JOHNSON
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NASA

3rd
SEI TECHNICAL
INTERCHANGE
Proceedings

Sponsored by
the

EXPLORATION
PROGRAMS OFFICE

and the
Univ. of Houston
Clear Lake

3rd Technical Interchange Meeting
May 5 & 6 1992

Coordinated by Doug Peterson
(713)283-5556

National Aeronautics and
Space Administration

NASA

Lyndon B. Johnson Space Center
Houston, Texas
77058

Reply to Attn of   XA

Dear Colleague:

Thank you for your participation in the 3rd SEI Technical Interchange held May 5 & 6 in Houston. We continue to be very pleased with these meetings and your feedback suggests that you also feel the interchange is valuable. SEI and NASA management as well as those in Congress see this type of "outreach" activity as an important part of the overall planning, analysis and decision making process for future programs. The whole spectrum of new concepts, alternatives and options must be identified and reviewed to enable the critical decisions required to undertake challenging SEI goals.

This meeting with over 300 participants from about 100 different organizations brings together the dispersed supporters of SEI to begin building the team necessary for the long-term. The interaction between industry, academia, and government organizations including DoE, DoD, and NASA provides the focus for working closer together on SEI related activities.

This book of proceedings should capture the "black and white" of the meeting to provide a record for on-going activities. Hopefully the material within will be as valuable to you as it is to those in the Exploration Programs Office and other SEI organizations.

Thanks again for your participation.

Sincerely,

Douglas R. Cooke
Manager, Exploration Programs Office

3rd
Space Exploration InitiativeTechnical Interchange

Hosted by the
Exploration Programs Office

NASA

EXPLORATION
PROGRAMS OFFICE

Tuesday, May 5, 1992

8:30 a.m. **Welcome**
Dr. James Lester, Interim Dean, School of Natural and Applied Sciences, UH-CL
Meeting Objectives and Near-Term Activities
Doug Cooke, Manager, Exploration Programs Office (ExPO)

9:00 a.m. **SEI Master Plan**
Dwayne Weary, ExPO 1 *S1*

9:15 a.m. **Lunar Resource Mapper**
Mike Conley, ExPO 9 *S2*

9:30 a.m. **Artemis, the common lunar lander, an update**
Steve Bailey, New Initiatives Office, JSC 16 *S3*

9:45 a.m. **Science Themes for Early Robotic Missions: LPI Workshops**
Paul Spudis, Lunar and Planetary Institute 40 *S4*

10:00 a.m. **Break**

10:20 a.m. **Review of Rover/Mobility Systems Workshop**
Dave Weaver, ExPO 49 *S5*

10:35 a.m. **FLO Mission Overview**
B. Kent Joosten, ExPO 59 *S6*

11:05 a.m. **FLO Science and Payloads**
Dave McKay, Solar System Exploration Div, JSC 75 *S7*

11:35 a.m. **FLO Space Transportation Concepts and Issues**
Ron Kahl, New Initiatives Office, JSC 116 *S8*

12:00 p.m. **Lunch**

1:00 p.m. **The Space Exploration Initiative** *— O*
Mike Griffin, AssociateAdministrator for Exploration

2:00 p.m. **FLO Surface Systems Concepts and Issues**
John Connolly, New Initiatives Office, JSC —O

2:20 p.m. **FLO Lunar Habitat Concepts and Issues**
Molly Elrod, Marshall Space Flight Center 139 *S9*

2:40 p.m. **FLO Earth-to-Orbit Concepts and Issues**
Gene Austin, Marshall Space Flight Center 150 *S10*

3:00 p.m. **Break**

3:15 p.m. **First Series of Short Technical Presentations**

5:00 p.m. **Adjourn**

Wednesday, May 6, 1992

8:00 a.m. **Second Series of Short Technical Subject Presentations, followed by splinter sessions on FLO mission overview, surface systems, space transportation, risk, science, habitats, and Artemis**

9:30 a.m. **Break**

9:50 a.m. **Organizational Change: Incentives and Resistance**
Dr. Peter Bishop, UH-CL, Studies of the Future 161 *S11*

10:10 a.m. **Getting What You Want: Technology to Overcome the Limits of Organizational Control**
David Peterson, Ventana Systems, Inc.

10:40 a.m. **DOD Acquisition Streamlining Initiatives**
Jesse Stewart, Defense Systems Management College 178 *S12*

11:00 a.m. **SEI Management and Implementation**
Hum Mandell, ExPO 206 *O*

11:30 a.m. **Streamlined NASA Acquisition**
Gene Easley, Director of Procurement, JSC

12:00 p.m. **Lunch**

1:00 p.m. **Effecting Workforce/Cultural Change**
Richard Grant, Boeing 207 *O*

1:30 p.m. **NASA and Industrial Project Management**
A. Guastaferro, Lockheed 212 *S13*

2:00 p.m. **Skunk Works Type Approach for F-SAT**
Gary Turner, Lockheed 228 *S4*

2:30 p.m. **Project Management Lessons Learned on SDIO's Delta Star & Single Stage Rocket Technology Programs**
Paul Klevatt, McDonnell Douglas, SSRT Program 239 *S15*

3:00 p.m. **MTS Report on New Ways of Doing Business**
Bruce McCandless II, Martin Marietta

3:30 p.m. **Break**

4:00 p.m. **Science and Payloads Session Report**
Session Leads

4:15 p.m. **Surface Systems Session Report**
Session Leads

4:25 p.m. **Lunar Habitats Session Report**
Session Leads

4:35 p.m. **Space Transportation Session Report**
Session Leads

CONTENTS - continued

Main Auditorium

First Series of 5 min. Tech Briefings Tuesday 3:30 pm to 5:15 pm

Second Series of 5 min. Tech Briefings Wednesday 8:00 am to 9:30 am

6. Dana Andrews   Boeing   498 532
    common modules for SEI, ACRV, & LPLS

7. John Hodge   Martin Marietta   507 533
    multi-stage lander; storable vs. cryo

8. Dave Plachta   LeRC   522 534
    benefits of cryo vs. storable for return

9. Larry Schooley   Sp Eng. Res Ctr   540 535
    self contained module materials processing

10. Donald Curry   JSC/ES3   555 536
    crew module TPS design sensitivities

11. Carolyn Cooley   Martin Marietta   570 537
    habitat design and habitability test facility

12. Rob Meyerson   JSC EG   577 538
    Earth landing options

13. Bradley Salles   Tracor   592 539
    pneumatically erected rigid habitat

14. Bill Rochelle   Lockheed   597 540
    plume induced environments

NASA

Space Exploration InitiativeTechnical Interchange

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Exploration Programs Office

EXPLORATION
PROGRAMS OFFICE

Main Auditorium
First Series of 5 min. Tech Briefings Tuesday 3:30p.m to 5:15 p.m.

1 Steve Jolly   Ctr for Sp Const.
lunar shelter construction analysis

2 Milton Schwartz   LANL
lunar base concept

3 Gerald Leigh   U. N. M
Center for Extra-terrestrial Eng/Const.

4 Ray Leonard   LANL
innovative radiation shields

5 Willy Sadeh   Colorado St U
inflatable structures for a lunar base

6 Costas Cassapakis L'Garde Inc.
advances in large inflatable structures

7 Dave Criswell   U. of Houston
lunar pwr sys's relevance to landers

8 Maribeth Hunt   Rockwell-Rocketdyne
dynamic isotope power for FLO

9 Jerry Peterson   GE
thermoelectric nuclear pwr sys's for SEI

10 Robert Burke   Rockwell-Rocketdyne
lasers for power beaming

11 Gerald Faibel   Consultant
Moon as a solar power satellite

12 Roger Lenard   Phillips Lab
nuclear propulsion for lunar operations

13 Stan Borowski   LeRC
"Fast Track" lunar NTR

14 Steve Howe   LANL
NTR for FY 2000: a DOE perspective

Main Auditorium
Second Series of 5 min. Tech Briefings Wednesday 8:00 a.m. to 9:30 a.m.

1 Bryce Reimer   Aerojet
SDIO propulsion technology' for SEI

2 Robert Sackheim   TRW
low cost booster system for LEO

3 Mike Jordan   Mitre
space network for lunar communications

4 Don Brown   NASA/JSC/EG
network quality function deployment - avionics

5 Hatem Nasr   Honeywell
off the shelf avionics for future SEI missions

6 Dana Andrews   Boeing
common modules for SEI, ACRV, & LPLS

7 John Hodge   Martin Marietta
multi-stage lander; storable vs. cryo

8 Dave Plachta   LeRC
benefits of cryo vs. storable for return

9 Larry Schooley   Sp Eng. Res Ctr
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crew module TPS design sensitivities

11 Carolyn Cooley   Martin Marietta
habitat design and habitability test facility

12 Rob Meyerson   JSC EG
Earth landing options

13 Gordon Woodcock   Boeing
intg'd sys's for direct mode lunar operations

14 Bradley Sallee   Tracor
pneumatically erected rigid habitat

Technical Splinter Sessions Wednesday, May 6

Room 2-311	9:30 a.m.	11:00 a.m.	1:00 p.m.
	Mission Overview (Kent Joosten)	**Surface Systems and Rovers** (Cab Calloway)	**Space Transportation and Lander** (Ron Kahl)

	General FLO discussion
	Initial Capability: Is it enough?
	  Supportability Issues. EVA Emphasis
	Evolution: Can it happen? Paths
	  Timescales
	Earth landing alternatives
	Mission operations approaches
	Navigation and communication
		General FLO Discussion
		Resupply
		Rover designs
		Habitat designs
		Overall operational philosophy
		Science/ISRU Operations on
		  the surface
			General FLO Discussion
			Earth landing alternatives and systems- land or water
			Ascent vehicle propulsion optimization
			Reusable vs. expendable crew module design
			Radiation protection implementations and capabilities
			Lunar landing risk mitigation
			Fire in the hole assessment for return stage
			Weight reduction methods-lightweight materials, TPS
			Risk vs. performance of return stage alternatives

Room 1-417	9:30 a.m.	11:00 a.m.	1:00 p.m.
	Science & Payloads (Dave McKay)	**Artemis** (D. Weaver/S. Bailey)	**Lunar Habitat** (Molly Elrod)

	General FLO Discussion
	Site Selection Criteria
	Power/Comm accommodations for
	  surface science payloads
	Robotic rovers support to manned
	  exploration traverses
	Safety issues w/ extended pressurized
	  rover exploration EVA's
	Science in FLO-How are we doing?
		General discussion
		Payloads
		  Objectives
		  Priorities
		  Interfaces
		Schedules
		Relationship to FLO
			General FLO Discussion
			Lightweight structure materials
			radiation protection requirements
			  and concepts

Room 2-518	9:30 a.m.	Open for ad hoc splinter sessions
	Risk (Hatice Cullingford) Review of survey results	

SEI Reference Mission

Presented at the ExPO Technical Interchange Meeting

May 5, 1992

N92-33310

1

Expand human presence to the moon and Mars to enhance our understanding of the universe, to seek terrestrial benefits from this exploration, and to establish the beginnings of a sustainable spacefaring civilization.

Approach

Encourage wide participation in the Initiative and aggressively pursue a broad set of exploration goals, while assuring the basic objectives of enabling a human return to the Moon as early as 1999 and human missions to Mars as early as 2005.

❑ The SEI actually involves the simultaneous pursuit of two highly interactive tracks of activity, i.e.

- "Return to the Moon to Stay", and

- "Human Exploration of Mars".

❑ To implement the SEI, the concept of incremental programmatic milestones will be employed. These milestones are a series of major decision events affecting the emphasis and scope of the Initiative as time advances.

- Permits more detailed planning and analysis of the initial capabilities based on

- the more general definition of subsequent <u>potential</u> SEI programmatic milestones.

Near Term Plan
- Well defined
- Linkage to first Milestone clear
- Significant achievement/milestone focus
- Supports
  Budget planning
  Technology/AD planning
  Functional Requirements
  Focused studies

Second Programmatic Milestone
- Less well defined
- Provides general guidance
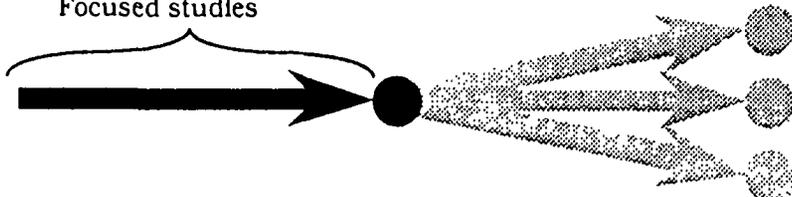
First Programmatic Milestone
- Well defined
- Clear defendable capability levels
- Provides clear guidance for planning

Next Programmatic Milestone(s)
- Goals and strategy defined
- Provides very general guidance
- Includes wide range of possibilities

4

Aggressively pursue development of concepts for early missions and flight hardware that:

- Are significant achievements

- Are phased to fit within available funding

- Contribute to the infrastructure necessary for support of follow-on waypoint capabilities

- Provide flexibility for downstream decisions and implementation choices based on:
    - Operational Experience
    - Test Data
    - Technology Developments
    - Engineering Analysis
    - Mission Discoveries
    - Available Budget
    - National Imperatives

- Encourage investment in additional infrastructure, technologies, and achievements

- Contribute to the goals of timely human and robotic exploration of the moon and Mars

ExPO/Weaver-Reference Mission/May 5, 1992

	1990	1995	2000	2005	2010	2015	2020

HLLV ∗

Lunar Capability Defined

Lunar Operations Begin

Lunar Robotics

Geochemical Orbiter | Geodetic Orbiter | Landed Robotic Lander Missions | Tele P/O rovers & payloads

Lunar Trans. System Dev.

Surface Systems Development

Outpost Systems Development

First Human Mission - First Lunar Outpost (45 day stays)

Sorties as required

Potential 45-90 day lunar stays

Lunar Piloted

Long duration lunar stays

Lunar Outpost Expansion

Technology (TRL 6)

△ Lunar EVA suits, RFC power storage, Autoland, HLLV, Cryogenic fluid handling

△ Advanced ECLSS, In-Situ Resource Utilization, Construction, Surface Nuclear Power

∗ NOTE: Current HLLV planning (i.e. NLS) does not support this plan.

9

Next Capability

Second man-tended outpost at new site
• Repeat local science activities

Outpost serves as core or as "construction shack" for expanded facilities
• Larger crews
• Large science instrument construction and support
• Mars mission support

Early Missions
• Lunar resources mapped
• Lunar topography and gravity model
• Data for informed site selection

Decision Point

Potential Sortie Phase
• Delay in "campsite" systems
• Lunar daylight only
• Scars to crew module must be assessed

Initial Capability
• Transport crew of 4
• ~30 mt cargo delivery
• Man-tended outpost - multiple 45 day stays
• Local surface roving
• Local surface science
• Resource extraction at demo level

Outpost maintained at current capabilities; emphases in different areas
• Long surface roves

SEI Reference Mission Scenario
- Mars Mission Timeline -

NASA
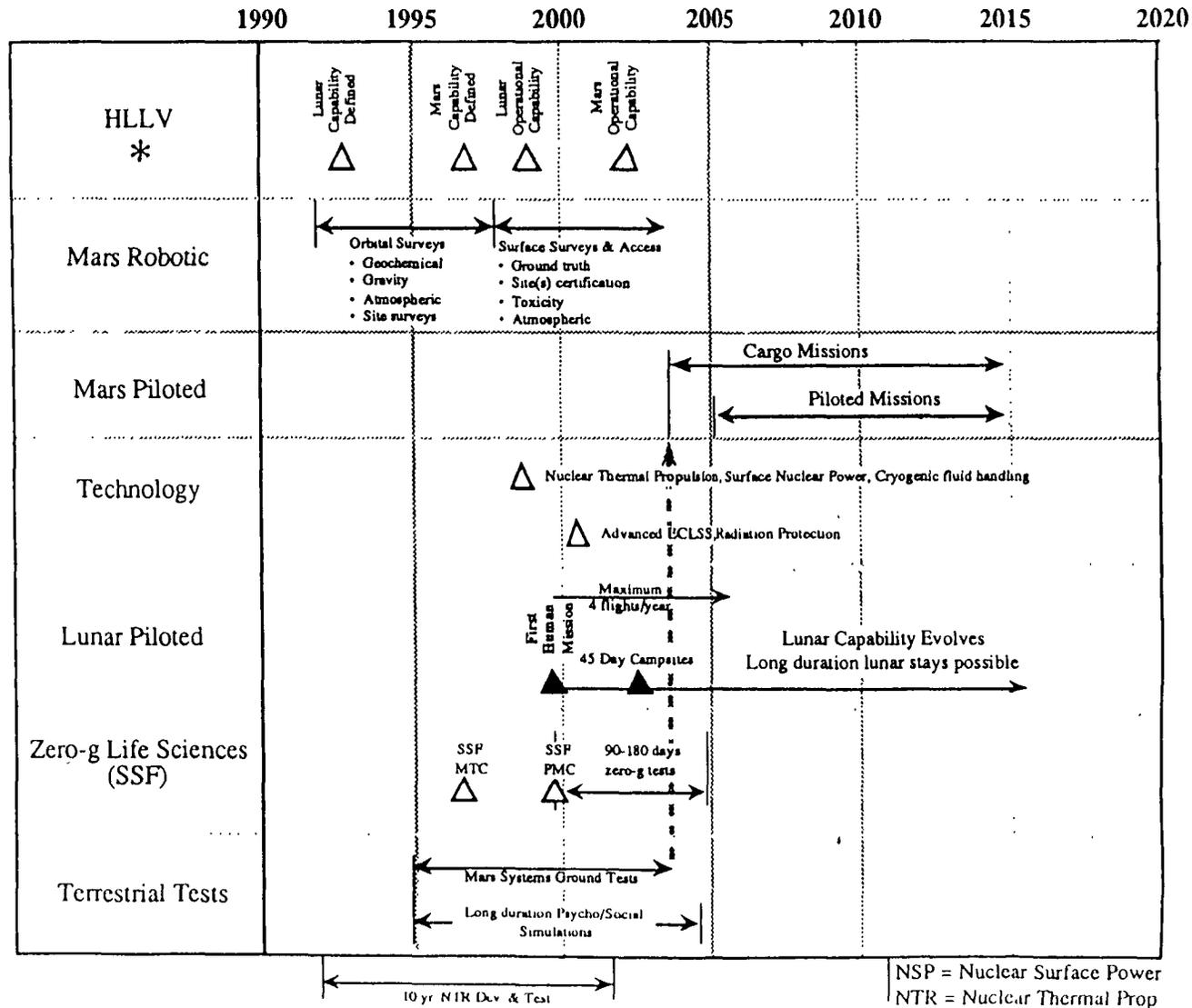
Lunar Resource Mapper /

Lunar Geodetic Scout

Program Status

3rd Space Exploration Initiative

Technical Interchange

N92-33311

April 5, 1992

❏ ***LEXWG Lunar Observer Science Measurement Priorities***

    1.   Global elemental composition (much higher priority than all others)

    2.   Global gravity & topography

    3.   Global mineralogy

    4.   Global imaging

    5.   Global magnetics

    6.   Global atmospherics

    7.   Global heat flow

This ranking is based purely on science priorities, and may not reflect exploration architecture needs.

❏ ***Space Exploration Initiative priorities***

    1.   Global elemental and mineralogical composition (much higher priority than all others)

    2.   Global gravity & topography

    3.   Global imaging

❏ *Science*

- Provides a wealth of fundamental scientific knowledge of the Moon

❏ *Resource characterization and location*

- Provides a detailed geochemical map of the Moon - Fundamental to utilizing the local resources to reduce the cost of exploration

❏ *Technology and Engineering*

- Provides an improved gravity model - A key element in precision landing and long-duration orbital missions
- Provides improved surface imaging - Site selection
- Improved topographic knowledge for outpost design/emplacement

- Lunar Orbiter Programs managed at JSC

- Two orbiter missions planned

  - Lunar Resource Mapper (LRM) launch, March 1995
  - Lunar Geodetic Scout (LGS) launch, early 1996

- Both vehicles small spacecrafts ~ 1000 kg

- Each mission collecting specific data sets to be used for landing site selection

  - Elemental and mineralogical (LRM)
  - Geodetic map, lunar gravity model and global stereo imaging (LGS)

- Experiment selection based on ability for the instrument provider to satisfy mission objectives defined in the LPI working group and Lunar

SELECTION PROCESS

Based on SEI Information Needs

Criteria are Cost, Schedule, Performance

LPI Workshop Screened Potential Instruments

Currently Reviewing Candidates

  -- Soft Xray, Gamma Ray & Neutron Spectrometers
  -- Imaging Spectrometers

First Series of Reviews Completed

Second Phase/Programmatic Reviews This Month

Instrument Selection Early June

Second Flight Screening to Start July

1992 | **1993**

| Jan | Feb | Mar | Apr | May | Jun | Jul | Aug | Sep | Oct | Nov | Dec | Jan | Feb | Mar | Apr | May | Jun | Jul | Aug | Sep | Oct | Nov | Dec |

▲ LPI Instrument Recommendations

▲ Final Instruments Selected

▲ Phase One Contract Award

▲ Phase One Study/Preliminary Spacecraft Design

▲ Non-Advocate Review

▲ Phase Two Contract Award

Detailed Design ▲

Critical Design Review ▲

Development

1994 | **1995**

| Jan | Feb | Mar | Apr | May | Jun | Jul | Aug | Sep | Oct | Nov | Dec | Jan | Feb | Mar | Apr | May | Jun | Jul | Aug | Sep | Oct | Nov | Dec |

▲ Development

▲ Instrument Delivery

▲ Integration & Test

▲ Launch Vehicle Integration

▲ Launch

On-Orbit Test & Checkout

Nominal Mission

Optional Second Spacecraft

LRM/LGS Program Office
Organization

Manager
M. Conley
(713) 283-5554

Business Management
M. Nebrig
(713) 283-5536

— Schedules

— Resources

— Documentation

Program Scientist
D. Morrison
(713) 483-5039

Instrument Selection —

Mission Science Requirements —

Science Lead for Operations —

Planetary Data System —

Launch Vehicle Integration
E. Svrcek
(713) 283-5751

— Delta II

Engineering
B. Drake
(713) 283-5313

— Spacecraft

— Instrument Integration

— Pallet Integration

— Ground Systems

— Mission Planning

Operations
B. Ward
(713) 283-5758

Com./Control (DSN) —

Payload Operations —

Mission Operations —

Data Reduction —

Instrument Development —

Artemis

Artemis **Common Lunar Lander Project Status**

SEI Technical Interchange

May 5, 1992
Houston, Texas

Introduction

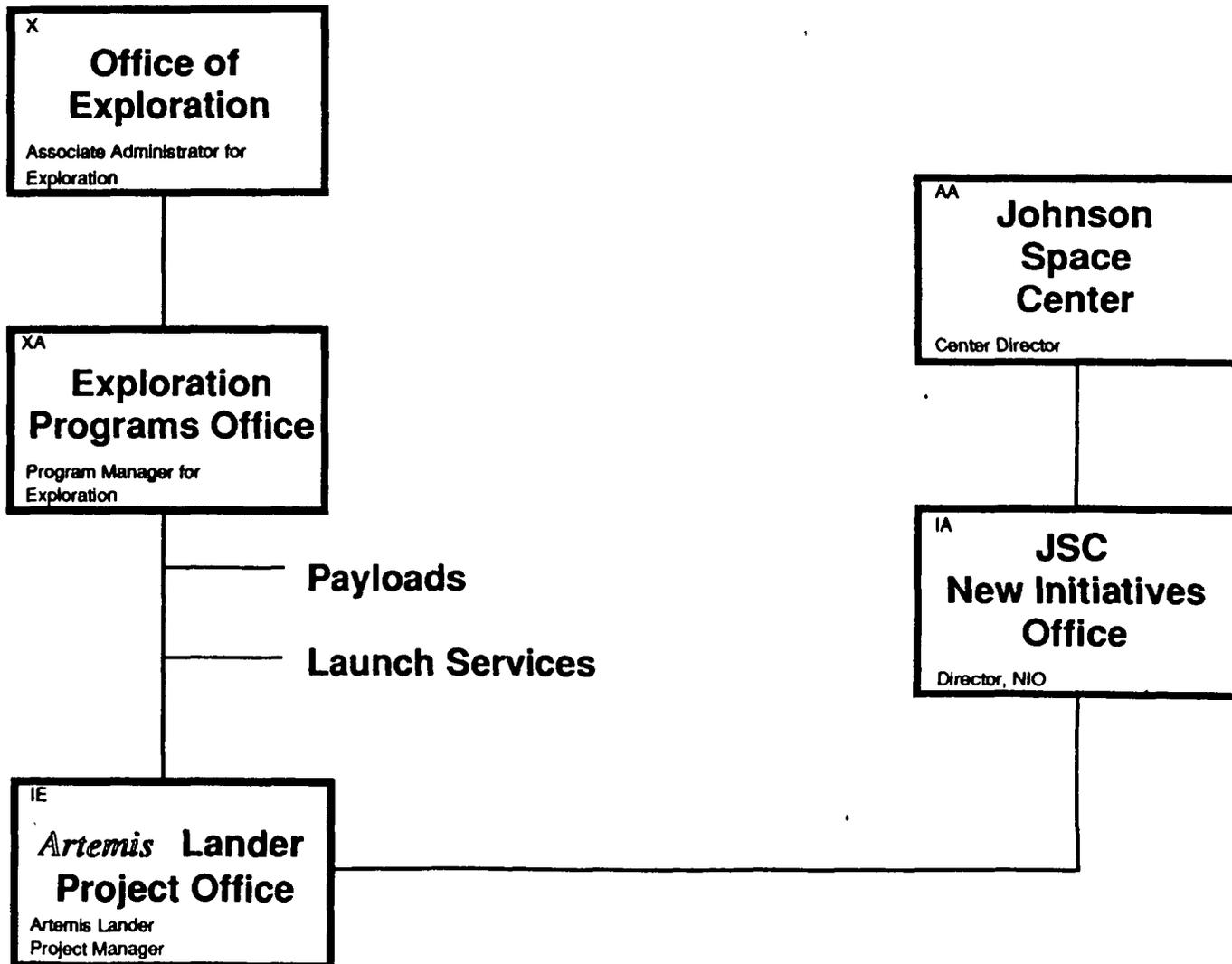*Artemis*

- **Mike Griffin's plans are to start the SEI with·lunar robotic missions that can demonstrate NASA culture change and provide a catalyst for human exploration of the Moon and Mars**

- **The Artemis Common Lunar Lander Concept developed by JSC has been accepted by Mike Griffin as the centerpiece of this lunar robotic exploration program, JSC proposes to develop the Lander in-house**

- **Aaron Cohen strongly supports the Artemis proposal**

- **Program new start (Phase C/D funding) is anticipated in FY '94**

- **First Launch as early as 4th quarter of 1996**

- *A JSC civil service design study is underway which will develop a Project Plan for the development of the Artemis Lander*

Stephen Bailey/IE3/(713)283-5411

New Initiatives Office

Anticipated Program Structure

Artemis

X
Office of Exploration
Associate Administrator for Exploration

XA
Exploration Programs Office
Program Manager for Exploration

Payloads

Launch Services

IE
Artemis Lander Project Office
Artemis Lander Project Manager

AA
Johnson Space Center
Center Director

IA
JSC New Initiatives Office
Director, NIO

Background
Headquarters Perspective

Artemis

"The Common Lunar Lander is exactly the kind of thinking we need to get the Exploration Initiative started and get some early missions going back to the moon."

