Space Station Freedom. Since the most difficult part of the assembly, test and checkout job must take place off Space Station Freedom the rest of the job probably should also.

Orbital debris presents risk to on-orbit operations. Probabilities of collision are large for SEI-class space transfer systems in low Earth orbit for typical durations of a year or more. Shielding is mandatory. The shielding should be designed to be removed before orbital launch and used again on the next assembly project.

Creation of debris must also be dealt with. This means that (1) debris shielding should be designed to minimize creation of additional debris, especially particles of dangerous size, and (2) operations need to be rigorously controlled to preventinadvertent loss of tools and equipment that will become a debris hazard.

Inadvertent re-entry is a low but possible risk. Some of the systems, especially electric propulsion systems, can have very low ballistic coefficient and therefore rapid orbital decay rate. Any of the SEI space transfer systems will have moderately low ballistic coefficient when not loaded with propellant. While design details are not far enough along to make a quantitative assessment, parts of these vehicles would probably survive reentry to become ground impact hazards in case of inadvertent reentry. For nuclear systems, it will be necessary to provide special support systems and infrastructure to drive the probability of inadvertent reentry to extremely low levels.

Launch Windows - Launch windows for single-burn high-thrust departures from low Earth orbit are no more than a few days because regression of the parking orbit line of nodes causes relatively rapid misalignment of the orbit plane and departure vector. For lunar missions, windows recur at about 9-day intervals.

For Mars, the recurrence is less frequent, and the interplanetary window only lasts 30 to 60 days. It is important to enable Mars launch from orbit during the entire interplanetary window. Three-impulse Mars departures make this possible; a plane change at apogee of the intermediate parking orbit provides alignment with the departure vector. Further

analysis of the three-burn scheme is needed to assess penalties and identify circumstances where it does not work.

Launch window problems are generally minimal for low-thrust (electric propulsion) systems.

Mission Risk - Comparative mission risk was analyzed by building risk trees and performing semi-quantitative analysis. The next chart presents a comparison of several mission modes; after that are the risk trees for these modes.

Ionizing Radiations and Zero G - The threat from ionizing radiations is presented elsewhere in this document. Presented here are the mitigating strategies for ionizing radiations and zero g.

Nuclear systems operations present little risk to flight crews. Studies by University of Texas at Austin showed that radiation dose to a space station crew from departing nuclear vehicles is very small provided that sensible launch and flight strategies are used. On-board crews are protected by suitable shieding and by arrangement of the vehicle, i.e. hardware and propellant between reactors and the crew and adequate separation distances. After nuclear engines are shut off, radiation levels drop rapidly so that maneuvers such as departure or return of a Mars excursion vehicle are not a problem. On-orbit operations around a returned nuclear vehicle are deferred until a month or two after shutdown, by which time radioactivity of the engine is greatly reduced.

Reactor disposal has not been completely studied. Options include solar system escape and parking in stable heliocentric orbits between Earth and Venus.

Crew radiation dose abatement employs "storm shelters" for solar flares, and either added shielding of the entire vehicle or fast transfers (or both) to reduce galactic cosmic ray exposure. Assessments are in progress; tradeoffs of shielding versus fast trips have yet to be completed. Expected impact for lunar missions is negligible and for Mars missions, modest.

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10. CONCLUDING REMARKS

The real SEI program will not follow any particular long-range scenario that can be defined now. Clearly, an ambitious program can start small and evolve. The potential for evolution to a larger scale must be included in technology selection decisions for small and moderate-scale programs. Programs of different scale at the Moon and Mars are logical. Lunar and Mars exploration will probably start with modest objectives and evolve as a result of early mission achievements and scientific findings. The recommended SEI program strategy is evolutionary and keeps options open.

During the course of this study, we identified and examined four basic propulsion architectures: cryogenic aerobraking, nuclear thermal rocket, nuclear electric, and solar electric, as well as three additional architectures, based upon variations of the cryogenic aerobraking architecture, namely, L-2 based Lunar oxygen, Mars direct, and a family of cycler profiles. Each of these architectures fulfill specific mission requirements; however, none of them individually can satisfy the requirements of all missions. Technical evaluation and system trade studies were conducted on the systems included within the architecture framework. System level studies quantified (from a conceptual design level) the entire vehicle's layout, performance, mass, and operating characteristics to fulfill specific missions. In addition, the physical and operational characteristics were assessed for vehicle systems; and in some cases, also its subsystems. These initial studies provide the basis for further, detailed, conceptual designs for the more promising concepts as well as technology advancements to better define their technical performance capabilities, risks, operational problems, and development and operations costs. Before final selection between these architectures is made, we need to define the nature and activity level of the Mars exploration and development program that will be a part of the Space Exploration Initiative.

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APPENDIX A

11. SPECIAL STUDIES - PILOTED ROVERS PARADIGMS

The purpose of the study was to determine technology advancements required to develop piloted lunar rovers. The rovers would be used as utility vehicles in establishing a lunar surface habitation facility and exploration base. The rovers would also be used to support exploration missions ranging out to 1000 kilometers radius from the base.

This study is intended to provide reference data that may be used for: planning technology development and options; identifying technologies required for planned and potential piloted rover uses; assessing on-going and planned technology development programs; and planning piloted rover missions. It provides a state-of-the-art assessment of existing relevant technologies and those technologies that are now in development and will be available for piloted rover development. It also identifies technology advancements required; identifies major test facilities required; and provides some cost and schedule estimates for advancing the technology to a readiness level of 5. The details of the investigation are provided in Reference 42.

11.1 Background

Twenty years has elapsed since the start of design and development of the Lunar Roving Vehicle used on Apollo Missions 15, 16, and 17. Since Apollo, relatively little has been done to use the operational data obtained on Apollo to advance the state-of-the art capability for building and operating piloted roving vehicles for lunar exploration.

The lunar surface operating environment and lunar terrain are better defined today because of data obtained and evaluated from the Apollo flights. However, since that time, the emphasis of space research and development has been on building the National Space Transportation System, orbiting space vehicles, and performing unmanned space exploration. Therefore, the plans for developing piloted planetary rovers was delayed to permit funding and maturation of higher priority near-term programs.

Since the President established a manned space exploration initiative in July 1989, requirements for lunar/Mars explorations have begun to emerge. This study, following NASA's in-house 90-Day Study (Reference 5), is the forerunner to developing the required technologies necessary to meet these new and evolving requirements.

11.2 Approach

The study specifically addresses assessment of technology requirements unique to piloted rovers in support of a human-habitable lunar base development and operations. Two types of rover vehicles were assessed:

a. A light, unpressurized utility vehicle to provide personnel and equipment transportation and to function as a prime mover for power tools and implements during base preparation and operation.

b. A medium range pressurized exploration rover vehicle to support surface exploration missions within ~1000 kilometer radius of the lunar base.

Study activities are summarized as follows:

a. Data collection.

b. Engineering design concept definitions.

c. Design trade studies and evaluations.

d. Definition of design, development and test programs to advance technology to the required level to support the on-going lunar base planning.

e. Preparation of cost and schedule estimates for the required technology advancements.

Technology discussions are focused on on-going development programs, proposed enhancements, goals, objectives, and schedules for research and technology development by NASA, other government agencies, and the private sector.

Technology readiness levels, as defined in Figure A-1 are used to define the level of development being discussed. Level 5 should be achieved prior to the start of program development Phase B effort, or be worked concurrent with Phase B. Unless otherwise specified, recommended technology advancements are required to achieve a readiness level of 5 prior to the start of Phase B activities. Phase B systems development for human habitation and exploration of the moon is currently scheduled to start in 1993. It is assumed that these technology advancements will be accomplished in Advanced Development. Required technology advancements not achieved by start of Phase B are assumed to be worked concurrent with Phase B in Advanced Development, or in other on-going technology development programs that address the technology need in question.

Level 1	Basic Principles Observed and Reported
Level 2	Conceptual Design Formulated
Level 3	Conceptual Design Tested Analytically or Experimentally
Level 4	Critical Function/Characteristic Demonstrated
Level 5	Component/Breadboard Tested in Relevant Environment
Level 6	Prototype/Engineering Model Tested in Relevant Environment
Level 7	Engineering Model Tested in Space

TECHNOLOGY READINESS LEVELS
FIGURE A-1

Information herein has been assembled from a variety of sources, including published reports, literature searches, briefing documents, personal communications with individuals at NASA centers and industry experts, and evaluations of applicable technologies. Use was made of the data available from the Apollo Program's Lunar Roving Vehicle (LRV) where applicable.