Mike Griffin (Space News - Nov. 11-18)

"You can buy these by the yard and do that well into the next century ... It will be a long time before we can afford to send men everywhere on the Moon."
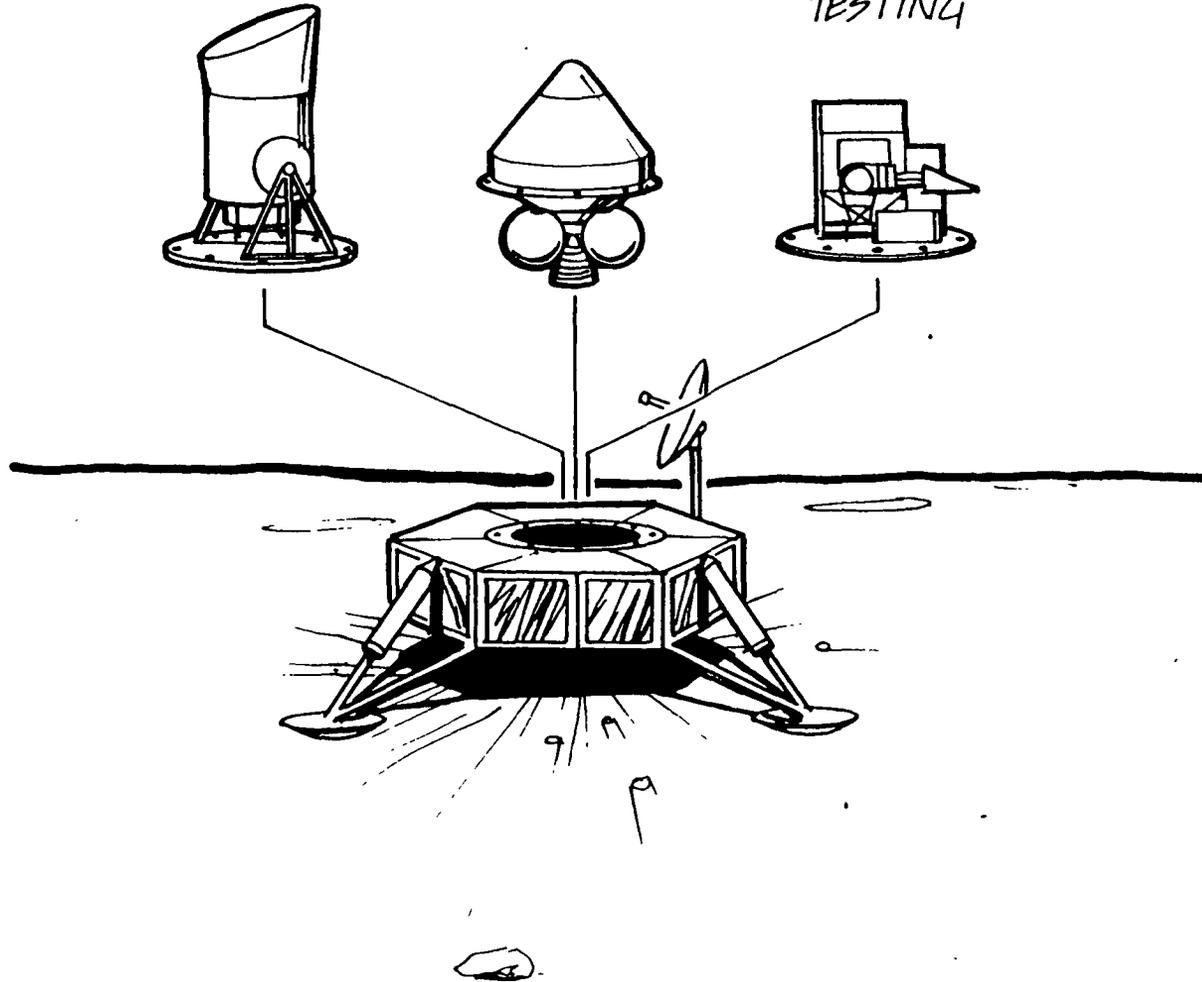
Mike Griffin (Aviation Week - Dec 2)

A principle goal of the Artemis Lander is to safely extend the reach of humans to areas of the lunar surface that would otherwise be inaccessible due to cost or risk.
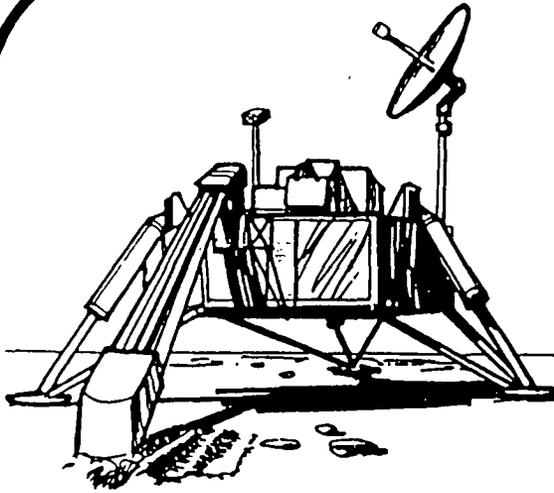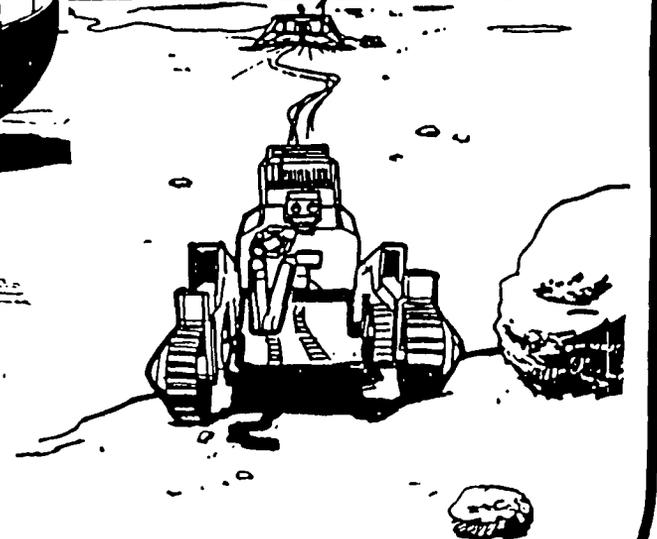
Background -
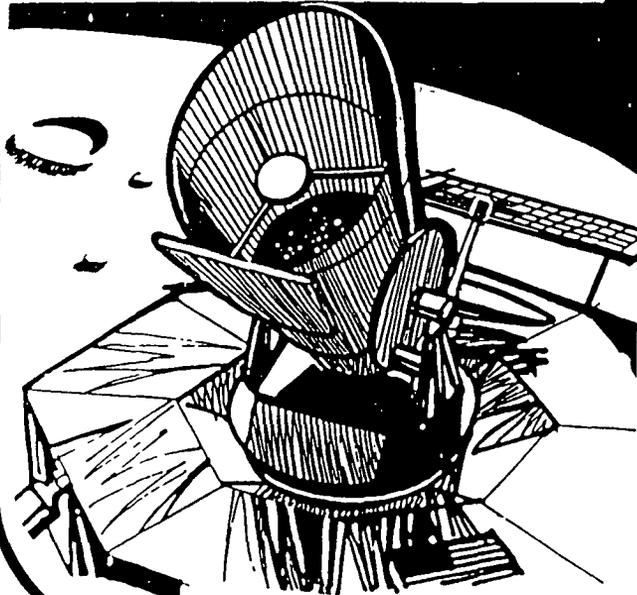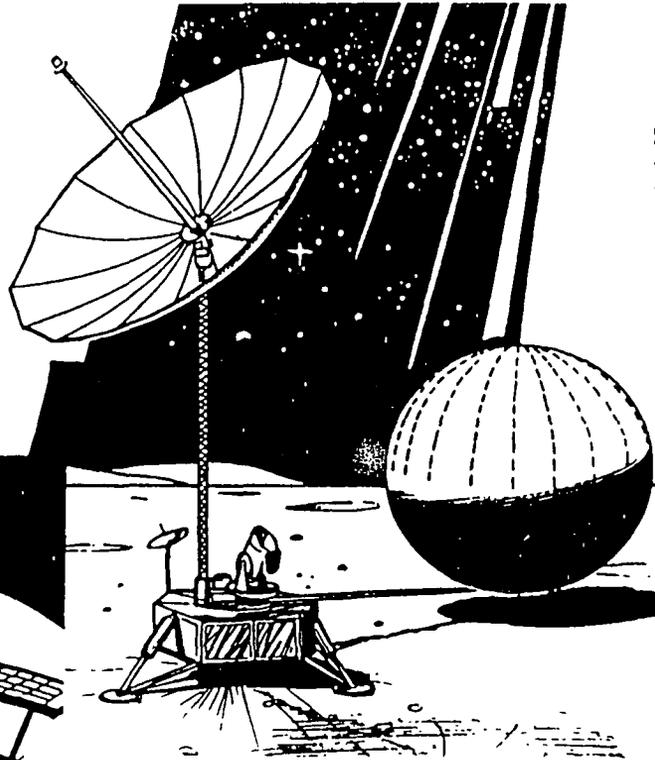# Concept Overview

ASTRONOMY   SAMPLE
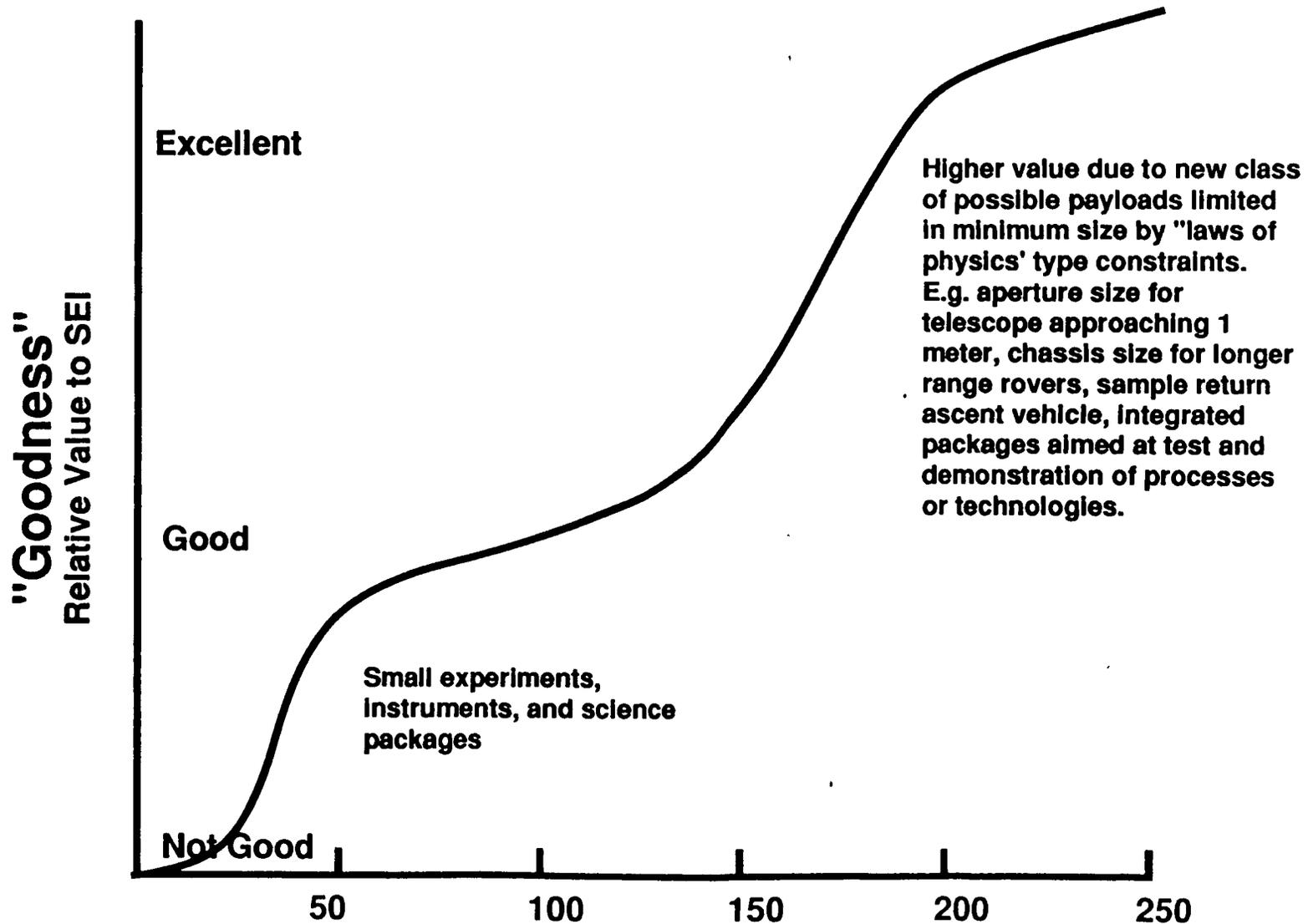   RETURN

MATERIALS
UTILIZATION
TESTING

Artemis

ISMU PILOT PLANT
ON UNMANNED
LUNAR LANDER

TETHERED MICRO-ROVERS

Stephen Bailey/IE3/283-5411

New Initiatives Office

Approach for JSC In-House Study

Artemis

- **Used a small, in-house study team focusing on the lander**
  - **New Initiatives**
  - **Engineering**
  - **Administration**
  - **Safety, Reliability and Quality Assurance**
  - **Mission Operations**
  - **Center Operations**
  - **Human Resources**
  - **White Sands Test Facility**

- **Developed Baseline Concept Design which provides:**
  - **Mass and performance model**
  - **Baseline subsystem design and development models**
  - **Interface and requirements analysis model**
  - **Basis for preliminary cost estimates**

Stephen Bailey/IE3/(713)283-5411

New Initiatives Office

Approach (Cont)

Artemis

- **Released a series of Commerce Business Daily announcements**

    - **Seeking an open dialog with industry on concepts & implementation**

    - **Seeking "off-the-shelf" components and assemblies which could support lander development**

    - **Building a database of potential vendors**

- **Primary objective was to develop the best possible estimates of cost, schedule and Center resources required for lander development**

- **Cost estimates generated from the bottoms-up**

    - **Tasks, level of effort, schedule relationships, equipment, materials, facilities and other costs generated from a series of interviews with all disciplines**

    - **All information captured for analysis and manipulation in the form of an Engineering Master Schedule (EMS)**

Technical Summary
- Requirements -

Guidelines:	Priorities:	Derived Guidelines:
• **Small**	• **Schedule**	• **Use off-the-shelf hardware**
• **Simple**	• **Cost**	• **Avoid block redundancy where single string is adequate**
• **Cheap**	• **Risk**	
• **Quick**	• **Performance**	• **Challenge culture where required to achieve project goals**

- **Requirements**
    - **Land at any latitude and longitude**
    - **Capture into lunar orbit prior to descent**
    - **First launch by June 1996, assuming 1993 new start**
    - **Architecture:**
        - **Launch vehicle is McDonnell Douglas Delta II 7925**
        - **Lander is a single stage spacecraft**
        - **Employ pulse mode engines in order to avoid new engine development**
    - **Best possible payload given the above constraints**

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Artemis

Artemis Lander Project
Architecture Selection
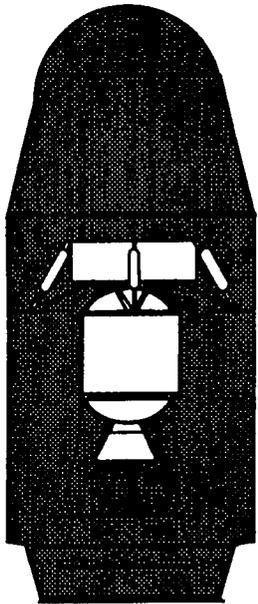
One-Stage
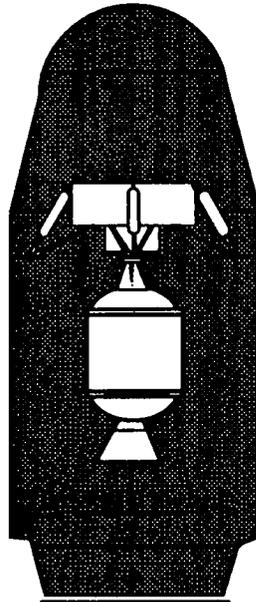
Two-Stage
SRM

1.5 Stage
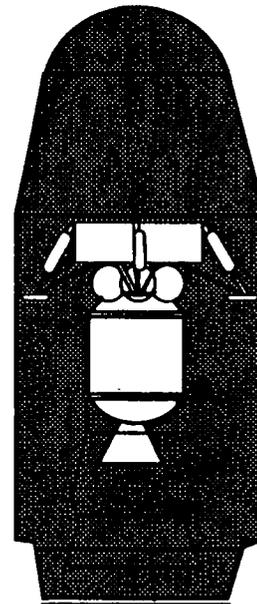
Two-Stage
Bi-Prop
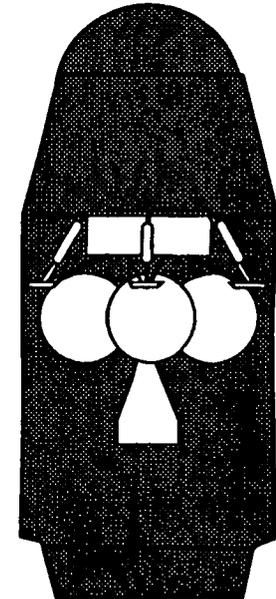
Phase 2
Reference

Phase 1
Reference

Growth Option from Reference Design

Artemis

Delta II 7925
64 kg payload
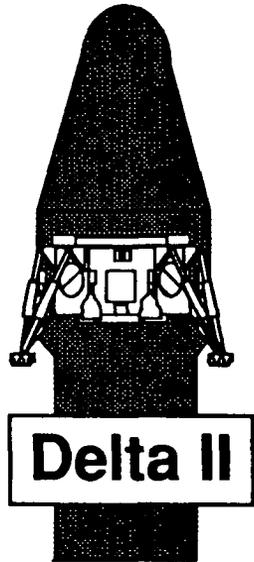
- **Single Stage**
- **Lander provides Trans Lunar Injection (TLI) attitude control**
- **Lander performs portion of TLI**
- **Lander performs Lunar Orbit Insertion (LOI)**
- **Lander performs descent from orbit**

Delta II
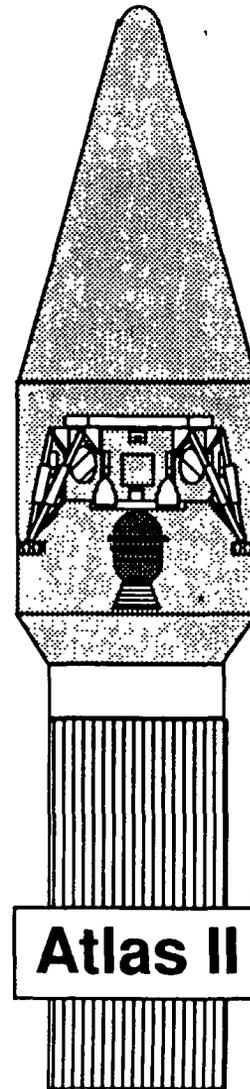
Atlas II Version
>150 kg payload

- **Two Stage**
- **Same Lander w/ some scarring for LOI solid motor**
- **Centaur does an accurate TLI**
- **Lander is passive throughout TLI**
- **Solid Rocket motor performs LOI**
- **Lander trims LOI, and performs descent from orbit**

Atlas II

Stephen Bailey/IE3/(713)283-5411

New Initiatives Office

Example Launch Vehicle Packaging Concept

McDonnell Douglas Delta II 7925
100 in inside diameter payload shroud
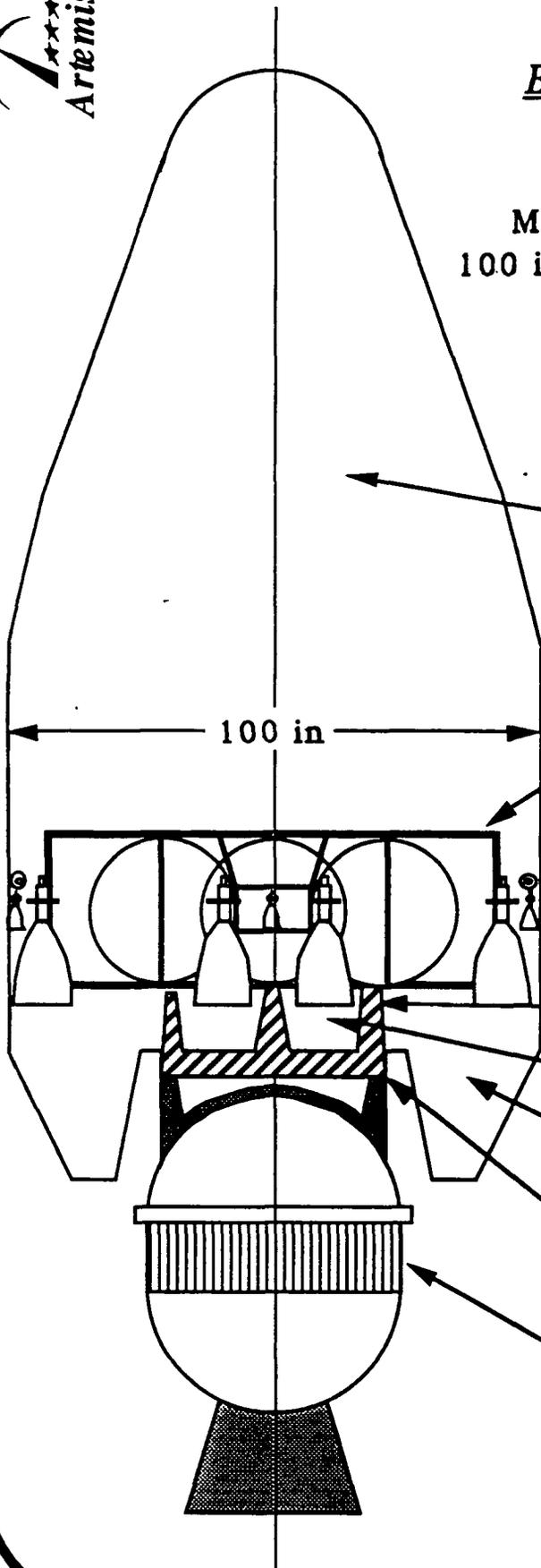
Allowable Payload
Volume Envelope

100 in

Artemis lander

Height above interface TBD

Launch Vehicle Adaptor

Solar Arrays stowed

Legs protrude into here

Payload attach interface

Morton Thiokol Star 48B
(Delta 7925 Third Stage)

Artemis

COMMON LUNAR LANDER
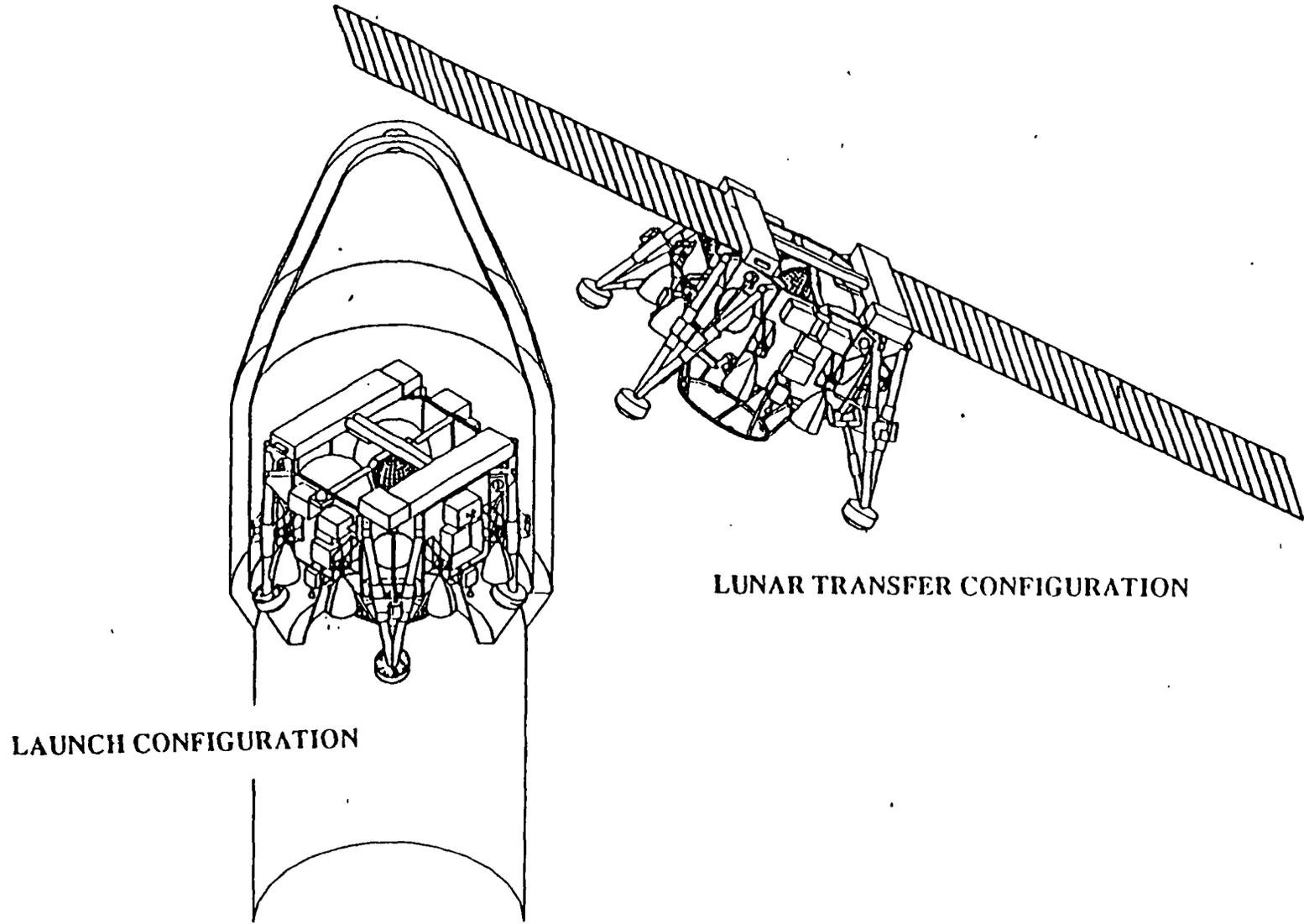
LUNAR TRANSFER CONFIGURATION

LAUNCH CONFIGURATION

Stephen Bailey/IE3/(713)283-5411

BATTERY CHARGER

GUILLOTINE

REAR VIEW PANEL 2

LEFT VIEW PANEL 3

RIGHT VIEW PANEL 1

SOLAR ARRAY STORAGE

GUILLOTINE

FRONT VIEW PANEL 4

SOLAR ARRAY CONTROLLER

TOP VIEW

RCS JETS

ALTIMETER
RADAR

RCS JETS

BOTTOM VIEW

Artemis

PROCESSING
MODULE

S-BAND ANTENNA
90° VIEW CONE (6 PL)

TRANSPONDER

RF ASSY

ALTIMETER

LAUNCH VEHICLE
ATTACH
STRUCTURE

FRONT VIEW

PANEL 4

LANDING
RADAR

GENERAL AVIONICS.
PROCESSOR

TIME GENERATION
UNIT

RCS JET
DRIVER ELECT.