11.3 Summary of Findings

Following a state-of-the-art survey of current capabilities and technologies related to lunar rovers, technology advancement needs were identified in the following areas:

1. Vehicle Mobility
2. Wheels
3. Drive Systems
4. Lubricants and Seals
5. Suspension
6. Attachments & Implements
7. Environmental Control and Life Support
8. Electrical Power
9. Thermal Control
10. Man Systems
11. Structures and Mechanisms
12. Radiation Protection
13. Navigation
14. Communications

Technology development needs as determined in a study of rover mobility and life inhibiting factors are presented in Figure A-2. Most of the efforts address a need for developing methods of increasing life expectancy of materials and mechanisms exposed to the harsh environmental constituents of the moon including dust, hard vacuum, and thermal cycling. Other efforts proposed include advanced development activity for ensuring that state-of-the-art hardware can survive the induced environments within a pressurized rover. In addition, an effort is proposed which would develop a computerized mobility modeling capability for evaluating lunar surface vehicles. A brief statement of the scope for each effort is also included. Most of the recommended technology developments are termed enabling when considered together with the preliminary rover requirements set forth in the Planetary Surface Systems Requirements Document (PSSRD). However, the need for continued mission studies and requirements definition tasks is emphasized.

System/ Subsystem	Recommended Technology Development	Scope
Piloted Vehicle	Adapt AMM/CAMMS mobility models for lunar environment	• Terrain mapping and code mods. • Model testing verification
Wheels	Develop and test three wheel concepts: • Wire mesh wheel • Loop Spring wheel • Convolute cone wheel	• Engineering analyses & trades • Develop and fab. test equipment • Conduct tests • Recommendations / design specs
Drive Systems	Develop and test two drive motors and two gear reducer concepts: • Brushless dc motor • AC induction motor • Planetary gear reducers • Harmonic drive	• Engineering analyses & trades • Develop and fab. test equipment • Conduct tests • Recommendations / design specs
Lubricants & Seals	Develop specific technologies related to mobility subsystem components: • Surface treatments • Bearing materials • Transfer film lubricants • Solid, thin-film lubricants • Space stable polymers and elastomers • Brake materials	• Perform applied research concurrent with advanced development of mechanisms. • Demonstrate component performancein representative environment under operational conditions

System/ Subsystem	Recommended Technology Developments	Scope
Shock Dampers	Develop and test four damper concepts: • Electromagnetic (passive and hybrid system) • Friction / Coulomb (passive and hybrid system)	• Engineering analyses & trades • Develop and fab. test equipment • Conduct tests • Recommendations / design specs
Attachments/ Implements	Develop and test prototypes of: • Wire cable / rope materials and coatings • Belt and chain materials and lubricants • Control and I/F commonality & concept devel. • Dev & eval. of electromech and magnetic actuators • Implement modeling (PUSH-IT) development & ver.	• Test hardware design • Materials anal, selection, & tests • Develop actuator test articles • Develop and perform tests • Code mods & soil bin testing
Environmental Control / Life Support (ECLSS)	Develop test articles and conduct tests for: • Equipment durability assessment • Fire suppression techniques • Reduced cabin pressure operation	• Test equip. for induced vib. env't • Demonstrate fire supp cleanup & compatibility with ECLS for rover • Modify & test h/w for reduced press
Electrical Power	• Refine composite, high press. reactant storage tanks • Qualify high efficiency solar cells (GaAs/CIS) • Develop alt. pwr resupply techniques with alt missions	• Engineering analyses & design • Development & Tests • Recommendations / design specs.
Thermal Control	Develop prototype hardware and test for: • Lunar dust, vibration, and shock effects investigation • Selective field-of-view radiator performance	• M&P research and development • Component development & tests • Design options / recommendations