REAR VIEW

PANEL 2

Stephen Bailey/IE3/(713)283-5411

Artemis

INERTIAL
MEASUREMENT
UNIT

STAR TRACKER
80° VIEW CONE

RIGHT VIEW

BUS CONTROLLER

PANEL 1

Delta II 7925 Launch

Evaluation of single stage reference design

- **Performance is acceptable ~ 64 kg**

- **Margins are trim but adequate ~ 56 kg**

- **A growth option to add an LOI strap-on solid motor exists**

- **Further avionics mass reduction may be possible through the integration or elimination of functions**

- **"Off the shelf" goal an implementation driver, significant level of development is required none the less**

- **Delta's third stage is a spin stabilized solid rocket motor (Star 48B)**

- **We decided to eliminate the spin table and nutation damping system and to provide attitude control during the TLI burn**

    - **Payloads have unknown properties**

    - **Our lander is a 3-axis spacecraft - not a natural spinner**

- **Main engines used for TLI attitude control (similar to Magellan S/C)**

    - **Outboard engine mounting locations required for attitude control**

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Delta II 7925 Considerations (Cont)

- **However outboard engine valve and propellant line heaters are required - a major driver for power requirements (over 100 watts continuous)**

    - **Spin stabilization may allow better engine selection/location relative to thermal mass in tanks to reduce power consumption**

    - **Solid rocket dispersions would probably require that main engine lines be wet for significant post-TLI midcourse burn in any case**

    - **Therefore propellant lines for this architecture must be wet at TLI**

- **If heater power requirements are reduced, it is possible that primary batteries could replace the solar arrays completely**

    - **A significant cost and complexity reduction**

    - **Propellant lines must be kept dry until LOI**

- **The Atlas' Centaur upper stage is a highly accurate 3 axis stage**

    - **Post TLI midcourse budget may be low enough that it could be performed with ACS engines - keeping main propellant lines dry until LOI for any main engine selection**

Delta II 7925 Considerations (Cont)

Artemis

- **A series of mass reduction activities are under way**

  - **High performance engine options**

    - **Acquisition of test engine (SDI heritage) underway**

    - **Test will focus on extending operational life to meet our reqts**

  - **Avionics options**

    - **Acquisition of evaluation avionics underway**

    - **Evaluation will focus on trying to consolidate as many avionics functions into a compact system as possible**

    - **Has strong secondary influence on power consumption and associated mass reduction**

- **These mass reduction options may yield a considerably lower parasitic mass fraction which would enable a payload in the 200 kg range to be flown on a one stage, Delta-launched spacecraft**

  - **Time and detailed engineering analysis will tell...**

Stephen Bailey/TE3/(713)283-5411

Next Capability

Second man-tended outpost at new site
- Repeat local science activities

Outpost serves as core or as "construction shack" for expanded facilities
- Larger crews
- Large science instrument construction and support
- Mars mission support

Early Missions
- Lunar resources mapped
- Lunar topography and gravity model
- Data for informed site selection

Decision Point

Outpost maintained at current capabilities; emphases in different areas
- Long surface roves

Initial Capability
- Transport crew of 4
- ~30 mt cargo delivery
- Man-tended outpost - multiple 45 day stays
- Local surface roving
- Local surface science
- Resource extraction at demo

Potential Sortie Phase
- Delay in "campsite" systems
- Lunar daylight only
- Scars to crew module must be assessed

Artemis

Artemis Fits the JSC Strategic Plan

- **JSC as the lead center for exploration must pursue and implement the SEI, the central vision of the Agency**

- **Builds the talent, knowledge and capability of our people**
  - *"Perform selected in-house projects..."*
  - *"Perform selected precursor activities..."*
  - **Strengthens requirements and project management capabilities**
  - **Create experience wedge for future exploration missions**
  - **Emphasize the role of civil servants early in the project**

- **Helps to establish requirements for human exploration**

- **Will help prepare us for living and working on the moon**

New Initiatives Office

38

Conclusions

Artemis

- **We expect to produce a Project Plan which can be supported by Mike Griffin and the Johnson Space Center**

- **We are proceeding with the expectation of a funding commitment at the beginning of FY '94, and are aiming for a 1996 launch**

- **Final architecture and launch vehicle decisions have yet to be made**
  - **System trades will lead to final architecture and launch vehicle selections are anticipated near the end of summer**

- **Component level make/buy decisions as well as procurement strategy will depend in part on the response of industry to an ongoing review of the design in progress - mailing now underway**
  - **We encourage proposals to supply items at all levels from single components to integrated assemblies, subsystems or systems**

- **We welcome review from Industry, other NASA Centers, and other Government Agencies with spacecraft development experience**

Science Themes for Early Robotic Missions: LPI Workshops

Paul D. Spudis

Lunar and Planetary Institute

Presented to 3rd SEI Technical Interchange

Houston, Texas

May 5, 1992

Office of Exploration Precursor Mission "Themes"

Lunar Resources

global, regional, and site compositions and states

Lunar Terrain

topography and morphology

Lunar Gravity

location and magnitude of "anomalies"

Lunar surface lander

soft-landed exploration payloads

Workshop on Early Robotic Missions to the Moon

- Sought to evaluate technical readiness and scientific return from instruments to be flown in lunar orbit or operated on the lunar surface

- Held at the LPI on February 4-6, 1992

- 31 workshop members represented broad cross-section of science community likely to be active in SEI scientific program

- Workshop used "advocate" system to understand technical issues on equal basis

- 60 submissions; 30 orbital and 30 landed payloads or mission concepts

- Orbital concepts evaluated on basis of instrument maturity and degree to which they met LEXSWG orbital measurement requirements; landed payload science requirements still being defined

Workshop on Early Robotic Missions to the Moon

Orbital Mission 2 - Theme: Lunar Terrain

- Laser altimeter

- Gravity experiment, with subsatellite for far side gravity

- Imaging system; global stereo coverage, 15 m/pixel resolution

Lander Mission 1 - Surface rover(s)

- Alpha particle backscatter for chemical analysis

- Mossbauer spectrometer for mineralogical data

- Stereo, high-resolution imaging

- Other instruments (e.g., evolved gas analyzer) as resources permit

Lunar Geoscience Explorer

Theme: Explore the Moon, picking up where Apollo left off and capitalizing on its discoveries

Site: Hadley-Apennines, Apollo 15 site, 26 N, 3 E

Instruments: stereo, high-resolution color imaging, imaging spectrometer, XRF/XRD/Mossbauer spectrometers, gamma-ray/neutron spectrometer, laser breakdown spectrometer (if possible)

Operations: land at LM site for LDEF investigation. Traverses to North Complex, into and out of Rille (studying stratigraphy), up slope of Hadley Delta massif to Silver Spur

Artemis Program

– Rover/Mobility Systems Workshop Results –

NASA ExPO Technical Interchange Meeting

May 5, 1992

N92-33314

- Establish a set of viable rover capabilities and concepts as a function of delivered size.

  — Constrain to a 1997 launch

  — Artemis delivery capabilities of 65 kg and 200 kg

    (Rough estimate is that the rover mechanisms will utilize ~60% of the landed mass and the rover science instruments and payloads comprise the remaining 40%.)

- Address science/payload support requirements.

- Define potential complete Artemis missions:.

  — 65 kg class (single/multiple rovers)

  — 200 kg class (single/multiple rovers)

- Develop cost boundaries and development critical paths.

- Educate ExPO and science community on rover capabilities and limitations.

Mission Theme: **Outpost site survey and resource assessment**

Landing site: Mare Tranquillitatis, highest Ti maria near 15 N, 22 E

- Survey and characterize a potential outpost site on the Moon.

- Large-scale topographic maps/high-resolution stereo imaging

- Surface soil properties, block distributions, and lateral variability will be mapped during rover traverses

- Characterize in situ hydrogen abundance and its variation over a 2 km square grid.

- Vertical distribution of hydrogen in the regolith will be inferred by measuring its concentration in the ejecta of fresh, small craters (20 - 50 m diameter).

- Viable mobility systems appear to be capable of supporting a 1997 launch.

- Achieving the defined mission objectives within a 65 kg payload appears to be possible, although with limited capability.

- The two missions defined can potentially be accomplished within a lunar day, obviating the need for RTG-type power systems.

  - surviving a lunar night would be highly desirable and may be possible without extravagant systems

- Several strategies were advanced for involving educational institutions and the public in a first Artemis mission.
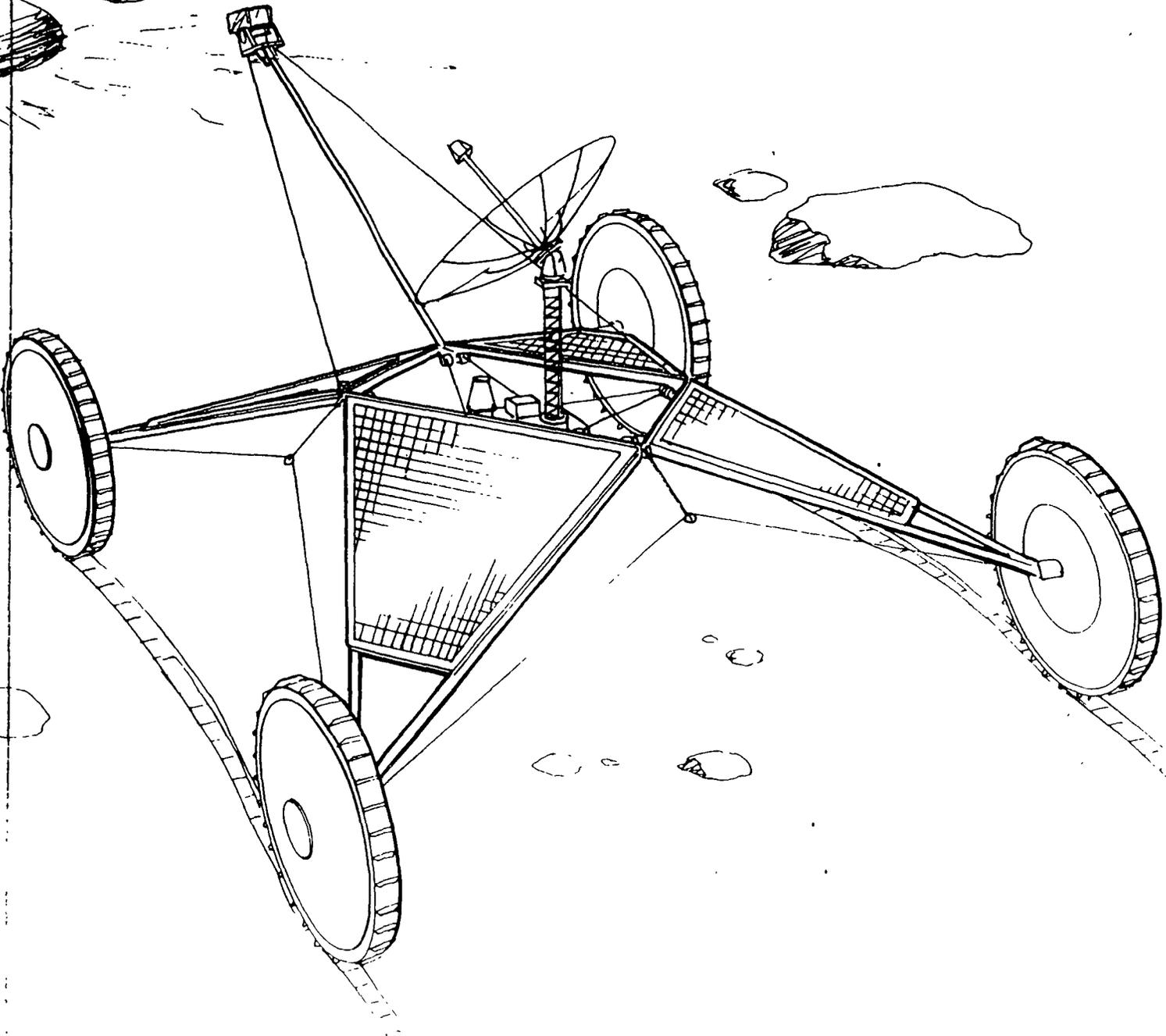
NASA

EXPLORATION
PROGRAMS OFFICE

First Lunar Outpost

Mission Overview

Third Space Exploration Initiative
Technical Interchange

N92-33315

Kent Joosten
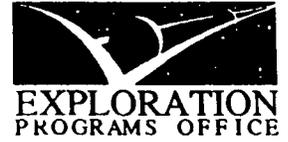Exploration Programs Office
NASA/Johnson Space Center
May 5, 1992

- **Ensure high mission content**

  - Provide lunar science and exploration capabilities exceeding those of Apollo

  - Allow evolution and growth along a number of potential paths as experience and knowledge are gained, and as resources allow

    1. *Augustine* Report - "Go as you pay"
    2. *Synthesis Group* Report - "Waypoint" Philosophy

- **Reduce program costs**

  - Reduce the number of elements and in-space operations required for space transportation

  - Limit number of surface elements required to establish initial lunar capabilities

  - Identify and employ improvements in management, procurement and operational practices

- **Reduce development and deployment times**

  - Mission-driven designs; not "technology for technology's sake"

  - Reduce number of flights and amount of surface operations required before establishing significant science and exploration capabilities

- Each flight to the moon requires a single launch. This greatly simplifies the operations that would be required in a multiple launch scenario.

  - No parallel launch vehicle processing is required.

  - No requirement for long-term on-orbit cryogenic propellant storage.

  - Shorter on-orbit lifetime for transportation elements is required.

  - The need to synchronize multiple launch windows and the trans-lunar injection window is avoided.

- A one-way cargo mission precedes the arrival of the first crew.

  - It is assumed that one or more robotic precursor missions have characterized the landing site to a degree sufficient for certification.

  - A lander carrying the lunar habitat and consumables for the first mission lands autonomously.

  - The habitat is pre-integrated. No construction, emplacement or outfitting is required.

  - The habitat "self-deploys" (solar panels, radiators, etc.)

  - Confidence of habitability is established prior to the departure of the crew from Earth.

RAWLINGS 72

62

RAWLINGS 92
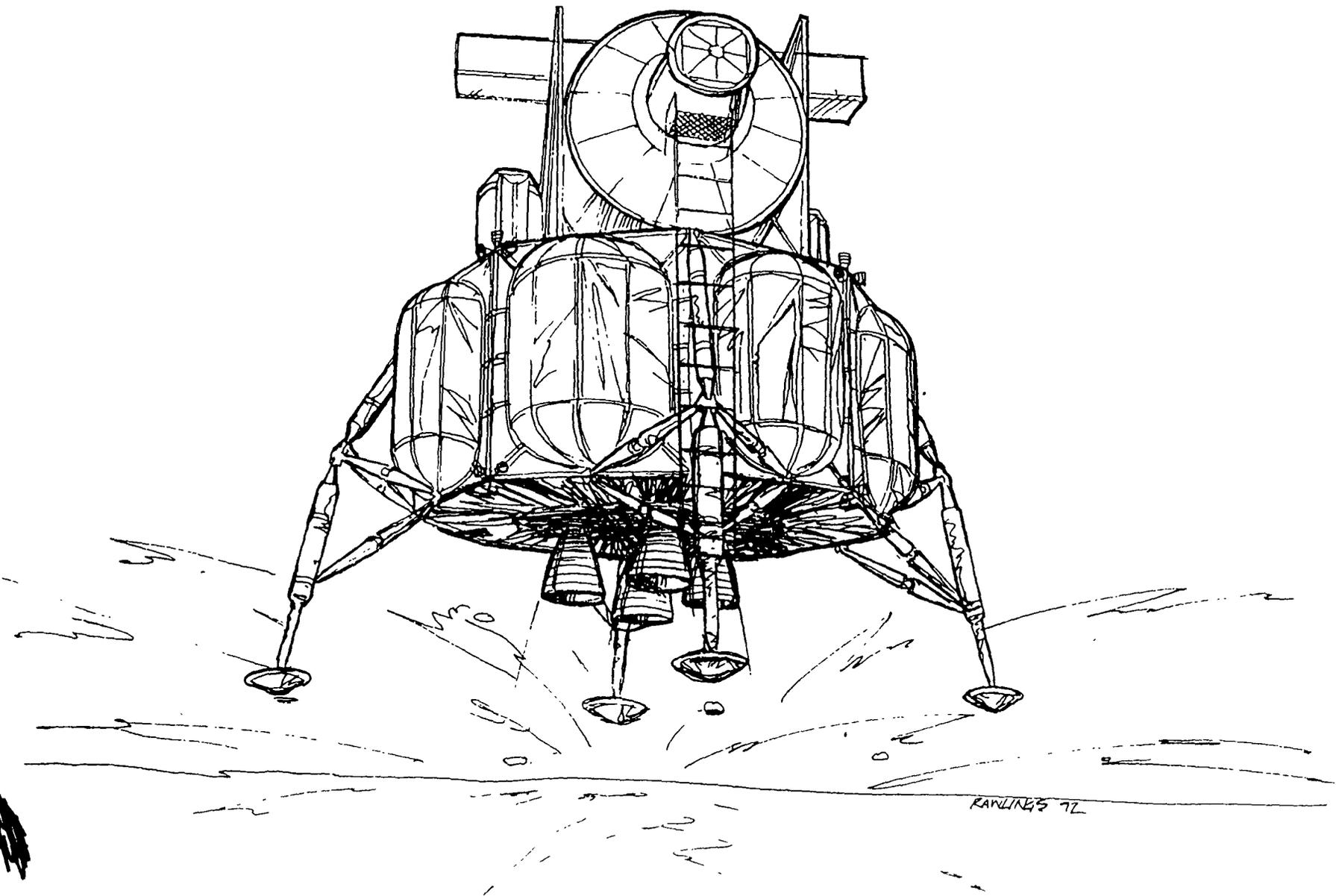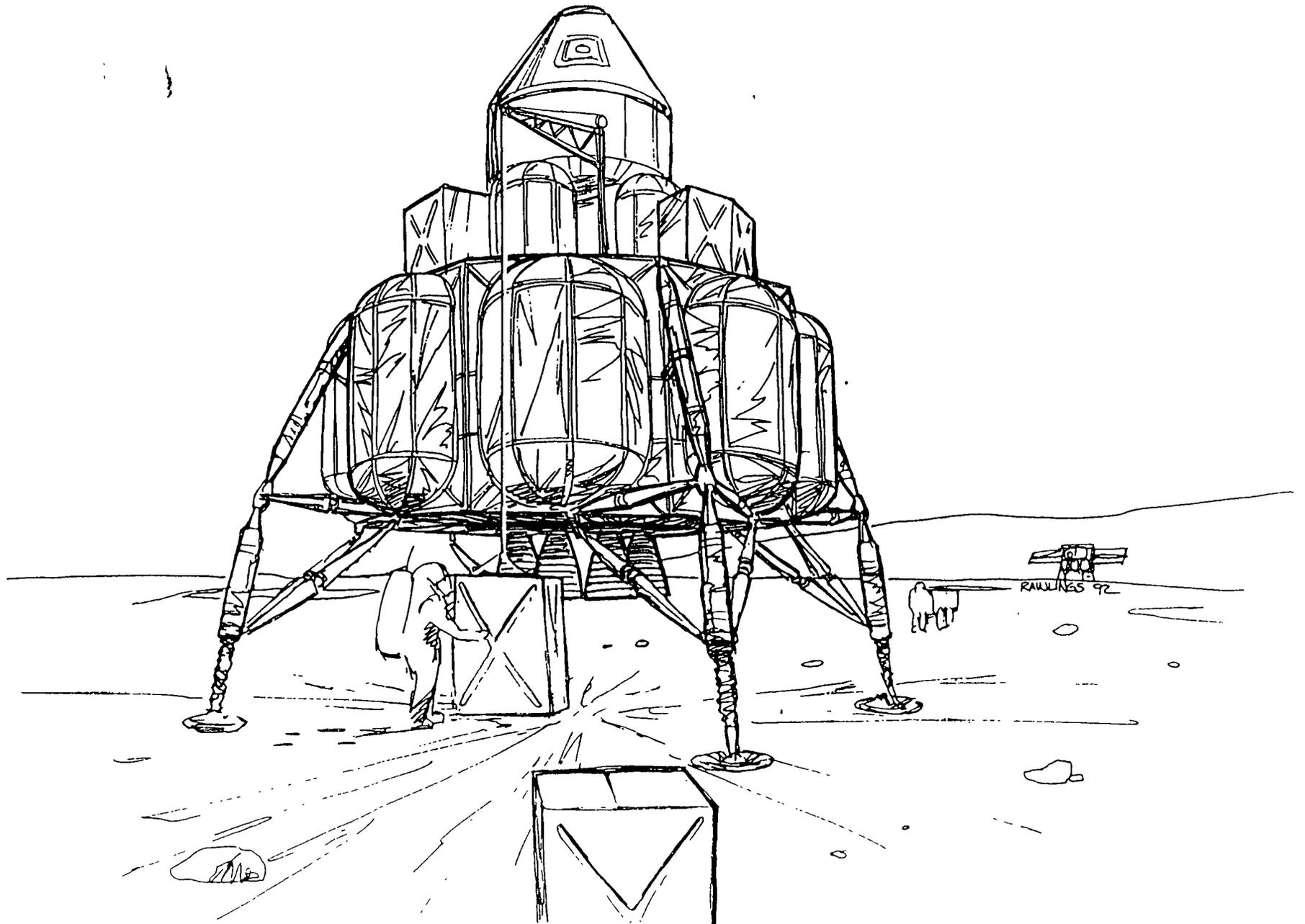
- The crew performs both EVA and IVA surface exploration and science activities.

- After the surface mission is complete, the crew returns to Earth.

  - The crew configures the habitat to an unmanned mode and returns to the piloted vehicle.

  - Lunar liftoff and trans-earth injection are performed by the return stage.

  - Crew module performs a direct entry at Earth, similar to Apollo.

  - The capability of landing on land is being pursued in anticipation of reducing recovery operations costs and enhancing crew module reusability.

- Plan to revisit the outpost at ~6 month intervals.

  - Revisit logistics are delivered with the new crew. These are transported to the habitat using the rover.

RAWLINGS 92

RAWLINGS '92

NASA
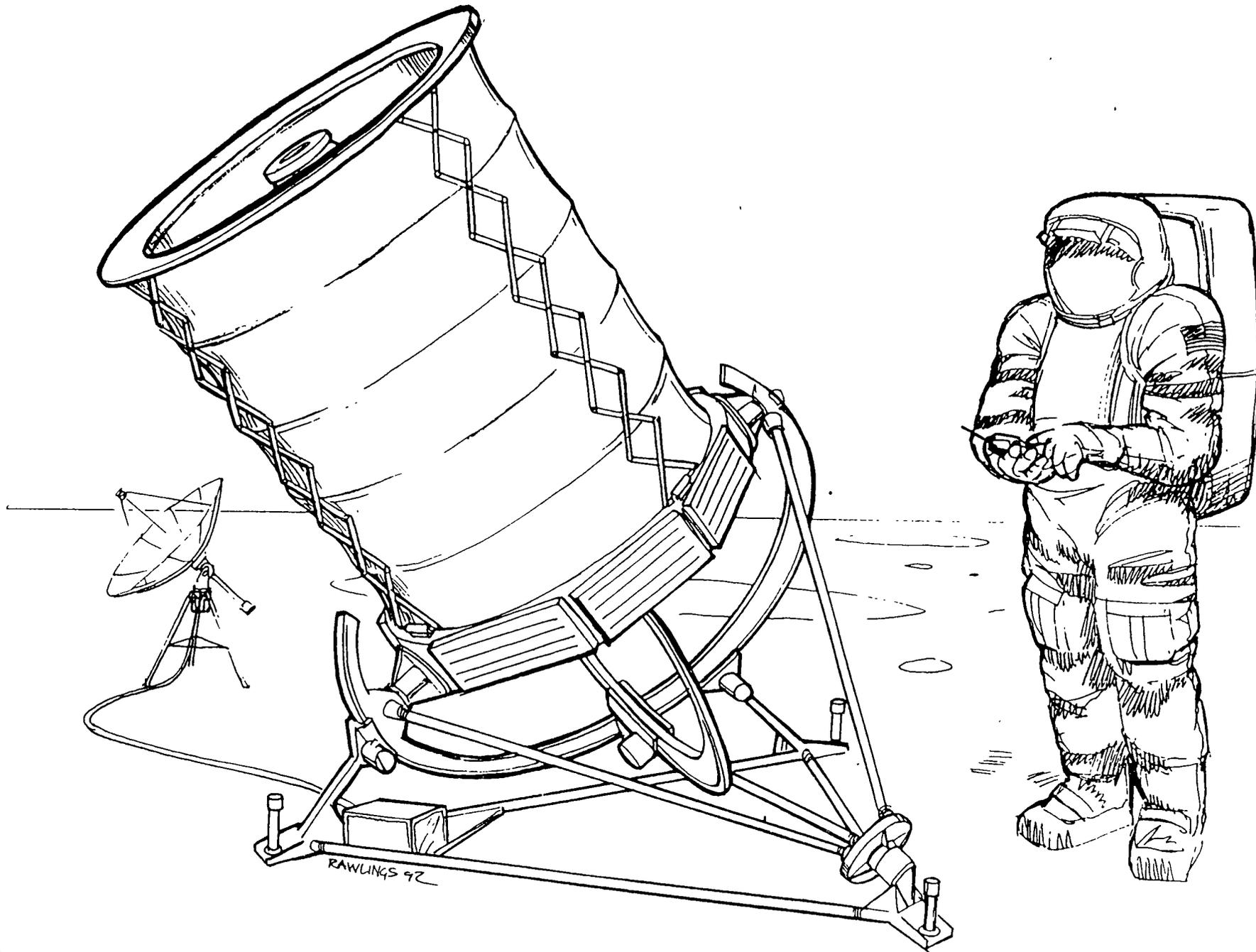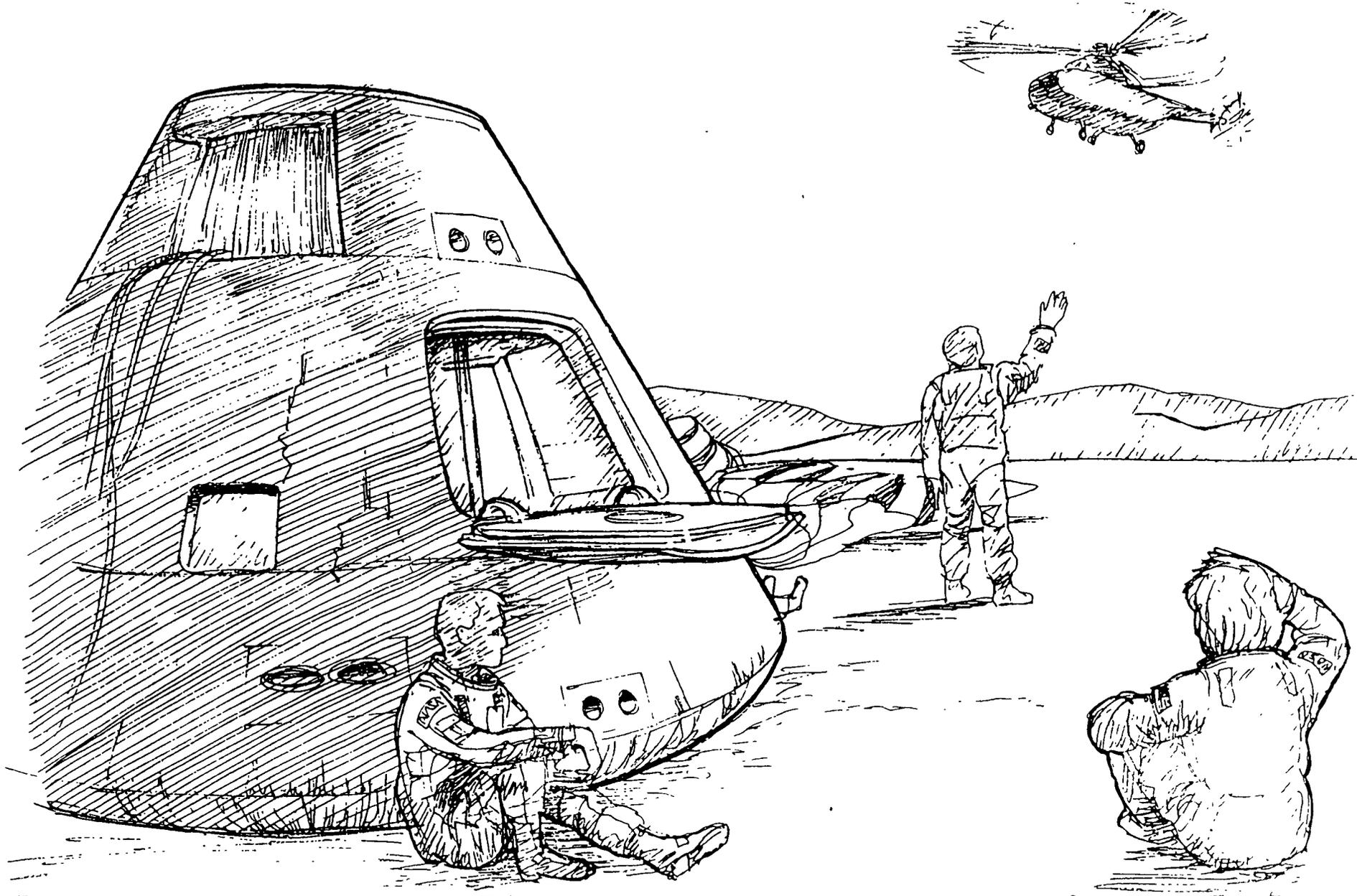
First Lunar Outpost
Evolution Options

EXPLORATION
PROGRAMS OFFICE

1) Transition to a continuously occupied outpost to increase the man-hours available on the moon.

- Allows deployment and maintenance of observatory-class optical telescopes and radio telescope array elements
- Allows for in-depth lunar science activities, such as drilling, trenching, etc.
- Allows for longer-duration partial-g life-sciences investigations and operational experience applicable to Mars missions
- This would most likely involve additional pressurized volume and power, enhanced radiation protection, and more extensive resupply capability.

2) Increase roving capability beyond the limits imposed by the EMU/rover.

- Allows geophysical investigations over much larger areas
- Allows support and maintenance of widely distributed instruments

3) Establish a second identical outpost at a remote site.

- This may be more efficient than long-range roving.

Science and in situ resources utilization (ISRU):
Design reference mission for the First Lunar Outpost

David S. McKay
Solar System Exploration Division
NASA Johnson Space Center
May 5,1992

Science and Payloads Team

1a. Disclaimer

1. Requirements and themes

2. General approach

3. Science payloads

4. Trade studies: past, present, and future

5. Science evolution

6. What are we doing now?

7. Where do we go from here?

Science and Payloads Team

DISCLAIMER:

- **All science activities and payloads in this reference mission are to be considered** *strawmen only.*

- **Actual science activities and payloads as well as the FLO landing site will be chosen by appropriate committees, panels, and representatives from the respective science disciplines after due deliberation and agreement.**

- **However, the following activities are believed to be reasonably representative of a set which may finally be chosen.**

- **These activities provide necessary information on masses, dimensions, EVA activities, power, etc. as required for planning and designing other parts of the FLO reference mission.**

FLO Science and Payloads Team
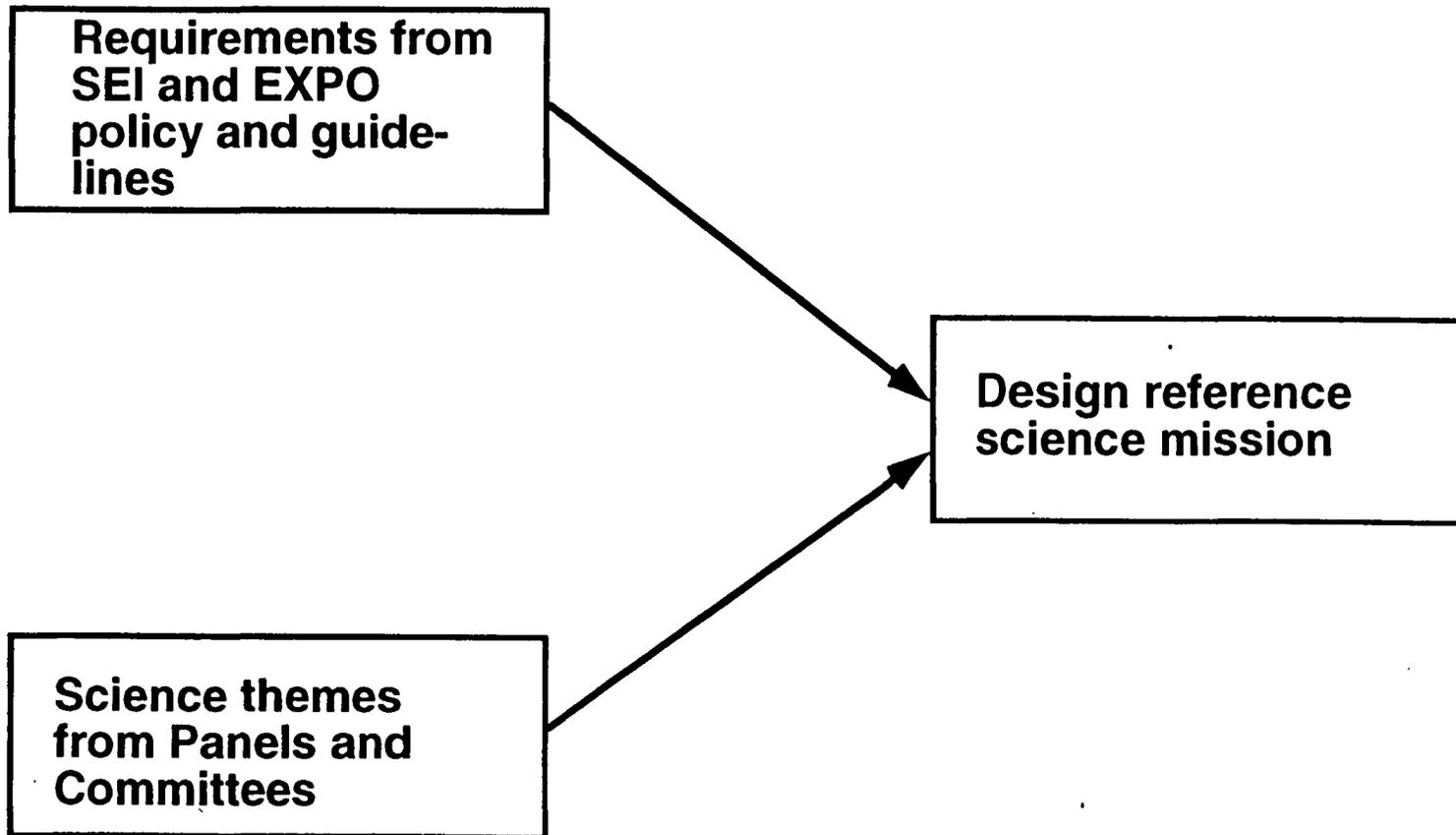
Science and Payloads Team

Science objectives of the first lunar outpost are derived from several top level goals of the SEI program as listed in the First Lunar Outpost Requirements and Guidelines (FLORG):

• The SEI shall expand knowledge of the solar system and the universe.

• The SEI shall establish continual human presence on the Moon.

• The SEI shall research the utilization of space resource.

Science and Payloads Team

Science Themes

- A series of science themes and first order scientific questions has been developed for each major science discipline

- These science themes and questions are mostly traceable to major reports of the scientific community and their representative bodies such as the National Academy of Science.

- The strawman science activities and payloads are designed to address these major science themes and questions

Science and Payloads Team

Comparison to Apollo:

- An additional scientific objective is to provide more and better scientific data from FLO than from all Apollo missions combined

- It is judged that this is achievable in most disciplines

- FLO science is designed to provide much better geologic and geophysical coverage of a local area than any Apollo mission, both laterally and vertically

- FLO science will provide pioneering new science in astronomy

- FLO science will provide basic new information on planetary life support and human performance

- FLO science combined with robotic orbiter and Artemis science will provide a significantly better understanding of the moon as a planetary body and as detector and recorder for events in the solar system and the universe

FLO Science and Payloads Team

Science and ISRU Activities

EVA geoscience and resource traverses

Outpost mission

IVA life science and geoscience activities

EVA payload deployment and operation

Robotic activites between missions:

(Payloads and rover)
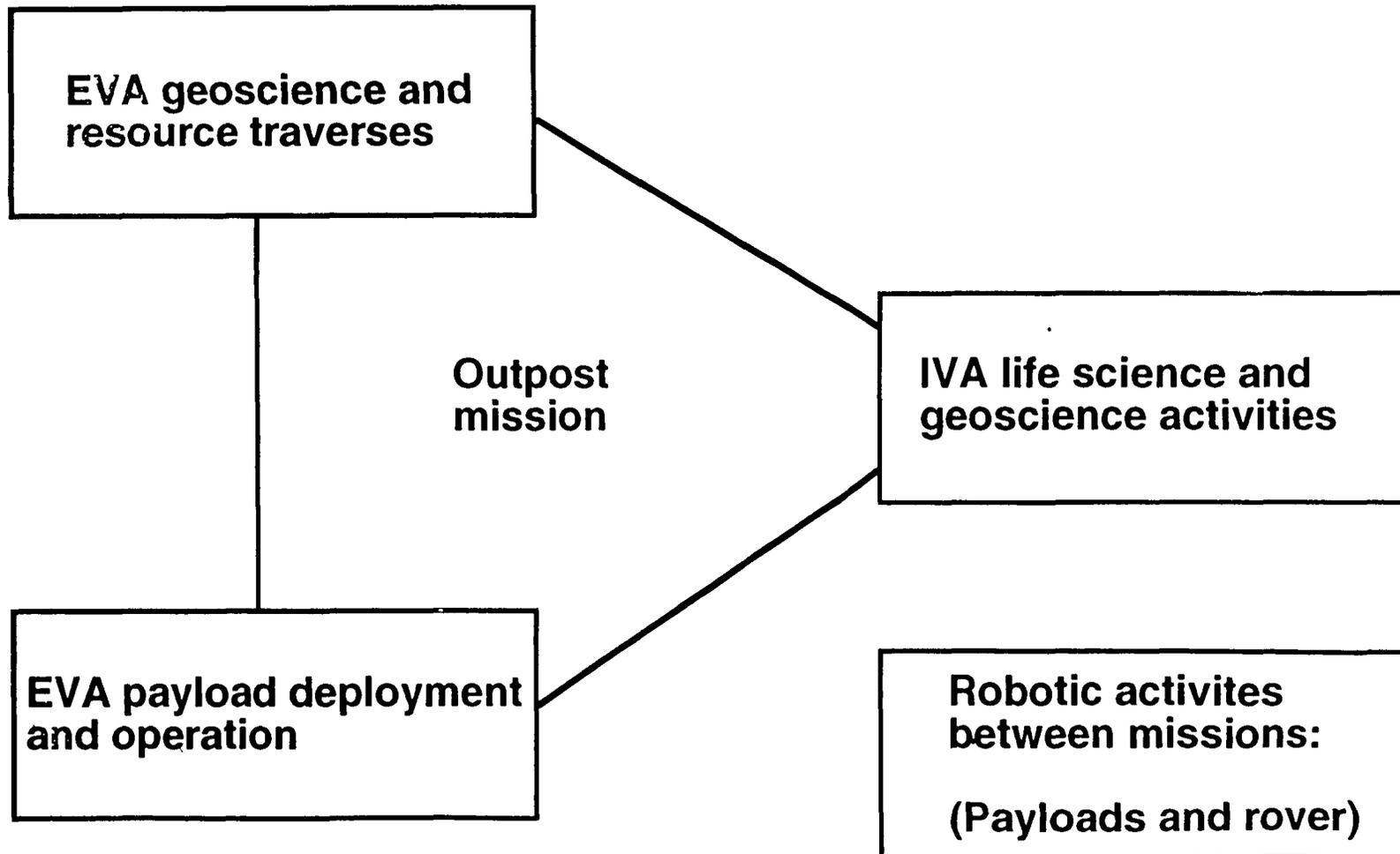
Science and Payloads Team

General Approach:

Science activities consist of four components:

1.EVA crew activities: performing life science, geologic, geophysical, and geochemical investigations

2. EVA crew activities: deploying and activating experiments

3. IVA crew activities: performing life science investigations, and analyzing, sorting, and packaging geologic and life science samples for return to earth

4. Complementary robotic activities including operation of deployed payloads and additional experiment deployment and sample analysis, collection, and return done by the rover in robotic mode.

Science and Payloads Team

Strawman sites

• An initial strawman site (Mare Smythii near the equator on the eastern limb) has been chosen for specific science planning, including traverse layouts for EVAs

• An alternate strawman (Aristarchus Plateau at 23 N, 48 W) was chosen to determine if it made significant difference in the design reference mission

• It was found that the alternate site made no significant difference in the reference mission in terms of payloads and EVAs. Detailed differences in the timelines resulted from a different set of preplanned traverse stations

The overall conclusion is that, except for some specialized sites (lunar poles, crater bottoms, unusual features, etc.) the mission science payload and EVA activities will not change much from site to site.

Science and Payloads Team

Geoscience and Resource ExplorationTraverses

- For the Mare Smythii site, 9 looping traverses were laid out which ranged out to a maximum of about 25 km from the outpost

- The traverse were designed to visit all major features and to provide detailed geologic recon of an area about 50 km in diameter around the outpost

- Each traverse was divided into segments suitable for one 8-hr EVA on the rover

- Initial timelines indicate that about 5 to 6 of the traverses could be completed on one mission leaving the rest for future missions

- The number of traverses is flexible and can adapt to available EVA time

Science and Payloads Team

Experiment Packages

- A number of experiment packages are deployed on the lunar surface

- Some experiment packages are "set and forget"

- Some experiment packages require interactive crew participation

- Four major scientific disciplines plus ISRU are represented by the experiment packages:

    - Astronomy
    - Geophysics
    - Life sciences
    - Space and solar system physics

Science and Payloads Team

Strawman Payload	Mass (kg)	Dimensions (folded)
Geophysical Monitoring Package	200	1m x 1m x 1m
Solar System Physics Expt Package	200	0.25m x 0.25m x 0.25m
Traverse Geophysical Package	400	0.5m x 0.5m x1m
Lunar Geologic Tool Set	400	0.5m x 1m x 1m
Lunar Transit Telescope	230	2.7m x 1.1m
Small Research Telescope	200	2m x 1m x 1m
Small Solar Telescope	100	0.5m x 0.5m x 1.5m
ISRU Demo Package	700	1.4m x 1.4m x 1m
Robotic Package for Rover	300	1m x 1m x 2m (initial est.)
Life Science Package (EVA)	200	0.2m x 0.3m x 0.3m(initial est.)
Total	2930	

Note: Habitat laboratory equipment *not* included in this payload budget

Science and Payloads Team

Geophysical Monitoring Package

- Measures values of a number of geophysical parameters

- Provides information on lunar heat flow, magnetic strength, seismic activity, micrometeorite and secondary ejecta flux, and precise distance measurements to earth

- Has ALSEP heritage and will contain very similar instruments

- Powered by central station, possible photovoltaic and batteries, or possibly RTG

- Must be deployed by crew and activated; robotically operated thereafter from earth

Science and Payloads Team

Solar System Physics Experiments Package

- Contains instruments to particles and fields

- Contains instruments to analyze the lunar atmosphere and its variations

- Also has ALSEP heritage and will contain some similar instruments

- Powered by central station, possible photovoltaic and batteries, or possibly RTG

- Must be deployed by crew and activated; robotically operated thereafter from earth

- FLO package may be identical to one designed for Artemis landing

Science and Payloads Team

Traverse Geophysical Package

- Contains instruments which can be operated from the rover

- Instrument package includes:

    - Electromagnetic sounder for subsurface data collection
    - Active Seismic Experiments for data on upper few kilometers
    - Traverse Gravimeter to track variations in lunar gravity from place to place
    - Electrical Properties Experiment to help determine subsurface structure
    - Profiling Magnetometer to measures local variations in magnetic field

- Instruments will be operated by crew and powered by rechargeable batteries
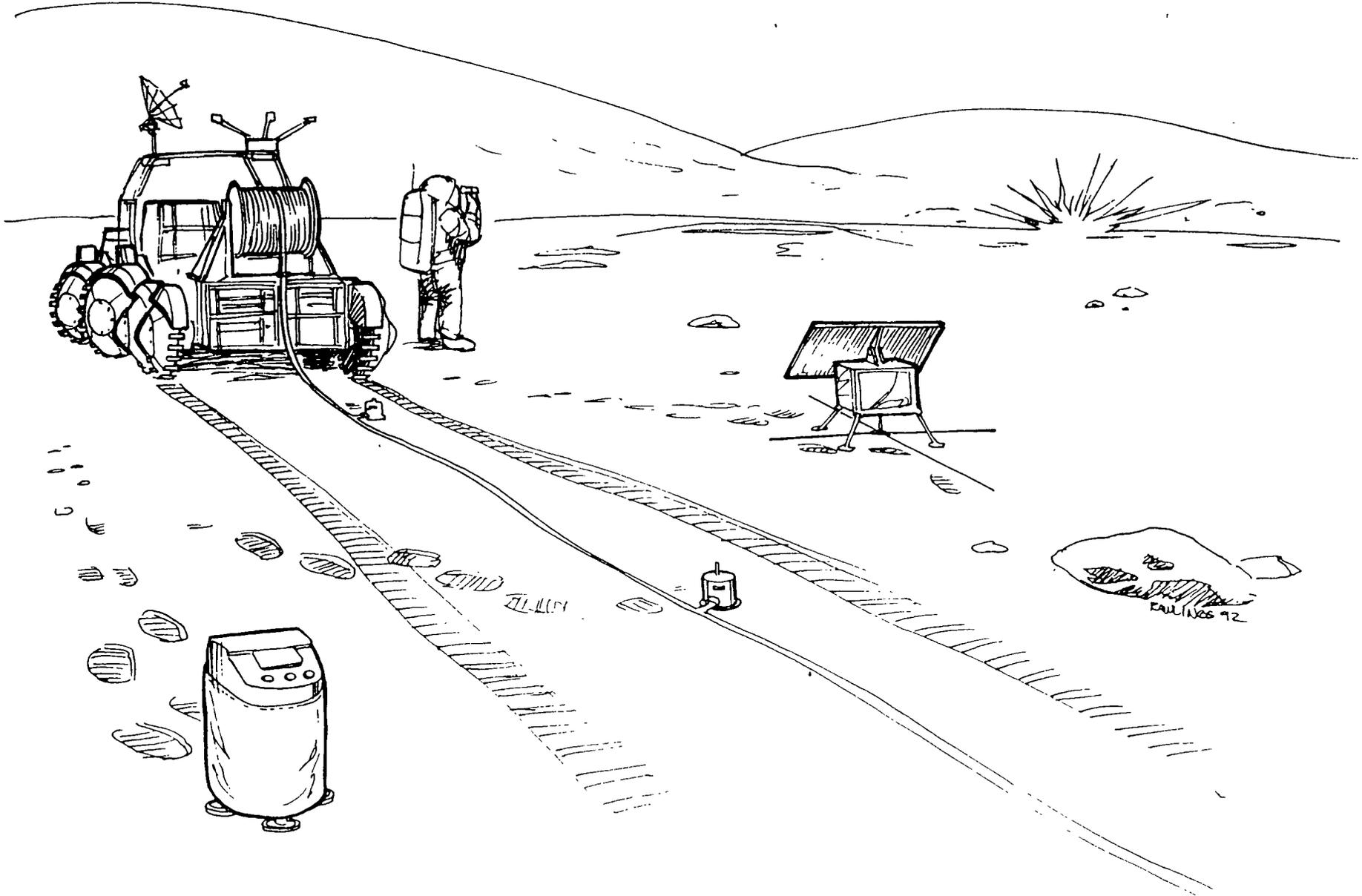
Science and Payloads Team

Lunar Geologic Tool Set

- Contains tools, cameras, and sample containers required for geologic investigations and exploration
  - Apollo-type 3-m drill
  - Hammers
  - Rake
  - Soil sampler
  - Scoops
  - Tongs
  - Drive tubes
  - Cameras/digital imaging
  - Sample containers
- Will be carried on traverses and used for collecting and documenting samples

Science and Payloads Team

Lunar Transit Telescope

- Will take advantage of minimal lunar atmosphere to survey the sky in the UV spectral range

- Will provide images of stars, galaxies, clusters, and the interstellar medium in the UV

- Deployed by crew at Telescope Farm and operated remotely from earth

- Can be powered by solar cells for lunar daytime operation only
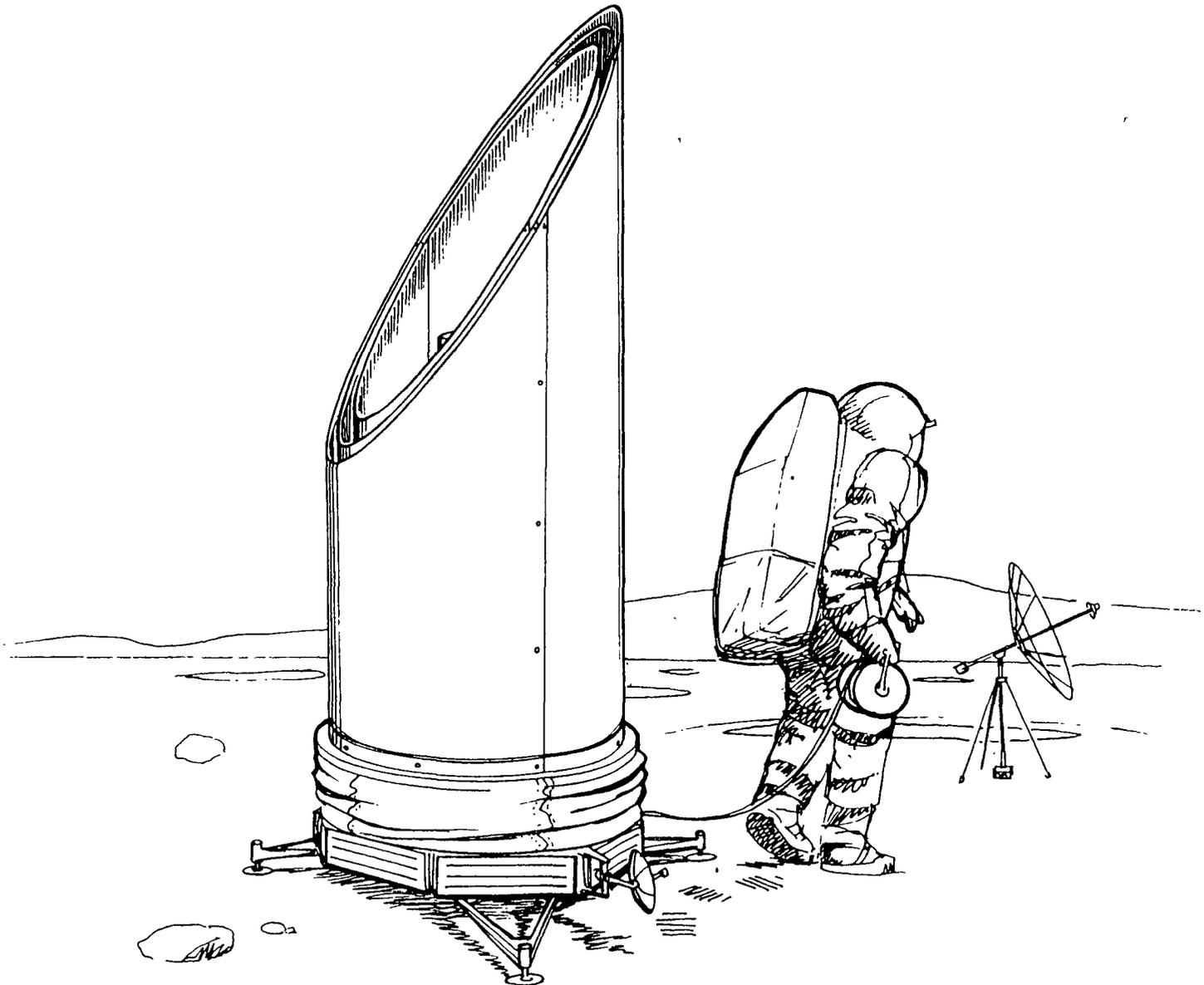
Science and Payloads Team

Small Research Telescope

- Pointable telescope for various astronomy objectives

- To be deployed by crew at 10 km or more from habitat: Telescope Farm

- Will be operated and pointed from earth and will transmit data to earth

- Will provide valuable engineering and operating data for future telescopes

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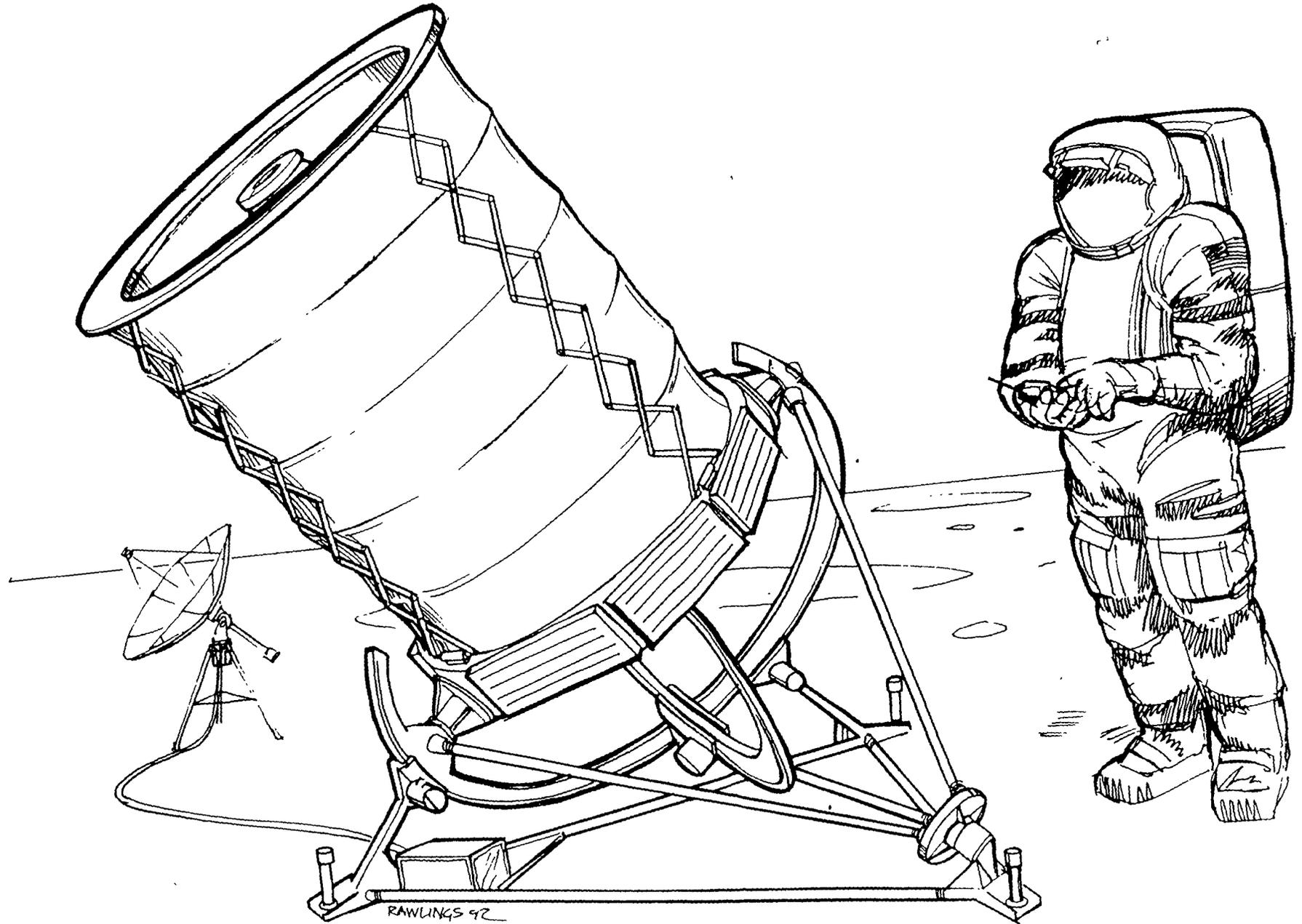
Science and Payloads Team

Small Solar Telescope

- Will provide high resolution images of sun to support solar flare tracking and other solar process investigations

- Will typically make on solar image every 30 seconds

- Operates only during lunar day

- Will be deployed by crew at Telescope Farm (10 km from outpost)
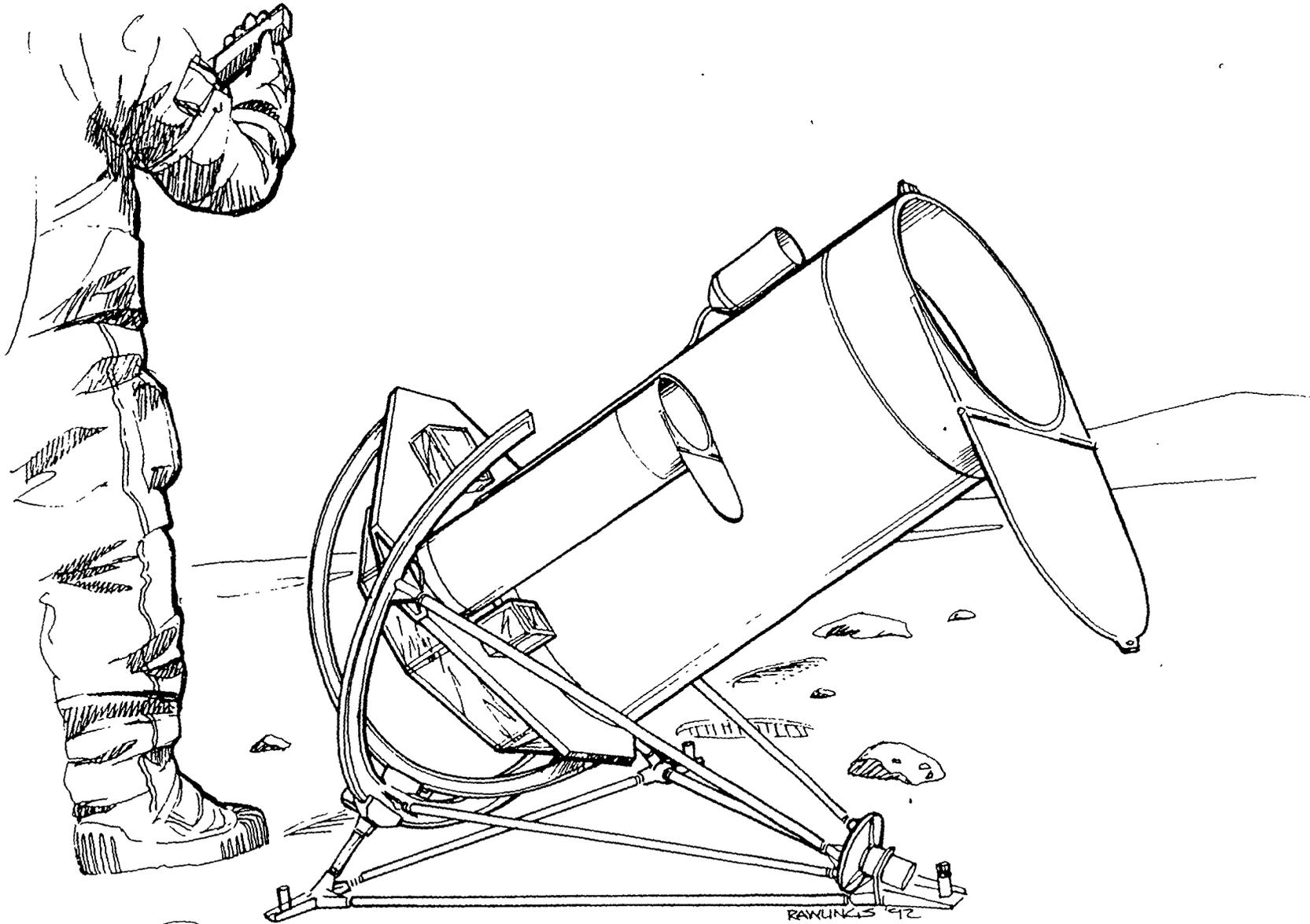
Science and Payloads Team

Life Science Packages (EVA):

• Monitoring equipment for human performance during EVA

• Will allow specialized testing for such attributes as vision, locomotion, balance, orientation, etc.