TECHNOLOGY DEVELOPMENT RECOMMENDATIONS

FIGURE A-2

System/Subsystem	Recommended Technology Developments	Scope
Man Systems	Crew accommodations for 1/6th g	• Analyze, model, & simulate
Structures and Mechanisms	Develop methods for applying composite materials: • Repairability • Radiation protection capabilities • Vibration dampening • Low thermal expansion/degradation properties	• Materials analysis & selection • Repairability research • Radiation computations • Fab & conduct tests • Make recomendations/design specs
Radiation Protection	Develop lightweight concepts for: • Innovative protection methods • Reductions of uncertainties in rad. env't assessments • Dev of radiation monitoring system for SPE warning • Develop shielding analysis codes (< 25% uncertainty)	• Engineering analyses & trades • Modeling & testing • Develop & fabricate test equipment • Conduct tests • Recommendations / design specs.
Navigation	Develop long-range precision lunar nav. technologies: • Use of Earth and/or lunar-based GPS • Inertial nav systems w/ solid state sensors • Failure detection, isolation, estim., and recovery	• Engineering analyses & trades • Simulations • Recommendations / design specs
Communication	Develop long-range high-speed voice & video comm: • Phased array antenna for cont. link(x,k bands) • Lunar relay satellite • More efficient power amplifiers	• Engineering analyses & trades • Recommendations / design specs

System/ Subsystem	Recommended Technology Developments	Scope
EVA	Develop techniques for advancing EVA capability: • Airlock operation: low power pumps, atm. recovery • Dust control • Suit maintenance & recharge facilities • Develop suit/rover integration & ergonomics concepts • External power interfaces	• Dev & tests w/ low power emphasis • Engineering analyses & trades • Develop & fab test equipment • Conduct tests • Recommendations / design specs
Finishes & Coatings	Develop and evaluate efficient long-life: • Glass (Windows) • " (Solar arrays) • Radiator finishes and thermal coatings	• Engineering analyses & trades • Conduct tests • Recommendations / design specs

TECHNOLOGY DEVELOPMENT RECOMMENDATIONS (CONTINUED)

FIGURE A-2

The recommended technology advancement will require test verification. A list of tests for verifying that each system or component technology reaches a technology readiness level of 5 is provided in Figure A-3. The parameters which are critical to rover performance and availability are shown with their dependence on the lunar environment variables of vacuum, thermal cycling, and dust. Control variables are also shown providing an exhaustive list of the tests and simulations necessary for evaluating candidate system and/or component technologies.

		Simulated env't conditions*			
System/Component	Parameter(s) Measured	Vacuum?	Thermal?	Dust?	Test variables
Wheels • convolute cone • loop spring • wire mesh	• Slip (Programmed Increasing Slip Test)	No	No	Yes	Velocity acceleration yaw angle slope load
	• Dust accumulation	Yes	No	Yes	
	• Steering Resistance	No	No	Yes	
	• Sinkage	No	No	Yes	
	• Fatigue life	Yes	Yes	Yes	
	• Deflection	No	Yes	No	
Drive Systems (1st - motor alone) • dc brushless • ac induction	• Torque - Dynamometer	No	No	No	Current Voltage Reduction ratios
	• Efficiency - Dynamometer	No	No	No	
	• Operation / surface temperatures	Yes	Yes	No	
Suspension • electromagnetic • friction • pass & semi-act	• Energy dissipation	No	Yes	No	Current Voltage load loading cycles
	• Reliability / Life cycle (MTBF)	Yes	Yes	Yes	
	• Efficiency	No	No	No	
	• Regenerative power capabilities	No	No	No	
	• Operation / surface temperatures	Yes	Yes	No	
Integrated mobility subsystem (M/4)	• Drawbar pull	No	No	Yes	Power inputs velocity yaw angle slope load acceleration
	• Dynamic frequencies & amplitudes	No	No	No	
	• Wheel slip	No	No	Yes	
	• Power / efficiency	No	No	Yes	
	• Energy dissipation	No	No	Yes	
	• EMI/EMC	No	No	Yes	
	• Stress	No	Yes	No	

* = gravity simulated with applied loads, etc.