• Data will be collected for later analysis

• Exobiology experiment may consist of cosmic dust collector deployed on lunar surface to be returned on a later mission for analysis

Science and Payloads Team

ISRU Demo Package

• **Enables early ISRU demonstration and validation**

• **Allows testing of basic processes**

• **Will provide operational experience for later production units**

• **Contain three basic subsystems:**

   1. Oxygen extraction unit which uses imported hydrogen and makes water from lunar oxygen testing various feedstocks and parameters

   2. Brick making unit which investigates a number of variables for optimizing brick/block fabrication by sintering of lunar soil

   3. Gas-solid flow unit which tests pneumatic transport and pneumatic size sorting methods

```
                    ┌──────────────┐
                    │   ISRU       │
                    │   Demo       │
                    │   Package    │
                    └──────┬───────┘
          ┌────────────────┼────────────────────┐
    ┌──────────┐     ┌──────────┐         ┌──────────┐
    │ Oxygen   │     │ Bricks   │         │ Solids   │
    └──────────┘     └──────────┘         └──────────┘
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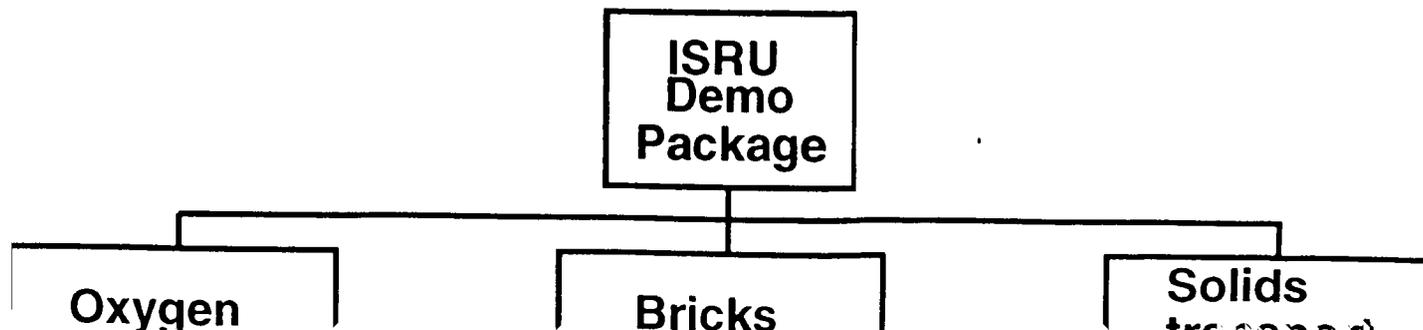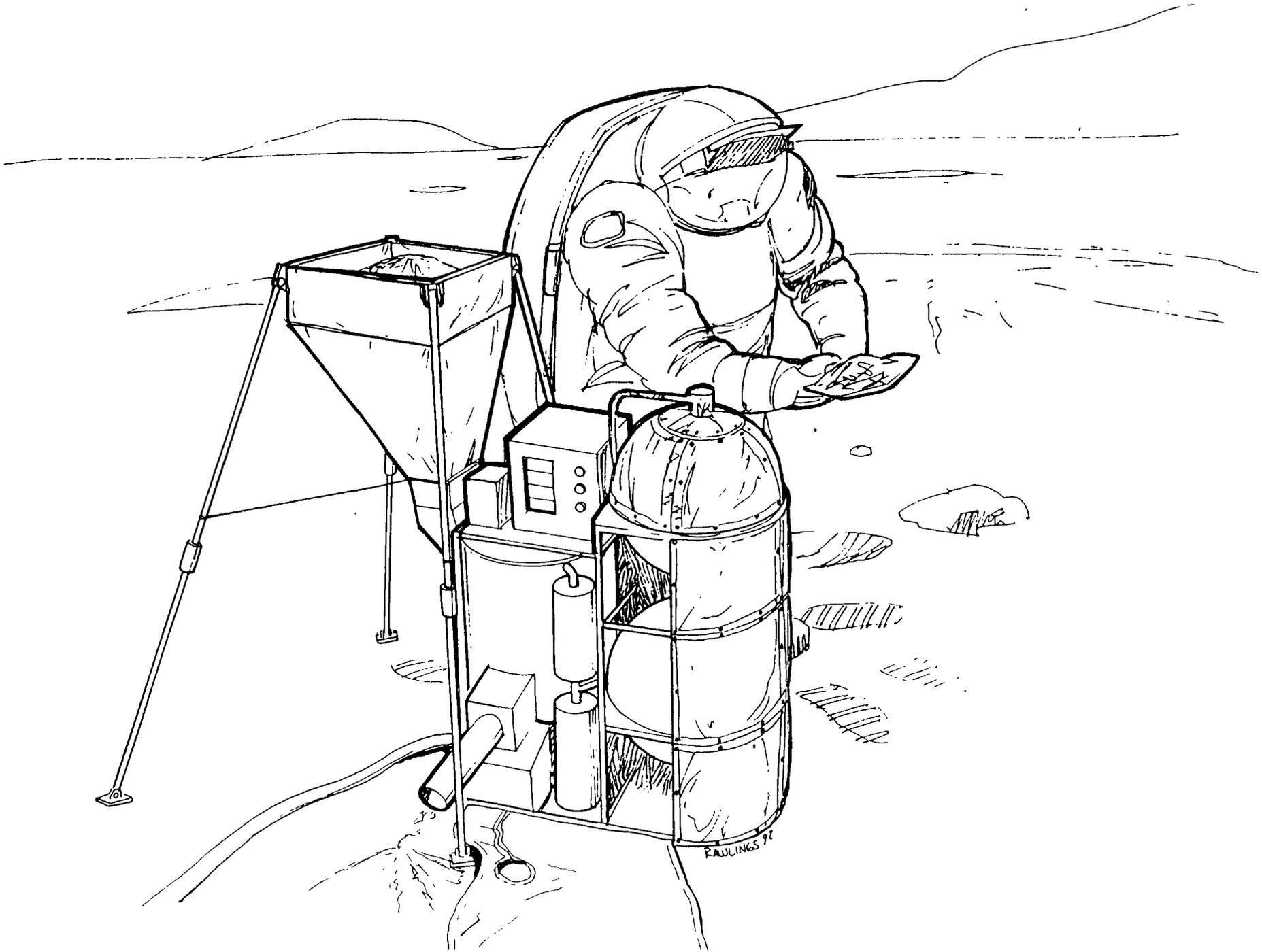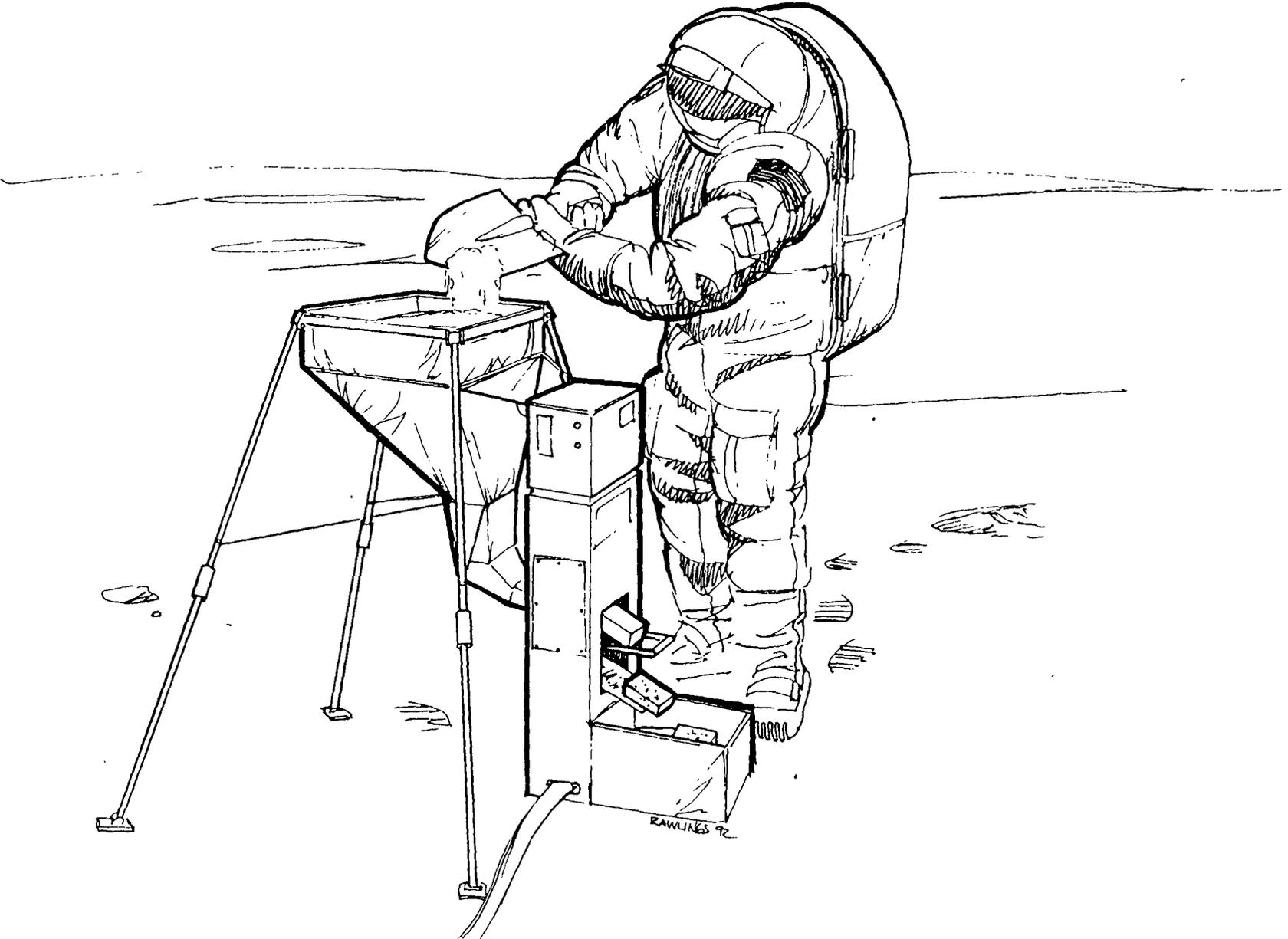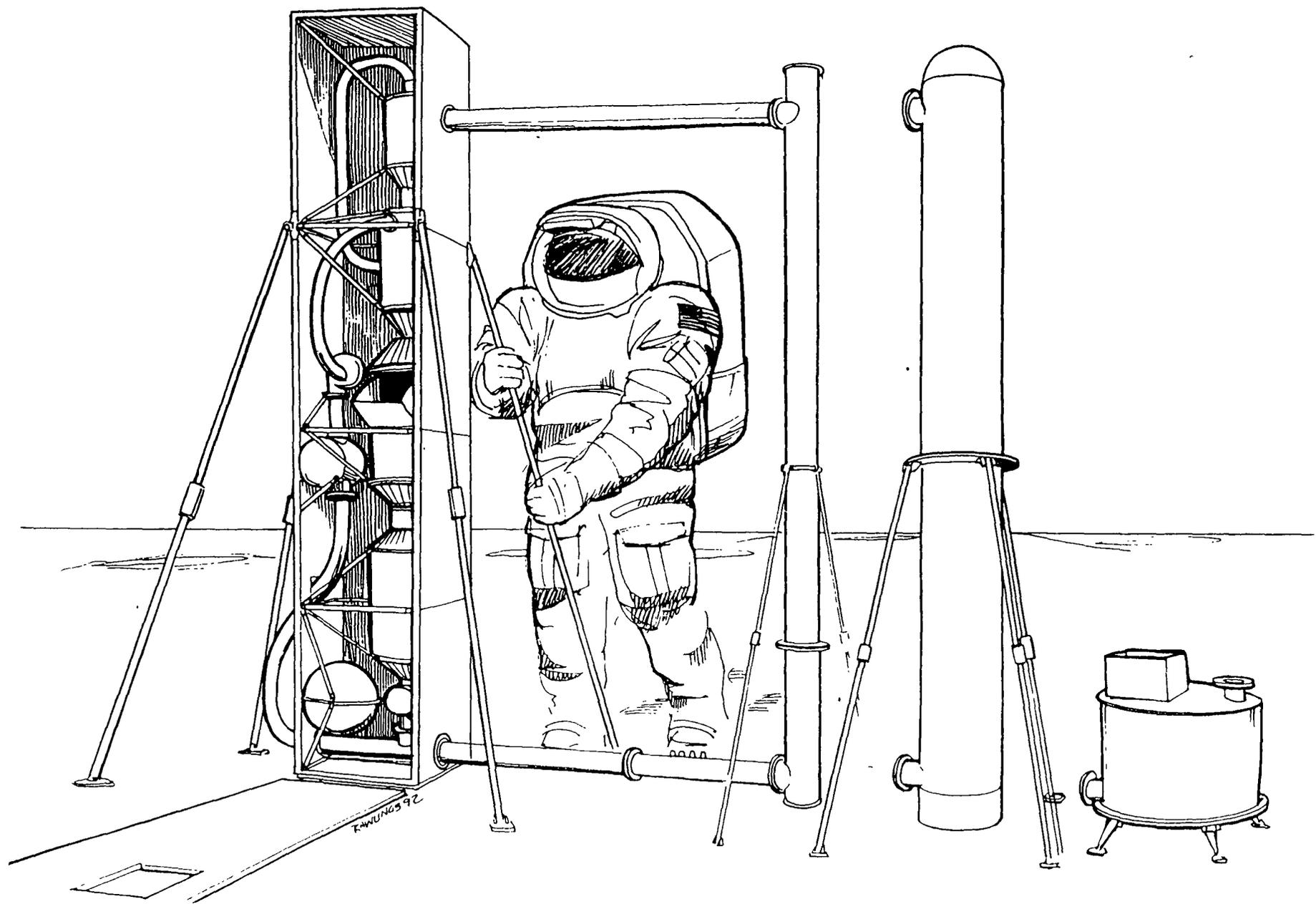
Science and Payloads Team

IVA Science

- Will include basic analysis, sorting, and packaging of samples for return to earth, including lunar samples and biosamples

- Will include gravitational biology experiments
    - Mutagenicity of cosmic radiation
    - Chloroplast movement

- Will include physiological experiments
    - Central nervous system
    - Thermoregulation
    - Body fluid analysis

Science and Payloads

Robotic Package for Rover Operations:

Current concept is an add-on package for manned rover which will allow robotic (mainly teleoperated) operation

- **Provide capability for continuing robotic activity between missions**

  - **Allows for teleoperation from outpost or earth**

  - **Includes manipulator arm with ability to dig, scoop, and sample**

  - **Contains set of basic analysis instruments**

- **Explore out to 100 km from outpost and return**

  - **Return samples to outpost for collection, analysis and return**

  - **Return samples from beyond human range for ISRU demo feedstock**

  - **Up to 4 round trips per year (outpost site to 100 km and return)**

Science and Payloads Team

Strawmen Science Payload Summary (kg)

Discipline	EVA	IVA	Total	Percent
Geophysics	600		600	18
Space Physics	200		200	6
Lunar Geology	400	50	450	13
Astronomy	530		530	16
ISRU	700		700	21
Rover Robotic Package	300		300	9
Life Science	200	400	600	18
Total	2930	450	3380	100

Science and Payloads Team

- Appropriate mix of science among represented disciplines
- Use of EVA time and capability vs IVA time and capability
- Demo and validation of ISRU vs. use of products on FLO
- Initial evolution of science
- Initial evolution path for ISRU
- Rover range vs. assured crew walkback
- Frequency of EVA: every day using alternating crews vs every third day with all crew members staying in habitat every third day
- Radiation storm shelter using local materials vs transported materials
- Effect of site selection on science payloads and activities

Science and Payloads Team

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Science on the second mission:

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1. Continue original detailed reconnaissance traverses not completed on
   first mission.

   • Estimate that 5 or 6 of 9 planned traverses at Smythii would be completed
     on first mission leaving 3 to 4 for second mission.

   • This would complete the mapping of the general geology and chemistry
     of an area about 20-25 km diameter around the outpost.

2. Begin a detailed drilling program with 10 meter drill (300 kg) using data
   from the traverses to determine optimum drilling locations within 20 km
   of outpost. Number of drill holes would be determined by EVA time available.

Science and Payloads Team

Science on the second mission (continued)

3. Bring and deploy initial elements of radio telescope array (300 kg).

4. Revisit optical telescope site and switch detectors as an operational test.

5. Analyze and sort samples returned by robotic science rover between missions.

6. Perform additional life science experiments with hardware on hand.

Science and Payloads Team

Science Evolution:

Over a series of missions to the same outpost, including the transition to permanent occupancy, science studies might be expected to evolve in the following ways:

Astronomy:

- Evolution will be toward bigger observatories, more observatories using different wavelengths (UV, IR, X-Ray, radio frequencies, and visible)

- Radiotelescopes will be added and array size can be increased

Space Physics and Geophysics:

- Evolution will be toward network emplacement to provide information on variations of many properties from place to place.

- Instruments may not change much or grow significantly larger

Life Sciences:

Additional and more elaborate experiments would be delivered

FLO Science and Payloads Team

> **Geology and geochemistry:**
>
> **Evolution may be in two directions:**

1. A need for broader coverage of the lunar surface may require *longer and longer traverses* which will in turn require a pressurized rover.

   (An alternative approach is to use *Sorte* missions instead of very long traverses).

2. *Much more detailed studies* in close to the outpost may be undertaken.

   • May include trenching into the regolith as deep as possible and in several directions.

   • Very detailed study of local features such as large craters, rilles, or volcanic vents might also occupy considerable time.

FLO Science and Payloads Team

ISRU Evolution:

Evolution will be toward producing larger amounts of oxygen, volatiles, and construction materials which can be used to support the outpost and the transportation system.

- Early production plant in 4-5 years with capacity of 10-15 MT oxygen/yr

- First real use of oxygen or water in infrastructure and outpost support

- Full production plant with capacity 100 MT/yr (8-10 years?)

- Transition to heavy use of lunar propellant and volatiles in infrastructure and transportation support

FLO Science and Payloads Team

General: Evolution Philosophy:

- Allow payload room for new experiments

- Plan for new and different activities

- Stay flexible enough to take advantages of new discoveries

Science and Payloads Team

What are we doing now?

• Refining concept design and costing of strawmen payloads

• Iterating timeline planning and day to day activities

• Looking at synergy with robotic orbiter and Artemis program

• Working on design concepts and operations concepts for rover in robotic mode

• Planning science evolution

• Planning ISRU evolution

• Designing technology development plan for ISRU

• Preparing to plan for technology development for other science payloads

Science and Payloads Team

Where do we go from here?

• **Startup of development programs for science and ISRU payloads**

• **Organize workshops and committees on science priorities and site selection**

• **Organize mapping and mission planning activity**

• **Write plan for payload command and control and data flow and analysis**

• **Write plan for science management**

  • **Payload selection and integration**

  • **Payload command and operation**

  • **Data flow and analysis**

• **Prepare detailed science and ISRU evolution trees with options, waypoints, and decision points leading to additional lunar-based science and to Mars science and ISRU activities**

Space Transportation

Presentation to

Technical Interchange

5/5/92

Ron Kahl

- Design activity is an on-going requirements development process that will progress through numerous iterations before final selection of technical approach

- Current FLO concepts provide a framework for developing and testing requirements, the concepts are a "first cut" that will be refined considerably as analysis proceeds

- In Summary:

  - These approaches are not final and others have not been ruled out

  - Additional concepts, approaches, and issues will be identified and assessed

  - Input from the SEI community has been and continues to be valuable

  - Interim status reviews will continue as FLO products mature

- Introduction

- Requirements

- Concepts

- Issues

- Space Transportation/Lander Scope
  - Launch Escape System
  - Crew Module
  - Return Stage
  - Lander for Piloted and Cargo Mission

- KEY Interfaces
  - Earth to Orbit Launch Vehicle and Trans Lunar Injection Stage
  - Surface Systems
  - Habitat
  - Operations
    - Launch
    - Flight
    - Communications and Navigation
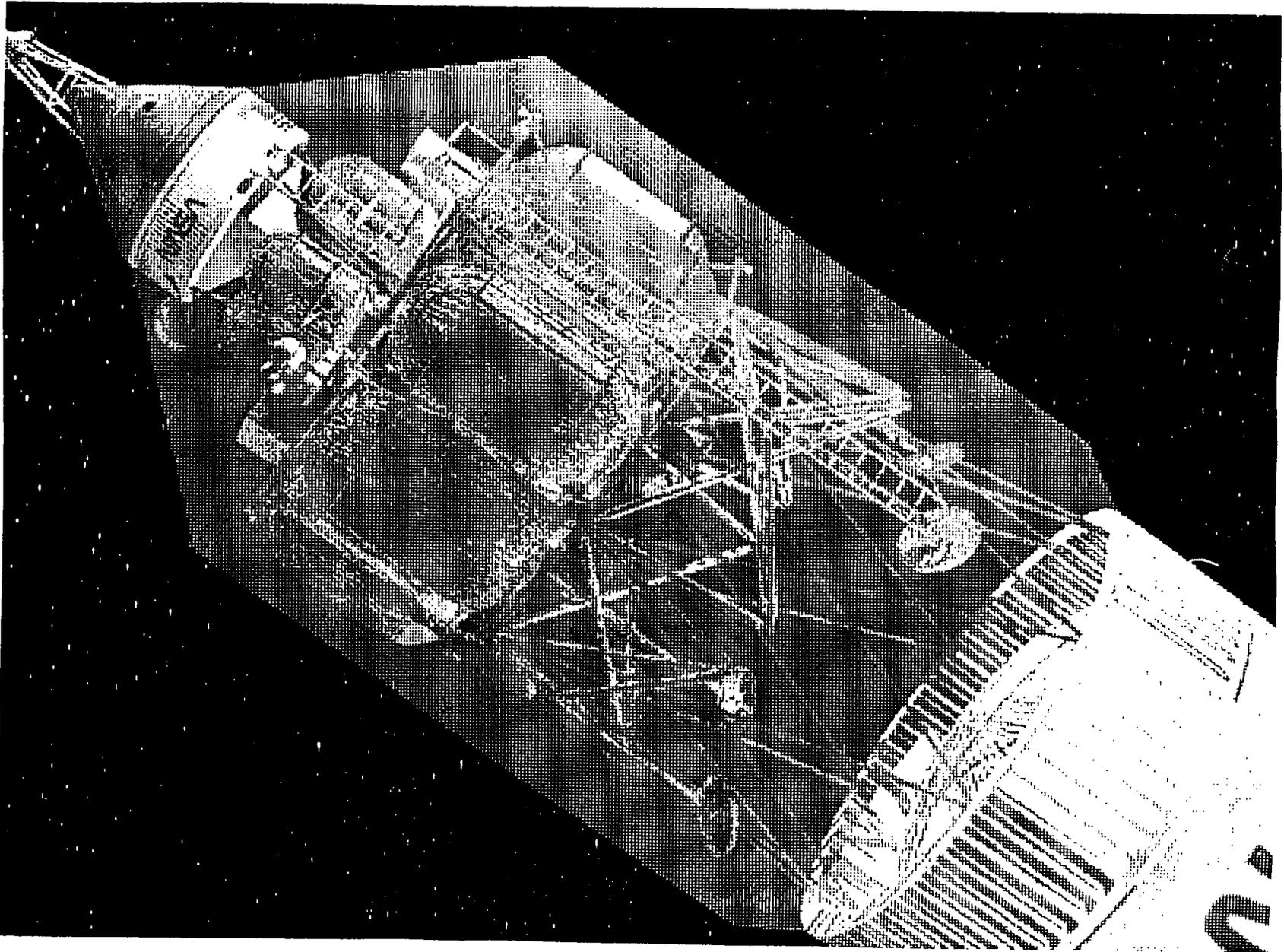
Requirements

- First Flight in 1999 - Schedule

- Anytime abort

- Crew size of 4

- 45 day stay on Moon

- Land landing at Earth

- Return Payload 200 Kg

- 5 tonne resupply on piloted mission

Design Guidelines and Assumptions

- Use existing hardware where appropriate - Cost/Risk

- Cryogenic Lander/Storable Return Stage

ETO Interface Assumption

- Shroud 10m dia. x 18m length - Envelope

- Apollo Capsule Shape, upscaled 5% for larger Crew and EVA Suits
- Lunar Landing site redesignation capability (50 meters at 100 meters altitude)
- Lunar Landing accuracy 100 meters
- 8.5 days transit time, 2 days occupancy and 43 days dormant on surface
- Suits for EVA transfer of Crew
- 200 Kg. Return Cargo
- Thermal Protection System Ablative plus Tile
- RCS for return flight and earth entry
- Power for Earth Entry and Landing
- Parachute and Retrorocket Deceleration System
- Couches and Attenuation
- Land Landings at Earth

Mass Summary (Kg.)

Dry	**6,370**
Non-Cargo	609
Residuals + FPR + Fluids	82
Propellants	199
Gross	**7,260**
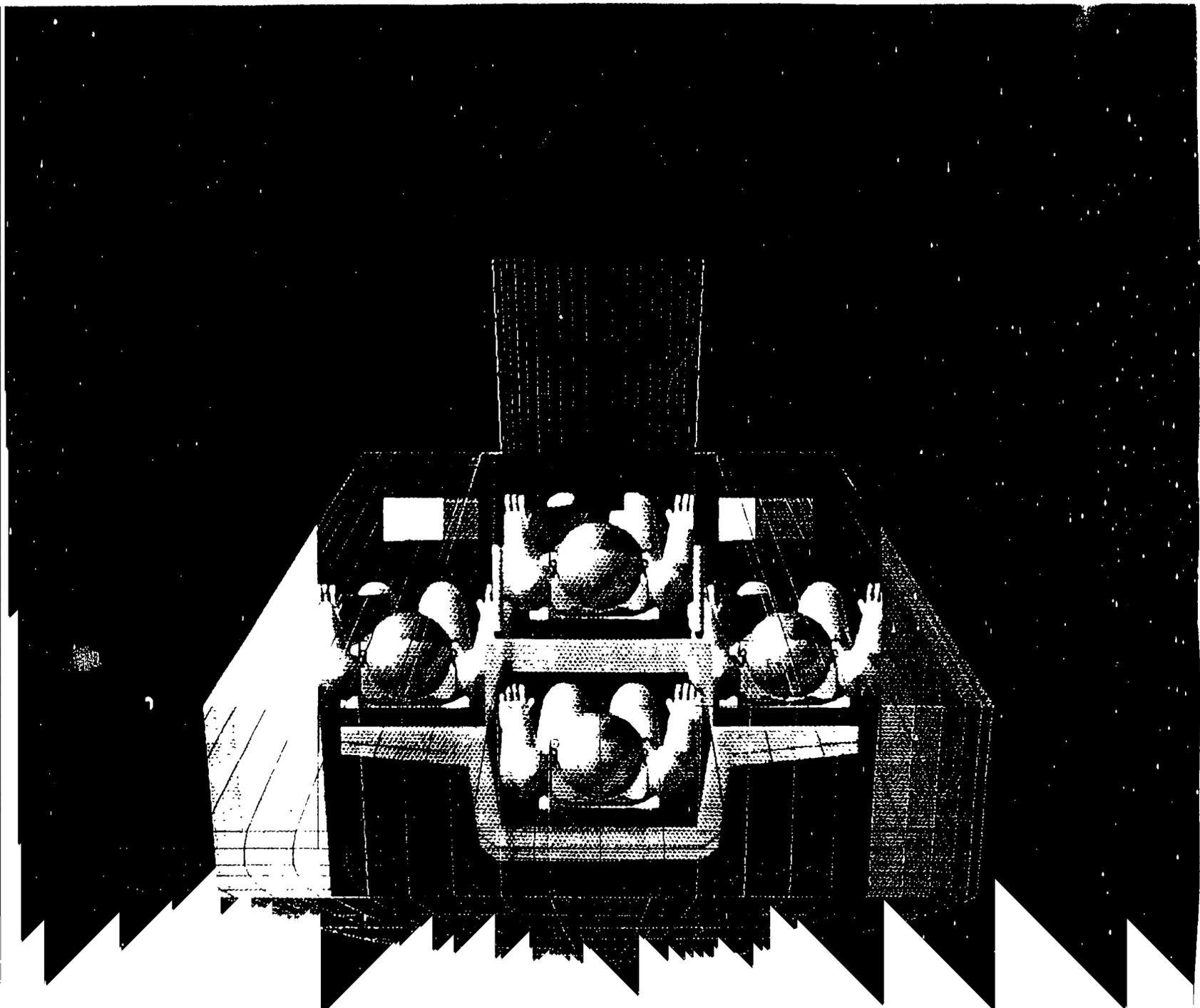
- Storable Propellants in Four Main Propellant Tanks

- Gaseous Helium Pressurization system

- Three Delta 2nd stage engines (320 ISP)

- Air Tank

- Three Fuel Cell Cryogenic storage tanks

- Three Fuel Cells

- Three Water Storage Tanks

- Radiator

Mass Summary (Kg.)

Dry	**4,469**
Residuals + FPR	543
Fluids	1,035
Propellants	18,077
Gross	**24,124**

124
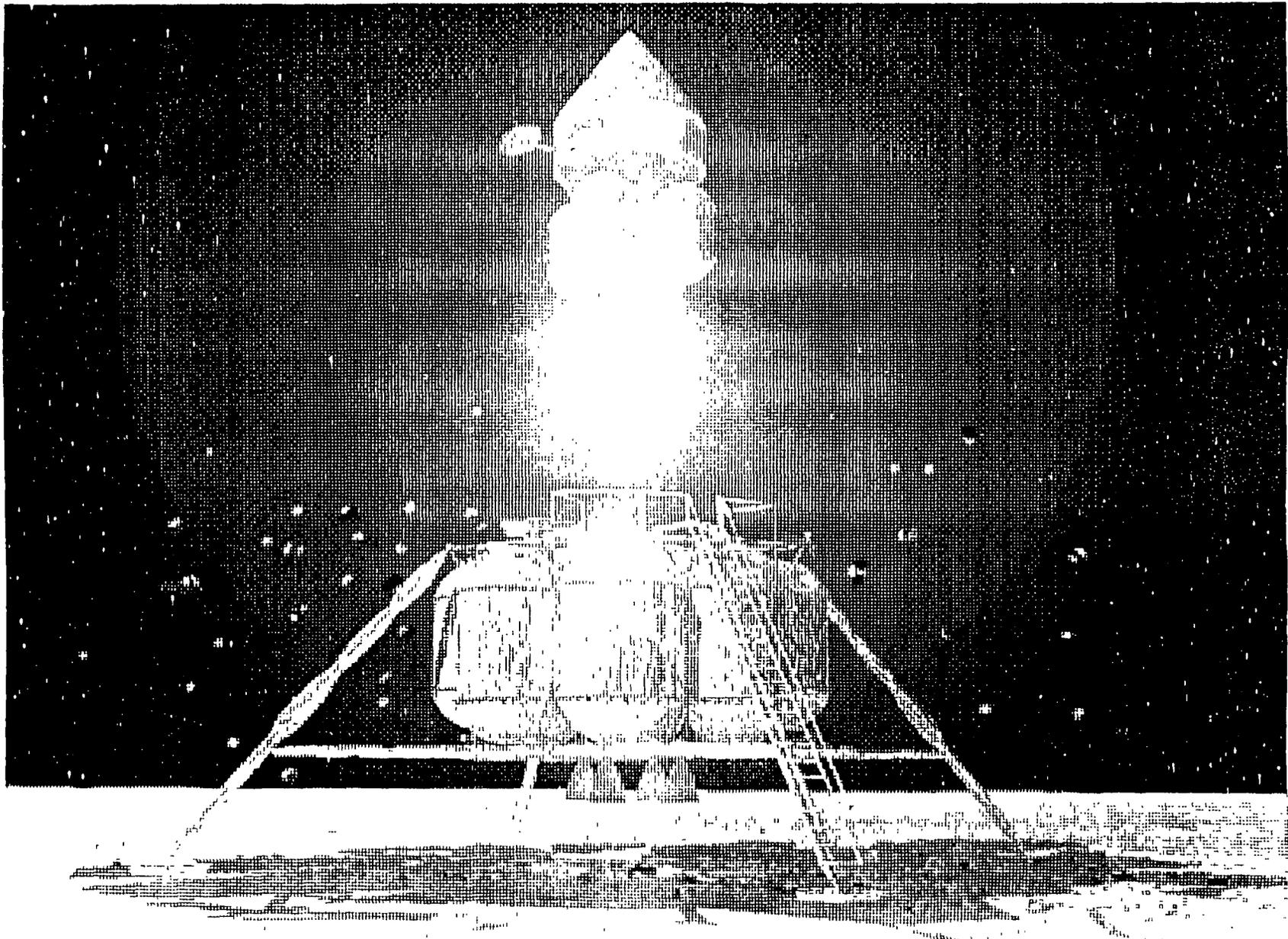
- Cryogenic Propellants in Eight Main Propellant Tanks
- Gaseous Helium Pressurization system
- Four RL-10 Derivative Engines (444 ISP) - Modified for (4:1) throttling
- Monopropellant Hyrdazine RCS
- Four Deployable Legs and Landing attenuation
- Stair/Rails for Crew Access to surface
- Landing accuracy 2 km in Cargo Mode

Lander Operational Configurations

Cargo Mode
- Habitat provides Power
- Avionics, DMS & Comm. added

Piloted Mode
- Return Stage Provides Power & Comm.
- Avionics in Crew Module

Mass Summary (Kg.)	Cargo	Piloted
Dry	10,868	10,660
Residuals + FPR + BO	1,732	1,732
Fluids & RCS	242	242
Propellants	43,954	43,954
Gross	56,796	56,588

Dimensions in meters

4.8

7.1

14.1

0.8

2.1

18.8
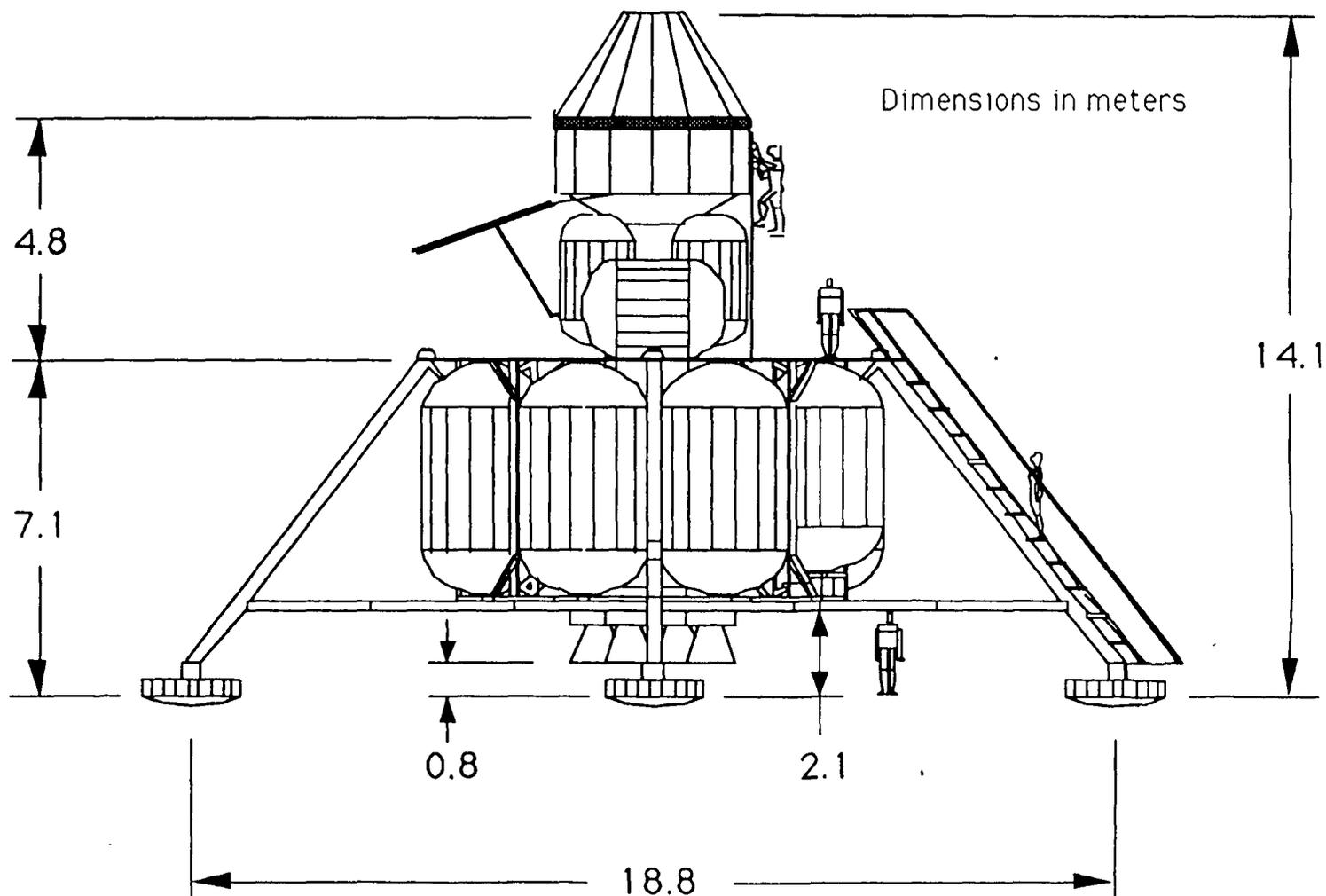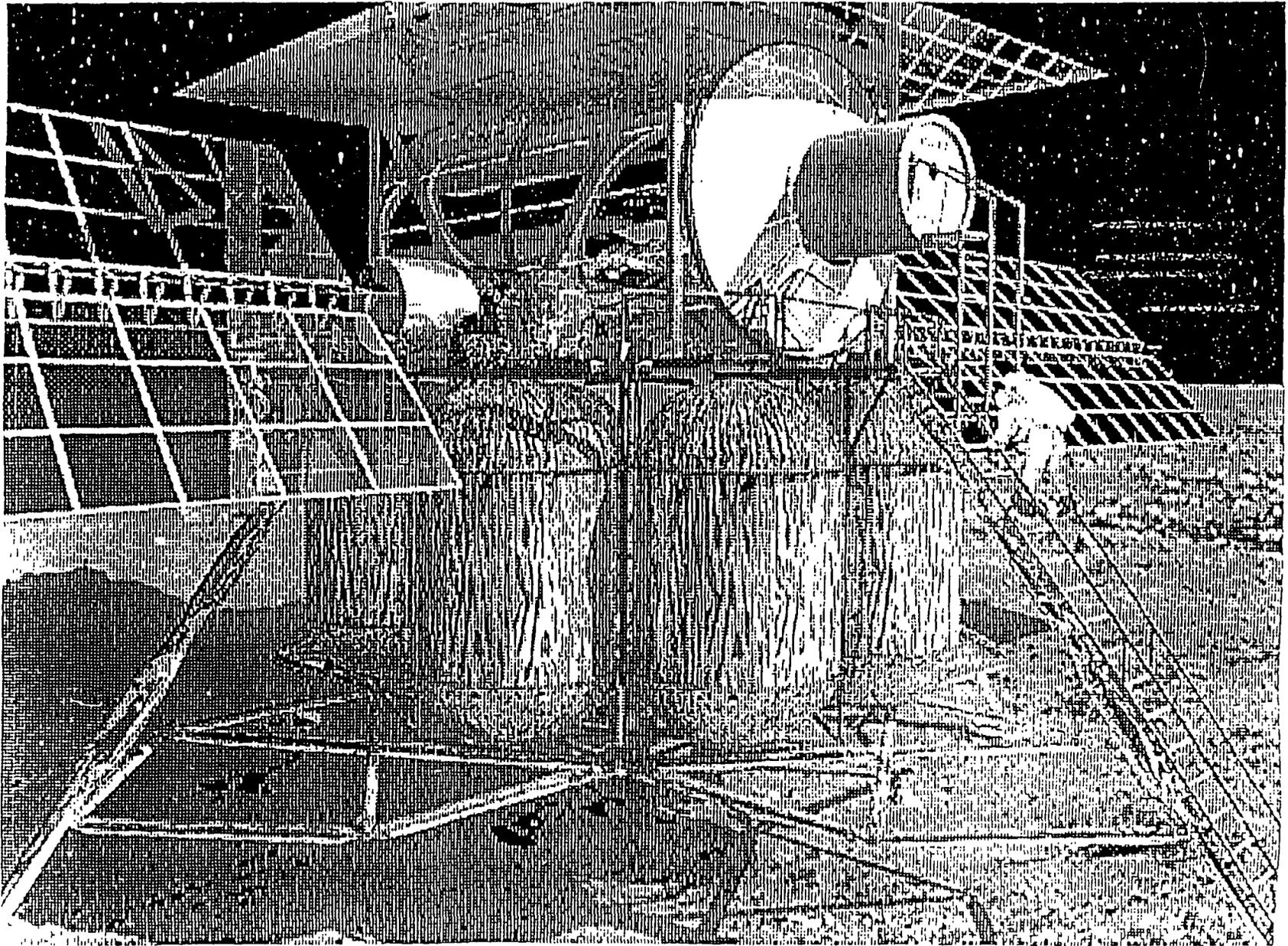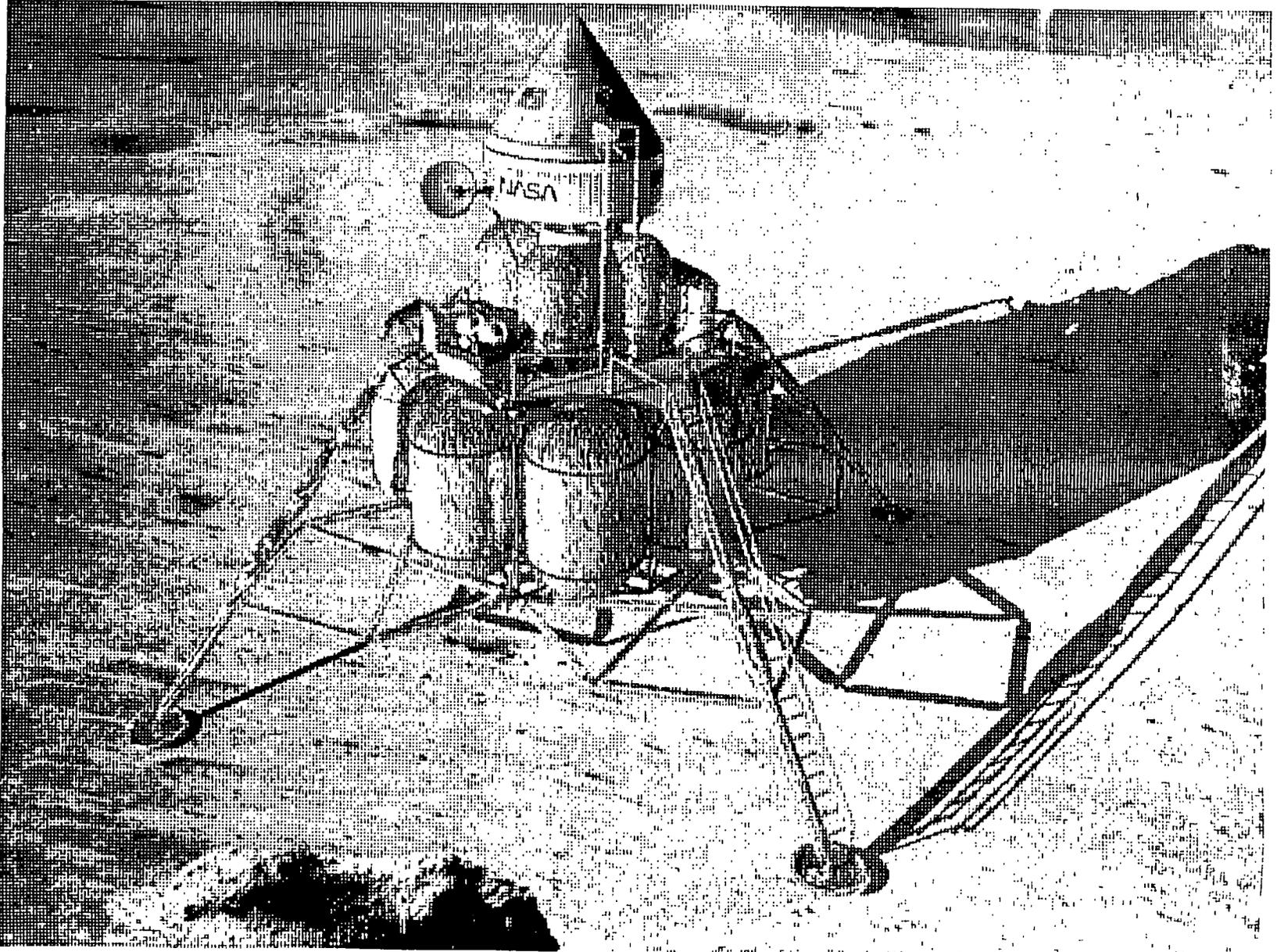
Space Transportation
Mass Summary

Mass Summary (Kg.)	Cargo Mission	Piloted Mission
Crew Module		6,370
Return Stage		4,469
Lander	10,868	10,660
Total Dry	**10,868**	**21,499**
Propellants	44,124	62,400
Residuals + FPR + BO + Fluids	1,804	3,464
Non-Cargo	0	608
Cargo	36,175	5,000
TLI Stage Adapter	2,789	2,789
Gross Mass at TLI	**95,760**	**95,760**

Design Issues

- System Mass
    - Lightweight Materials
    - Return Propulsion

- Land Landing at Earth (Reusability)

- Lunar Landing Accuracy/Hazard Avoidance

- Fire in the Hole

- Operations
    - Ship and Shoot
    - Recovery

- Radiation Protection

Surface Timeline

NASA

EXPLORATION
PROGRAMS OFFICE

LIGHTING ADEQUATE FOR LUNAR EVAs	SUN W/IN 20° OF VERTICAL — SURFACE TOPOGRAPHY WASHES OUT — RESTRICTED EVA OPERATIONS	LIGHTING ADEQUATE FOR LUNAR EVAs	AMBIENT LIGHT <1 lumen/m² EVAs DIFFICULT WITHOUT ARTIFICIAL LIGHTING	LIGHTING ADEQUATE FOR LUNAR EVAs	SUN W/IN 20° OF VERTICAL — SURFACE TOPOGRAPHY WASHES OUT — RESTRICTED EVA OPERATIONS	LIGHTING ADEQUATE FOR LUNAR EVAs

LUNAR SURFACE LIT SOLELY BY EARTHSHINE

LUNAR SUNRISE   LUNAR NOON   LUNAR SUNSET   LUNAR MIDNIGHT   LUNAR SUNRISE   LUNAR NOON   LUNAR SUNSET

Landing and Transfer | Outpost Startup | 3 Days | 4½ Days | 4½ Days | 6 Days | 14 Days | 6 Days | Outpost Shutdown | Transfer and Departure

Planned EVA Schedule:
(2 EVAs)   (3 EVAs)   (5 EVAs)   (12 EVAs)   (1 EVA)   (2 EVAs)

Expanded EVA Schedule:
(2 shortened EVAs)   (3 EVAs w/lighting)   (3 shortened EVAs)

6 Days of Work   7 Days of Work   6 Days of Work   6 Days of Work   6 Days of Work   7 Days of Work

- Exploration surface missions will be EVA-intensive in order to support the extensive science, exploration, and support activities planned.

- Zero-prebreathe EVA capability is considered mandatory.

  - This enables routine & contingency EVA operations from a pressurized enclosure (lander, habitats) without mission overhead impacts.

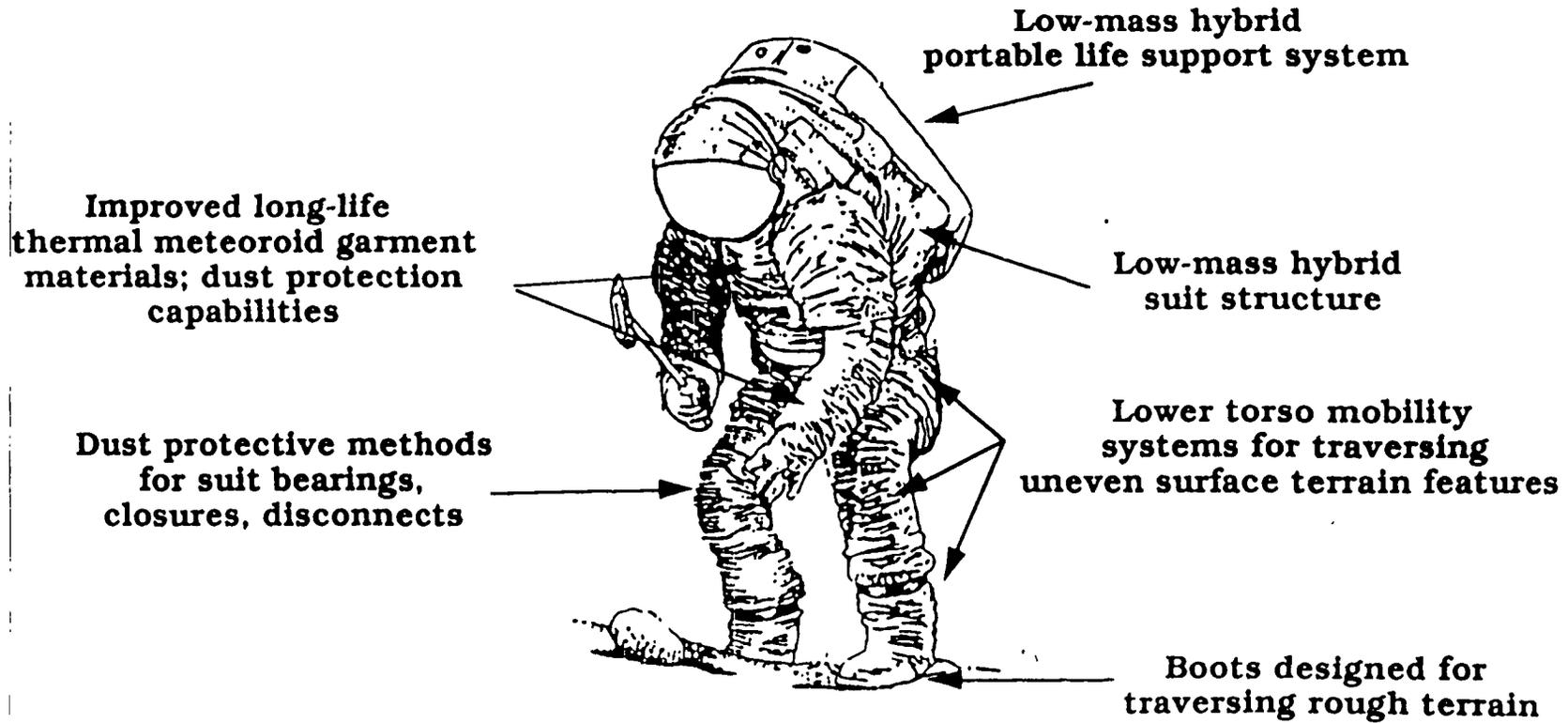 - It is necessary to select appropriate EMU vs. cabin pressure for physiologically acceptable bends risk (Example: 5.85 psia EMU from 10.2 psia habitat)

- New suits will be required which must provide improved mobility and which will require low maintenance.

  - Walking, climbing, bending at waist, sitting on rover will be difficult with Apollo-era suits.

  - Current zero-gravity, orbital based suit lower torso mobility features are not adequate for partial-gravity activities.

- Suits must protect crew during powered flight phases of mission

- Support transfer of the crew from lander to habitat and return

- Hard torso surface suits may be too bulky for use or transport within the crew module. A flight suit derived from the STS launch/entry suits could be used for transfer from lander to habitat.