TECHNOLOGY DEVELOPMENT TESTS
FIGURE A-3

System/Component	Parameter(s) Measured	Simulated env't conditions*			Test variables
		Vacuum?	Thermal?	Dust?	
ECLSS	• Operating pressure	Yes	Yes	No	• Gas selection • Stab. of bio waste
	• Air purity	No	No	Yes	
	• Waste management	No	Yes	No	
	• Fire suppression	No	No	No	
Electrical Power	• Fuel cell efficiency	Yes	Yes	No	• Reactant parameters • Cell types
	• Solar cell efficiency	Yes	Yes	Yes	
	• Heat rejection efficiency	No	Yes	Yes	
Crew Station	• Visibilty range	No	No	Yes	• Lighting angles • Sun angles • Terrain parameters
	• Command / Control	No	No	No	
	• Glare control effectiveness	No	No	No	
Integrated Mobility Test Article (IMTA) • Mobility system and chassis with weight and c.g. of other systems simulated	Vehicle dynamics • maximum speed	No	No	Yes	Terrain parameters • crevasse size • obstacle geometry • surface roughness • elevations
	• ride quality (avg. absorbed power)	No	No	Yes	
	• Roll and pitch stability	No	No	Yes	
	• Crevasse crossing capability	No	No	Yes	
	• Obstacle avoidance	No	No	Yes	
	• Minimum clearance	No	No	Yes	

* = gravity simulated with applied loads, etc.

System/Component	Parameter(s) Measured	Simulated env't conditions*			Test variables
		Vacuum?	Thermal?	Dust?	
Attachments/ Implements • cable materials • belt materials	• Lubricant life (metallic mat'ls.)	Yes	Yes	Yes	Temperature bend radius load velocity dust particle size
	• Fatigue life (metallic mat'ls.)	Yes	Yes	Yes	
	• Wear rate (non-metallic)	Yes	Yes	Yes	
	• Abrasion resistance (non-metallic)	Yes	Yes	Yes	
	• Low temperature flexibility (cold-bend) (all mat'ls)	No	Yes	No	
	• Outgassing in vacuum (non-metallic)	No	No	No	
• implement models	• work rates	No	No	Yes	
	• tractive force required	No	No	Yes	
Data / Communications	none required (technologies are > level 5)				Level of detail used to build digital data base.
Navigation	• Digitized terrain display as compared to actual terrain.				
Chassis	• Stress (static load test)	No	Yes	No	Payload weight

* = gravity simulated with applied loads, etc.

TECHNOLOGY DEVELOPMENT TESTS (CONTINUED)

FIGURE A-3

APPENDIX B

12. SUPPORT TO OFFICE OF EXPLORATION ACTIVITIES

Support was provided to MSFC at the request of the contracting officer's technical representative on a wide range of topic areas. General areas covered are as follows:

- Assess current design and develop alternate concepts where appropriate.
- Provide support to OEXP controlled trade studies.
- Deepen concepts definition in preparation for OEXP FY '90 study cycles.
- Perform parametric and sensitivity trades which cannot be accommodated during cycle II.
- Develop data supporting recommendation for FY '90 case studies.
- Take broader, more relaxed look at Mars/Lunar transportation requirements and concepts by relaxing study requirement document constraints.
- Initiate broad propulsion trade studies, comparing a range of potentially applicable concepts, including chemical with/without aerobrakes, NEP, SEP, NTR, and mixed.
- Provide support to development of a technology database for OEXP missions.

Specific issues addressed during the course of the contract are summarized below in chronological order. The detailed results may be found in References 6 to 12.

- Support to NASA analyzing SEI; Mars outpost briefing, vehicle mass and volume statements.
- Reusable MEV with Mars in-situ oxygen.
- Manifesting, mission, and assembly analyses.
- Aerobrake stagnation point heating.
- Option V configuration analysis, full-up Mars mission vehicle.
- Support for 90 Day Study.
- MSFC in-house briefing on Mars concepts.
- Material for advanced automation and robotics needs.
- Established new reference NTR.

- Draft SEI baseline description
- Evaluation of Level III tradesand advanced transportation options.
- NTR and chem-aerobrake configuration and mass statements.
- Lunar-based SSF commonality.
- Non-quantitative ECLSS weighting factors.
- Support to MASE Operations Integration working group.
- Support for aerobrake technology workshop.
- Preparation of SSF/SEI habitat commonality Statement of Work.
- Generation of HLLV manifest data.
- Presentation to JSC Human and Robotics group
- Artificial gravity for NTR conjunction class missions.
- Tutorial for MSFC prior to Stafford Synthesis group visit.
- PDR presentation material.
- Transportation system briefing material for Stafford committee.
- Fast trip analysis.
- Advanced propulsion
- Thermal protection system.
- Radiation dose rates.
- Advanced propulsion concepts for 500-day, opposition Mars mission.
- Split mission analysis.
- Support NASA synthesis group radiation assessment meeting.
- Fast trip, high delta velocity trade data.