- Design decision on suit implementation is not yet firm

Advanced EMU

EXPLORATION
PROGRAMS OFFICE

Low-mass hybrid
portable life support system

Improved long-life
thermal meteoroid garment
materials; dust protection
capabilities

Low-mass hybrid
suit structure

Dust protective methods
for suit bearings,
closures, disconnects

Lower torso mobility
systems for traversing
uneven surface terrain features

Boots designed for
traversing rough terrain

**NASA**   **EVA (cont.)**

EXPLORATION
PROGRAMS OFFICE

- Contamination protective measures will be required for connectors, bearings, mechanical linkages, visor optical surfaces, and airlock/habitat environments.

  - Apollo experience indicates that the severe dust environment will be the single most limiting factor to extensive long-term EVA operations

- Surface exploration EMU design is based on highly mobile, lightweight systems and long-life, simple portable life support system technologies which require low maintenance by the mission crew on-site.

  - Minimize crewmember carry-weight in partial gravity
  - Reduce/eliminate need for resupply expendables by means of regenerable techniques or adequate available supply source (LOX)
  - Capable of routine recharge, service, and maintenance by the mission crew far removed from Earth-based facilities

- EVA/EMU servicing and recharge capabilities will be provided from:

  - Habitat airlock EVA accommodations
  - Rover add-on EVA accommodations

National Aeronautics and
Space Administration

NASA   Rover

- Surface transport is required for crew and logistics transfer, scientific payload emplacement, and geological traverses.

- Unpressurized rover concept is similar to Apollo LRV, but with greater performance and payload handling capability.

- Preliminary requirements indicate capability to carry crew of 4, or crew of 2 and payload of up to 500 kg.

- The rover will be powered by regenerative fuel cells or batteries, which are recharged at the habitat.

- The rover will also support EVA add-on packages for long-duration traverses. This will allow full-duration EMU life support capability for an emergency walk-back to the habitat.
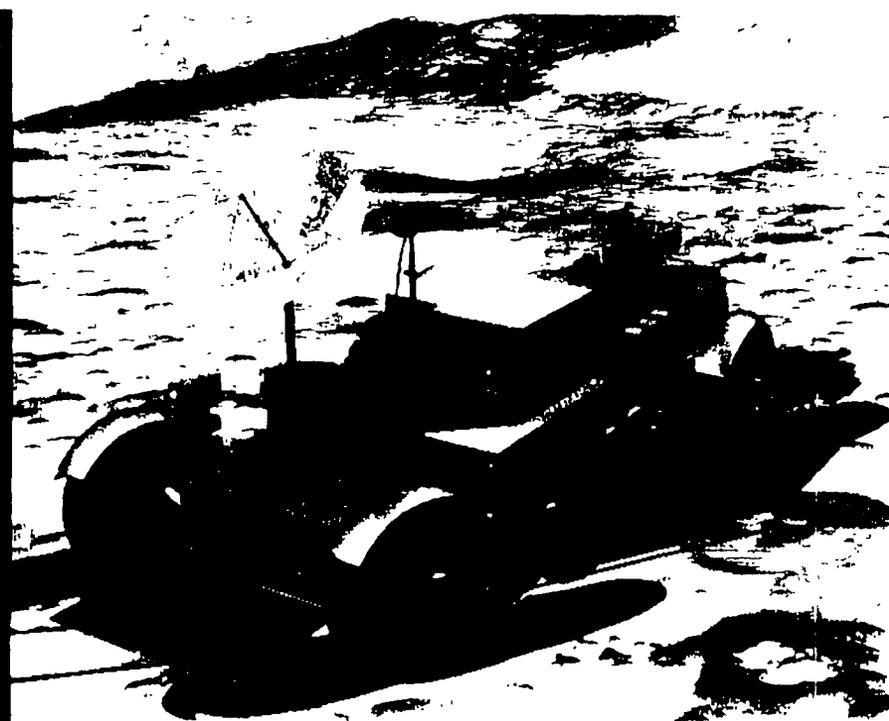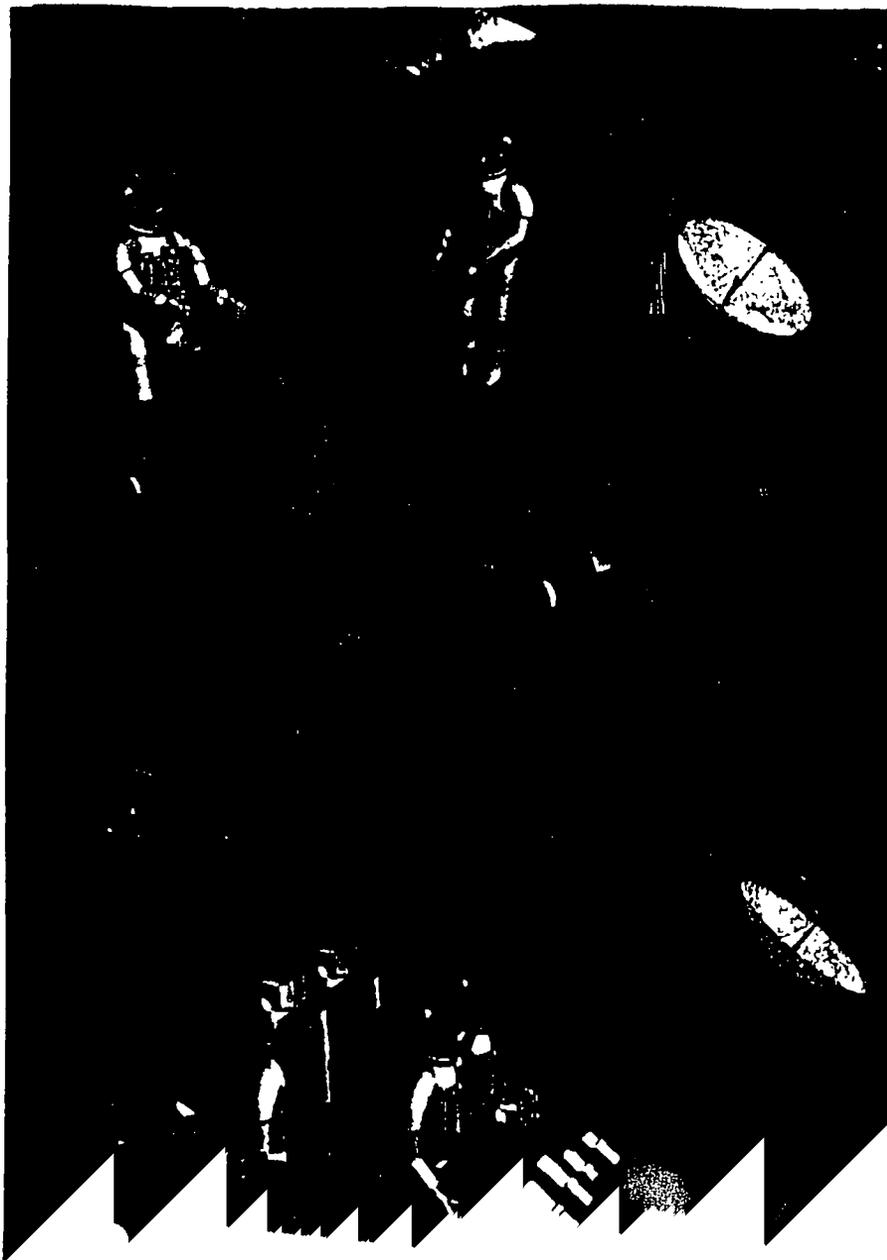
UNPRESSURIZED ROVER

Operating Modes:
Manned
Unmanned (Teleoperated)

Capacties:
4 crew
2 crew + 500 kg payload
0 crew + 1000 kg payload

Traverse Capabilities:
Manned - 50 km
Unmanned - 200 km

Speed:
15 km/hr (max) - smooth
4 km/hr (max) - rough upland
7 5 km/hr (avg)

Packaged Volume: (E)
1.5 m^3

Mass:	
	0.5 mt
20% Growth	0 1
15% ASE	0.075
Total	0 675 mt

Subsystem Mass: (E)	
Chassis	69 kg
Mobility	127 kg
Crew Station	20 kg
Navigation	14 kg
Communications	45 kg
Payload Support	10 kg

Surface Systems Splinter Session

ExPO Technical Interchange Meeting

An exchange of ideas on FLO Surface Systems

Wednesday, May 6
11:00 a.m. - 1:00 p.m.
Bertolotti's Cuchina Italiana

2555 Bay Area Blvd.
(just west of UH-CL, seating may be limited)

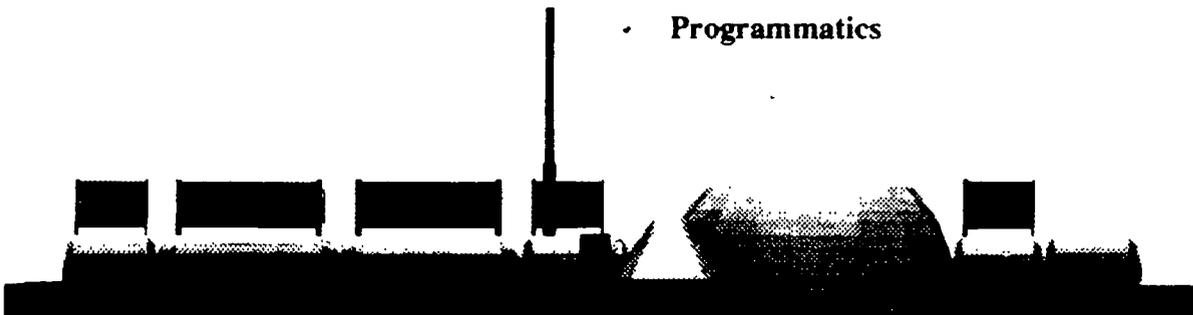Luncheon meeting   No formal presentations   Each table will have a PSS facilitator

Topics
- Surface Systems
- Surface Systems Integration
- EVA
- Rovers
- Logistics
- Programmatics

First Lunar Outpost

Lunar Habitat
Concepts and Issues

Third Space Exploration Initiative
Technical Interchange

Molly Elrod
Program Development
NASA/Marshall Space Flight Center
May 5, 1992

N92-33318

- Flight Elements Shall Provide for a First Launch as Early as 1999

- Initial Design for a Mission Capability is a Crew of Four and a Lunar Surface Stay of 45 Days (lunar day-night-day)

- Initial Mass Requirement is 25 mt. (**Current Assessment is 31 mt**.)

- Existing Hardware is Utilized where Practical (As determined by cost, schedule, risk, and performance impacts and savings)

- Specific Design Goals are:
    - Reuse every six months
    - Provide solar flare protection
    - Enables manned EVA activities
    - Capable of operation anywhere on the lunar surface
    - System test occurs before crew launch
    - Crew surface operations begin within 24 hours of crew arrival
    - Access to all hardware to provide for infinite life
    - Design for growth capabilities

SSF Habitat Module as Lunar Campsite?
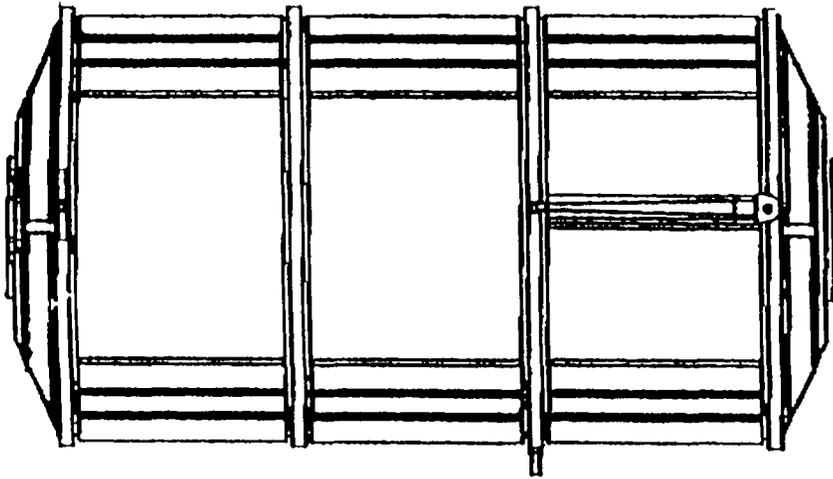
SSF Habitat Module Contains:

* Pressure Shell & Structural Support
* Galley
* Housekeeping Equipment
* Storage
* ECLSS/ Thermal
* Power Distribution
* Wardroom
* DMS/Communications
* Hygiene Equipment

Additional Equipment Required:

External
  * Energy Source, Converters, & Conditioning
  * External Structures
  * External Fluid Storage
  * Radiators
  *Communications

Internal
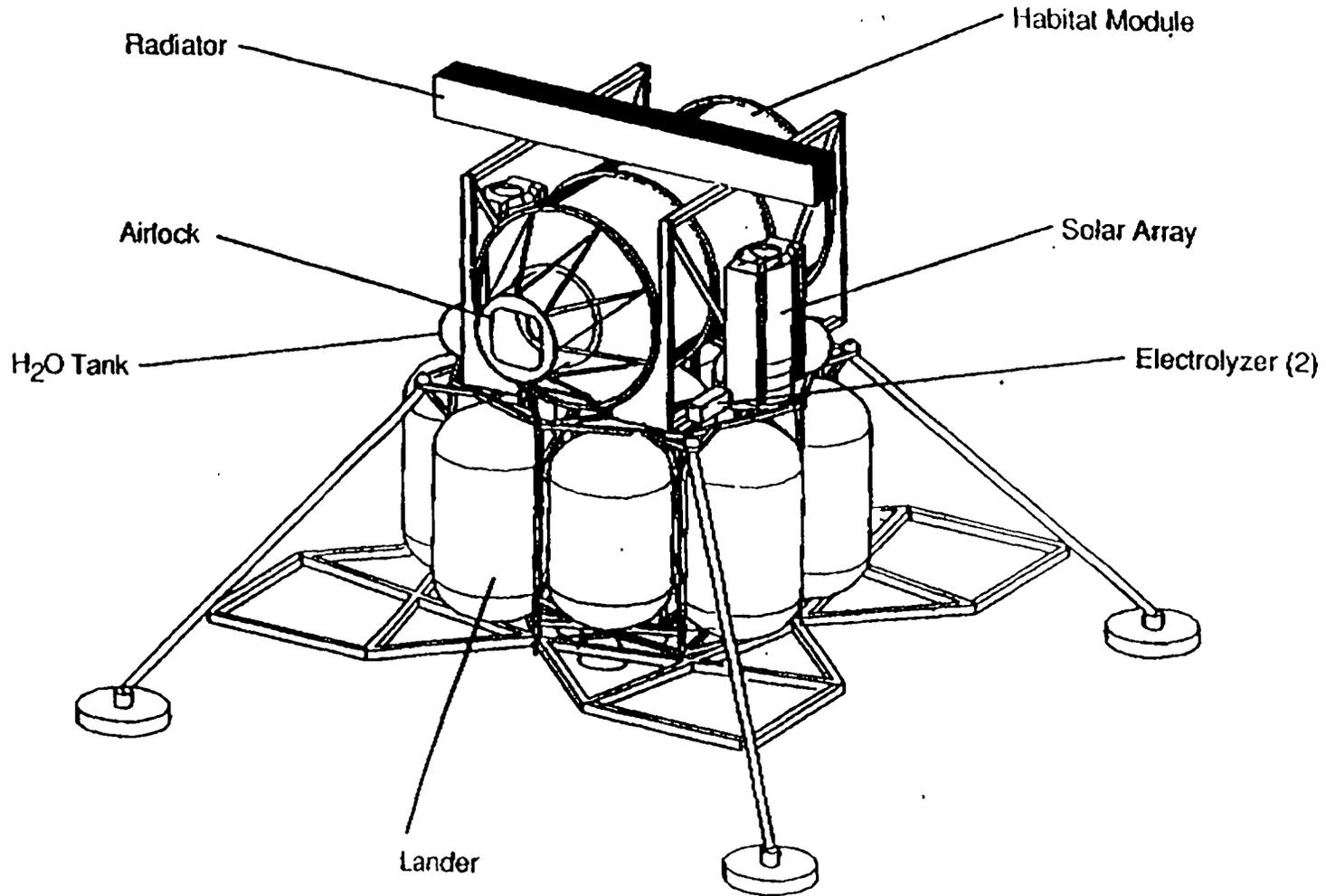 * Airlock/EVA Support
  * Maintenance/ Science Support
  * Communications/ DMS Equipment
  * Radiation Protection
  * Medical Capabilities

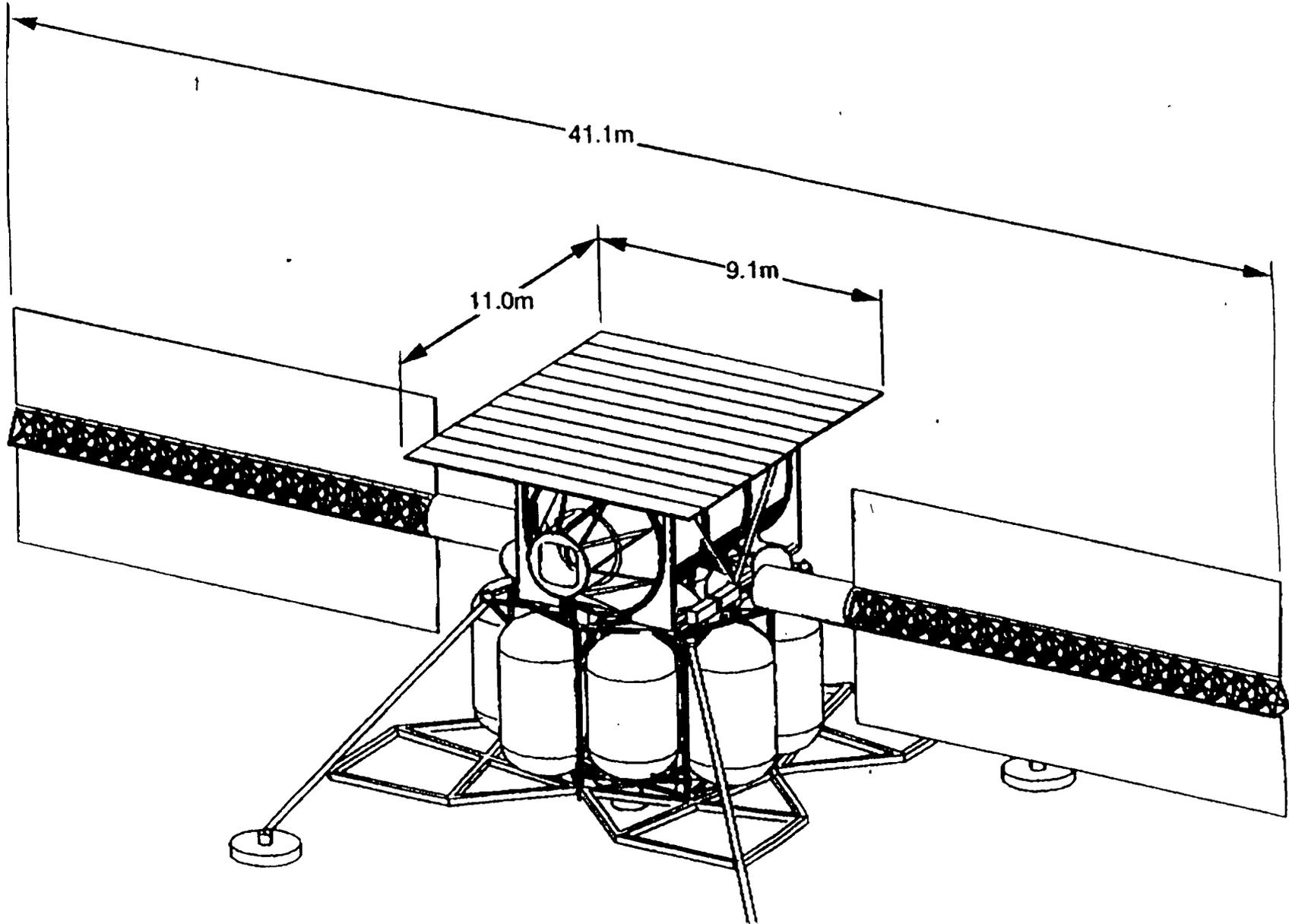Internal Equipment Removed/Reduced:

* Galley/ Wardroom
* Hygiene Equipment
*ECLSS

Lunar Habitat
Stowed Configuration

Radiator

Airlock

H_2O Tank

Habitat Module

Solar Array

Electrolyzer (2)

Lander

1-1324-2

Lunar Habitat Space Station Derived

Habitation Module Layout

Depress Pump/EVA Storage	EVA Storage	Galley	ARS (open loop)	ARS (open loop)	Av Air/ Crossover

Ceiling

AIRLOCK

SPCU	Science Glovebox	Water Storage & Processing	Waste Mgmt/ Hygiene	Urine Processing/ ARS/ACM	Crossover/ TCS/ Cabin Air

Wall

SPCU/ Airlock Control	Hyperbaric	Galley	Science/ DMS/ Comm Workstation	Medical/ CHeCS	Crossover/ TCS

Storage

Wall

EVA Storage	EVA Storage	Galley Storage	Science Storage	Personal/ CHeCS Storage	Critical ORUs Storage

Floor

Does Not Include:
Crew Quarters
Dedicated Wardroom (3)
Refrigerators/Freezers (1.5)
Trash Compactors (0.5)
Dedicated Shower (1)

Reduces:
SSF Galley Complement (3)
ECLSS (2)

Exchanges 11 Racks in SSF Hab A

Adds:
SPCUs (2)
Airlock Support (2)
EVA Stowage (3)
CHeCS/Medical (1)
Science Glovebox (1)
Science Workbench (1)
Science Storage (1)

Expands:
DMS/Comm

M. Elrod
5-4-92

	(kg)
Structure and Subsystems	
Module	7302
External Structure	1614
ECLSS	2656
Medical Support/Life Science	445
Crew Systems	1294
Storm Shelter	1000
CDMS	863
Power System	3461
Thermal Control System	1990
Airlock System	4236
Subtotal (Habitat)	24,861
Contingency	2,486
Total (Habitat)	**27,347 kg**
Consumables	1506
Fuel Cell Reactants	1811
EVA Suits	635
Science Equipment	62
Total (Mass to the Surface)	31,361 kg

Blood Forming Organ and Skin Dose Equivalent Comparison for Shelter Concepts A and B

Shelter Concept A range
Shelter Concepts B range
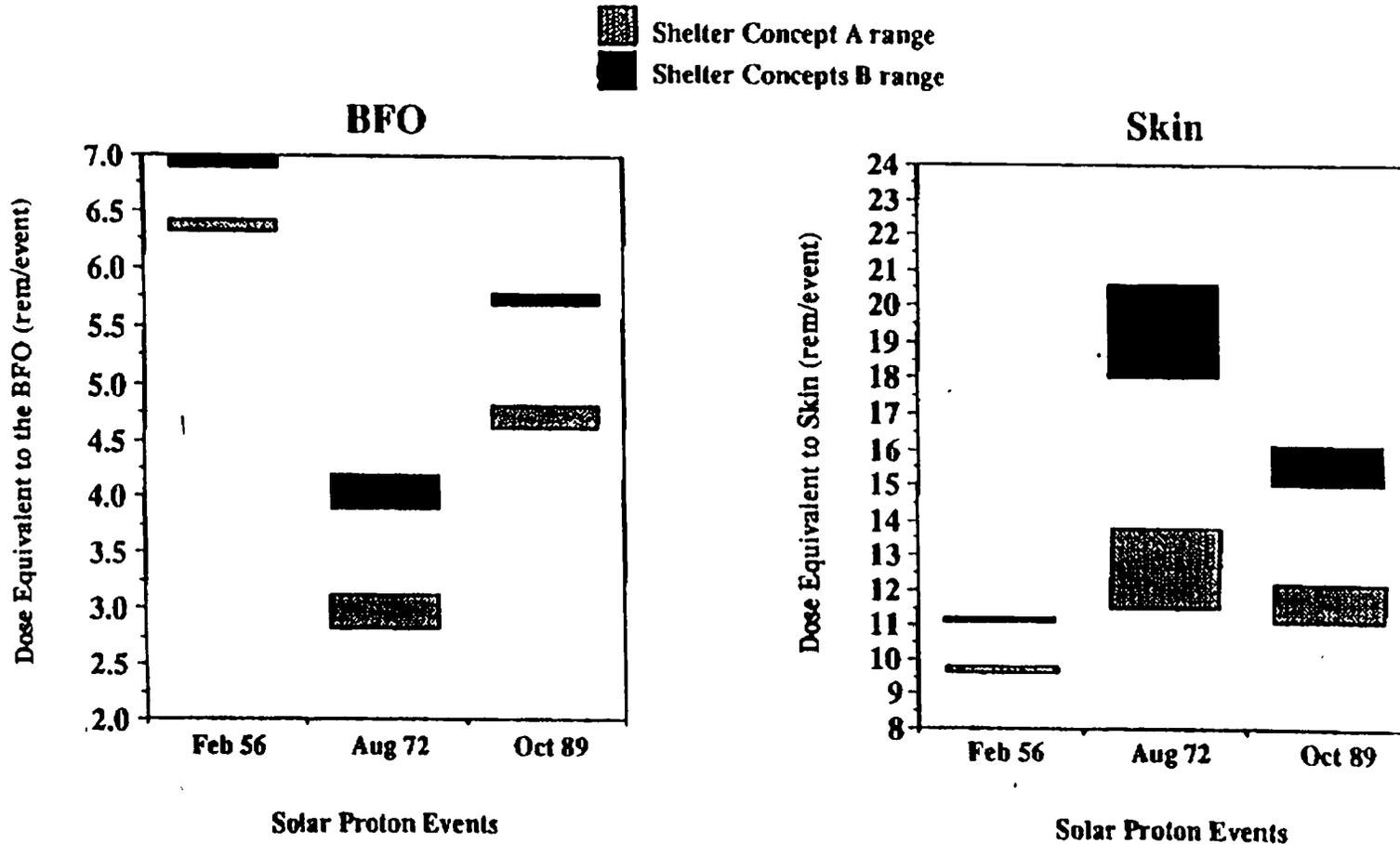
Dose equivalent rates determined using the Computerized Anatomical Man Model (CAM)

- A Feasible Habitat Has Been Defined That Can Be Emplaced In One Launch

- Habitat and Crew Module Utilize A Common Lander

- Habitat Is Derived From SSF Hardware (70% By Weight)
  - Module
  - Crew Systems
  - ECLSS
  - Data Management System
  - Internal Thermal Control System
  - Airlock

- Major Development Item Is the Power System (PV/RFC)
  - Lifetime
  - Reliability
  - High Pressure Tankage

- **Configuration Options**
  - Space Station Freedom Module
  - Clean Sheet
  - Inflatables

- **Design Definition and Assessments**
  - Detailed Structural Assessment
  - Systems Definition
  - Radiation Shielding Definition
  - Internal Configuration Layouts

- **Consumables Stowage/Transfer Logistics**

- **Growth Scenarios**
  - Joining of Multiple Habitats
  - Offloading and Burying of Habitats

- **Operational Issues (maintenance/dust mitigation)**

- **Airlock Options/Definition**

- **Integration of Lander and Habitat**

- FLO activity is an on-going requirements development process that will progress through numerous iterations before final selection of technical approach

- Current FLO concepts provide a framework for developing and testing requirements, the concepts are a "first cut" that will be refined considerably as analysis proceeds

- In summary:

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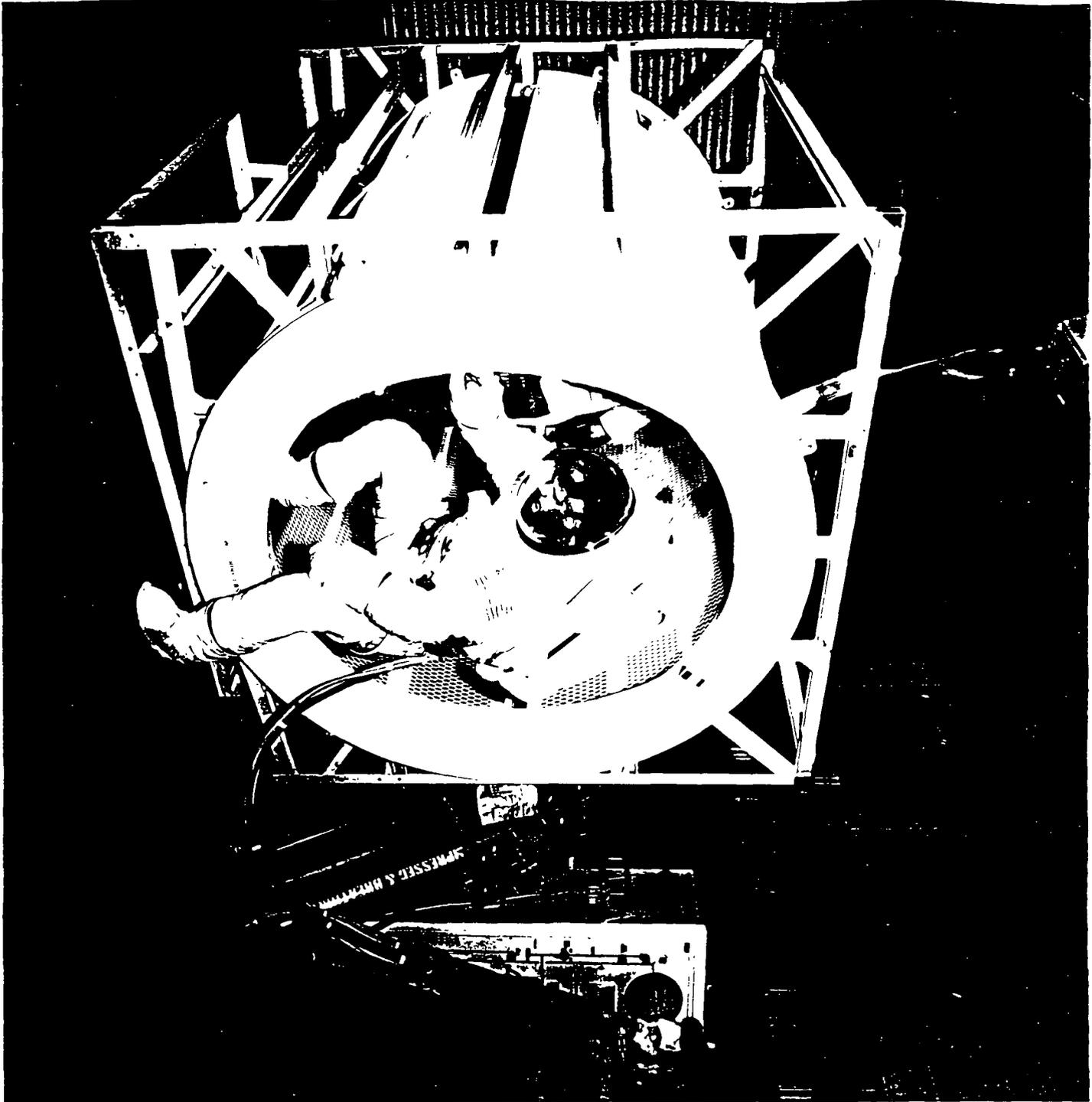
149 A.

First Lunar Outpost
Earth To Orbit Concepts
And Issues

Third Space Exploration Initiative
Technical Interchange

At The

University Of Houston - Clear Lake

Gene Austin
May 5, 1992

Mission Design Choices:
Implications to HLLV

- Lunar Direct Vs. Lunar Orbit Rendezvous

  - Lunar Orbit Rendezvous Mode; Provides Earth Return Opportunities Every 14 Days; Introducing Risk For Contingencies

  - √ **Lunar Direct Mode**: Carrying Earth Entry Heat Shield To Lunar Surface

- Earth Orbit Rendezvous Vs. Single Launch

  - Earth Orbit Rendezvous: Dual Launch Costs And Operational Complexity With A One Day Lunar Launch Window For Assembled Vehicle On Orbit .

  - √ **Single Launch**: One Flight For Cargo And One Flight For Piloted Missions Permits Mission Design Flexibility And Reduced Operations Cost/Risks.

Requirements
ExPO

- The Earth to Moon Transportation System (HLLV, TLI Stage, Lander) Shall Provide the Capability to Emplace 27.5 t (Including A 10% Manager's Reserve) on the Lunar Surface in a Single Flight. (**Current Assessment Is 34t Resulting In A 93t Requirement To TLI)**

- A Single HLLV Shall be Utilized for Each Flight to the Moon

- Flight Elements Shall Provide the Capability for a First Launch as Early as 1999

- The Capability Shall be Provided to Support Up To Four HLLV Flights per year

SEI Reference Launch Vehicles

Saturn V Derived

NLS Derived

408 ft

348 ft

1 J-2S

6 J-2S

5 F-1A

2 F-1A

1 SSME

4 STME

2 F-1A
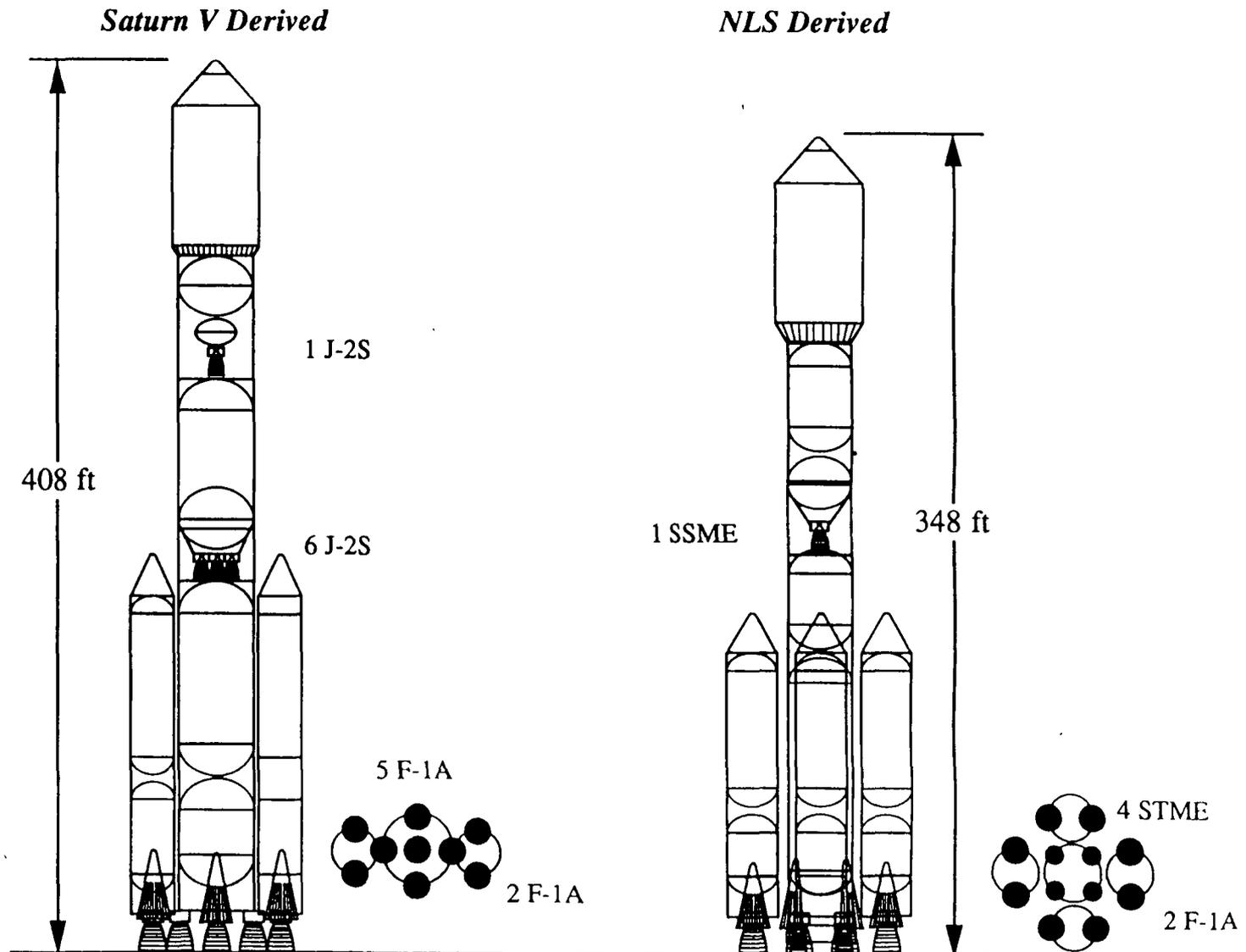
Lunar Mission Profile
Ascent Phase

100 nmi
Circ

MECO
(632 sec / 100 nmi)

Jettison Core/2nd Stage Start

(Time from Liftoff / Altitude)

Jettison Shroud/LES

Jettison Booster Pairs

Shutdown 1 Engine per Booster (pairs)

Step Throttle Boosters

Booster Impact

Core Impact
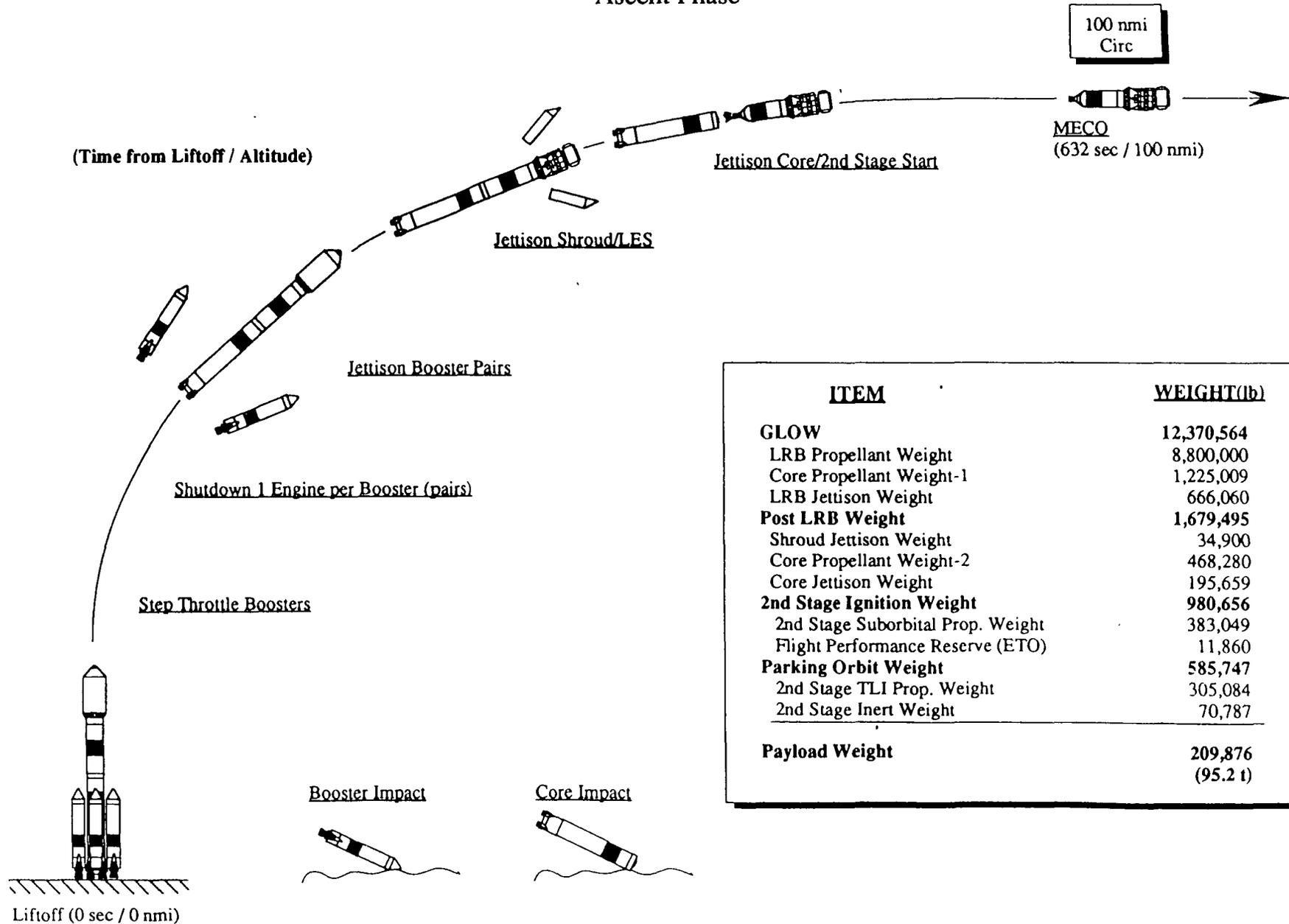
ITEM	WEIGHT(lb)
GLOW	12,370,564
LRB Propellant Weight	8,800,000
Core Propellant Weight-1	1,225,009
LRB Jettison Weight	666,060
Post LRB Weight	1,679,495
Shroud Jettison Weight	34,900
Core Propellant Weight-2	468,280
Core Jettison Weight	195,659
2nd Stage Ignition Weight	980,656
2nd Stage Suborbital Prop. Weight	383,049
Flight Performance Reserve (ETO)	11,860
Parking Orbit Weight	585,747
2nd Stage TLI Prop. Weight	305,084
2nd Stage Inert Weight	70,787
Payload Weight	209,876
	(95.2 t)

Liftoff (0 sec / 0 nmi)

Lunar Mission Profile
Orbital Phase

To Moon

Attitude Maneuver
(3:50)

Propellant Dump
(4:20)

Disposal Burn
(5:20)

Separation
(3:30)

Engine Cutoff
(3:05)

Up To 2 Orbits

MECO / Orbit Insertion
(0:00)

TLI Burn
(3:00)*
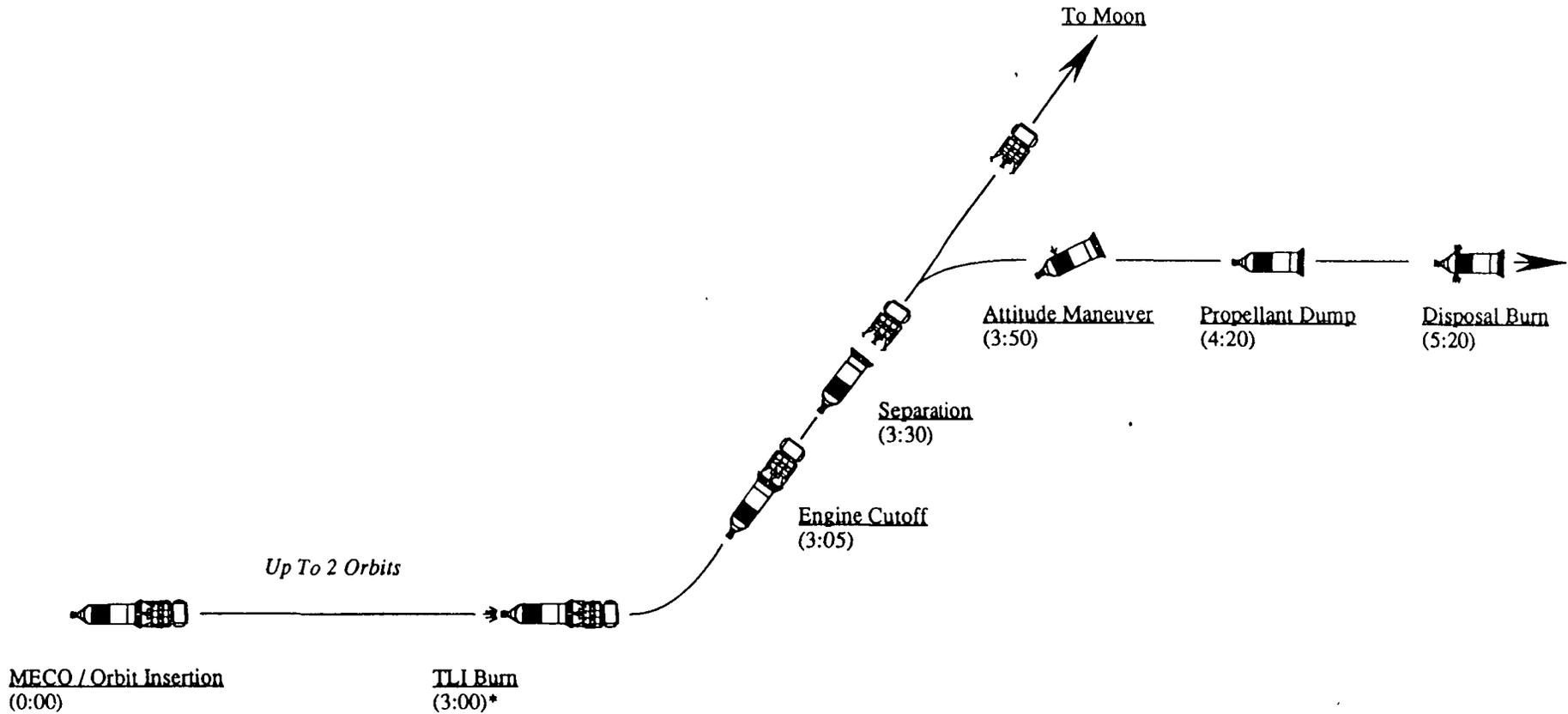
(Hr:Min from Orbit Insertion)
Nominal

Saturn V Derived HLLV w/ 2 LOX/RP Boosters
Single Launch - Piloted

Side View

56 ft
103 ft
160 ft
410 ft

22 ft D
33 ft D
83 ft

Base View
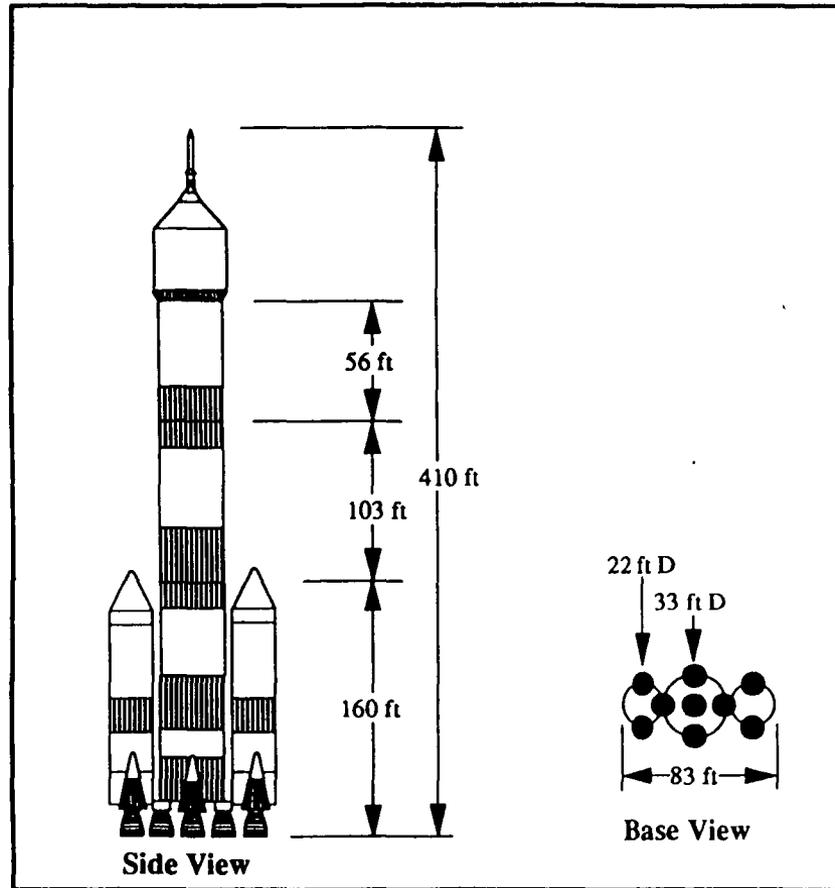
Payload:	215 klb (98 t) / 561 klb (254 t)
Final Position:	TLI/LEO Cutoff
GLOW:	13.3 Mlb
Engine Out:	None

BOOSTER:

Number/Type:	2/New
Inert Mass:	167 klb
Propellant Mass:	2.2 Mlb
Propellant Type:	LOX/RP
Engine Type/#:	F-1A/2
Vac /SL Thrust (Ea):	2.02/1.8 Mlb
Vac ISP:	303.1 s

CORE S-IC:

Inert Mass:	461 klb
Propellant Mass:	6.0 Mlb
Propellant Type:	LOX/RP
Engine Type/#:	F-1A/5
Vac /SL Thrust (Ea):	2.02/1.8 Mlb
Vac ISP:	303.1 s

CORE S-II:

Inert Mass:	134 klb
Propellant Mass:	1.4 Mlb
Propellant Type:	LOX/LH_2
Engine Type/#:	J-2S/6
Vac /SL Thrust (Ea):	265 klb
Vac ISP:	436 s

TLI Stage:

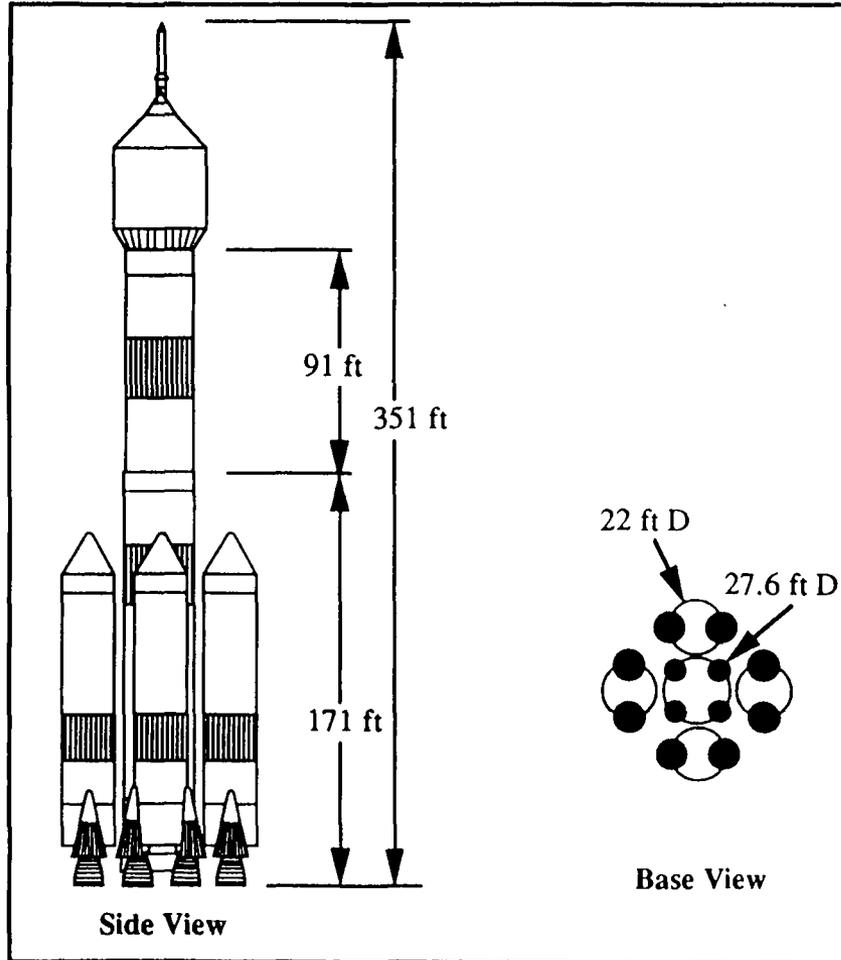|	
Inert Mass:	47 klb
Propellant Mass:	296 klb
Propellant Type:	LOX/LH_2
Engine Type/#:	J-2S/1
Vac Thrust (Ea):	265 klb
Vac ISP:	

Shroud - Size: 33 x 47 ft
Mass: 23,860 lb

Notes:
- F-1A's are 75% Step Throttlable
- Use Throttle For Loads
- Max G = 4.0 / Max q = 900 psf

NLS Derived HLLV w/ 4 LOX/RP Boosters
Single Launch - Piloted

Payload:	210 klb (95 t) / 586 klb (267 t)
Final Position:	TLI/LEO Cutoff
GLOW:	12.4 Mlb
Engine Out:	None

BOOSTER:

Number/Type:	4/New
Inert Mass:	166.5 klb
Propellant Mass:	2.2 Mlb
Propellant Type:	LOX/RP
Engine Type/#:	F-1A/2
Vac /SL Thrust (Ea):	2.02/1.8 Mlb
Vac ISP:	303.1 s

CORE:

Inert Mass:	195.7 klb
Propellant Mass:	1.69 Mlb
Propellant Type:	LOX/LH2
Engine Type/#:	STME/4
Vac /SL Thrust (Ea):	650/551 klb
Vac /SL ISP:	428.5/365 s

TLI Stage:

Inert Mass:	70.8 klb
Propellant Mass:	700 klb
Propellant Type:	LOX/LH$_2$
Engine Type/#:	SSME/1
Vac Thrust (Ea):	489.9 klb
Vac ISP:	452.4 s

Side View

91 ft

351 ft

171 ft

22 ft D

27.6 ft D

Base View

Shroud - Size:	38 x 47 ft
Mass:	23,860 lb

Notes: • F-1A & STME are 75% Step Throttlable

First Lunar Outpost Mission Performance
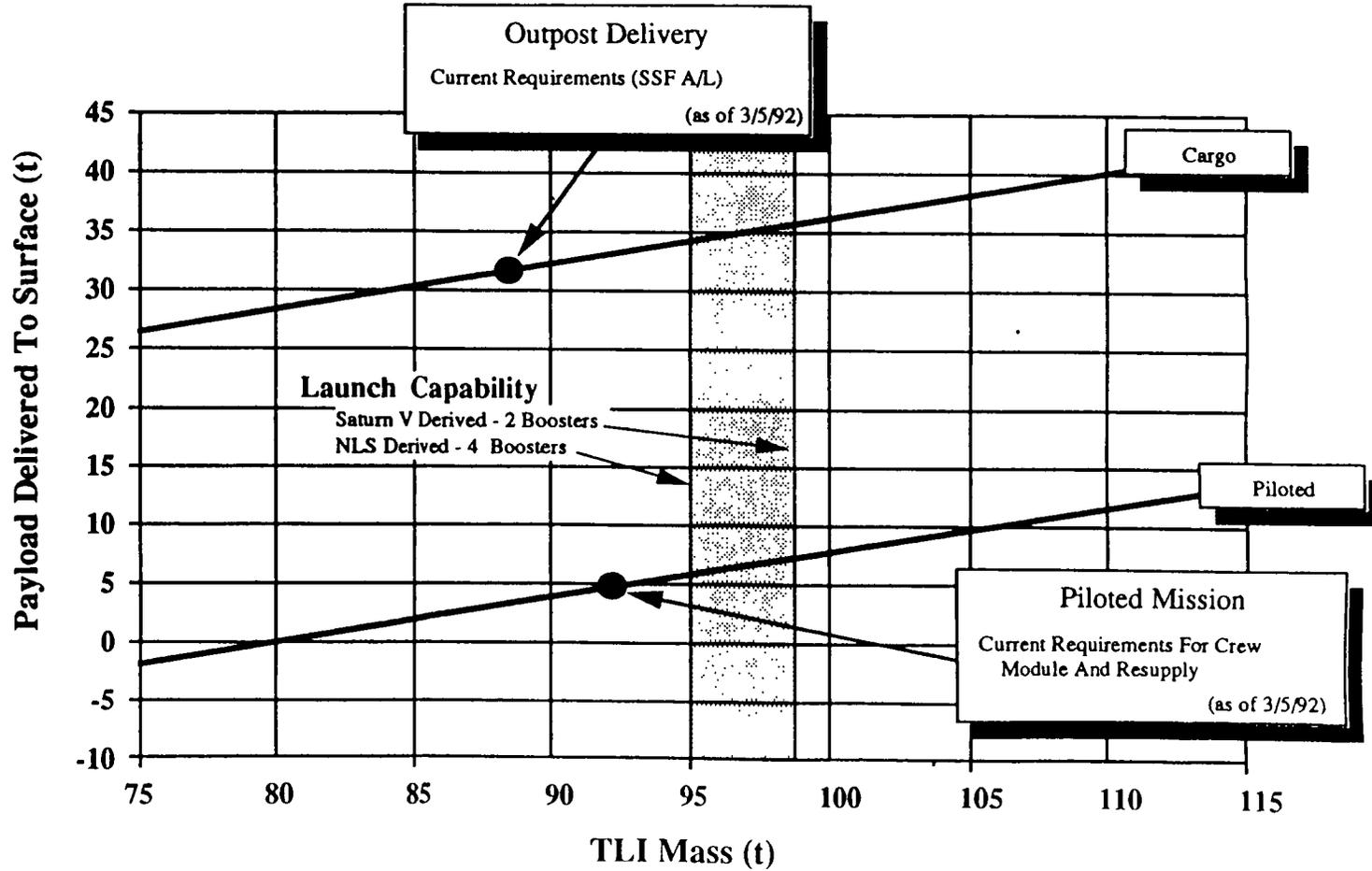
Future Trade Studies

- System Design, Performance, Operations And Cost Comparison Of Alternate Concepts
  - Clean Sheet Concepts
  - Minimize Hardware Elements
  - No Strap On Boosters

- Finalize Common Shroud Study
  - Piloted Mission
  - Cargo Mission

- Use Of Electro Mechanical Actuators For TVC On Large Engine Systems
  - Power Budget, Duty Cycle, Power Supply, Etc
  - System Weight And Cost Comparisons With Hydraulics

- Alternative Propulsion Concepts
  - Expendable SSME On NLS Core
  - New Or AlternateLOX/RP Engines

- FLO activity is an on-going requirements development process that will progress through numerous iterations before final selection of technical approach

- Current FLO concepts provide a framework for developing and testing requirements, the concepts are a "first cut" that will be refined considerably as analysis proceeds

- In summary:

  - These approaches are not final and others have not been ruled out

  - Additional concepts, approaches, and issues will be identified and assessed

  - Input from the SEI community has been and continues to be valuable

  - Interim status reviews will continue as FLO products mature

Organizational Change Incentives and Resistance

Dr. Peter C. Bishop, Chair
Studies of the Future
University of Houston-Clear Lake
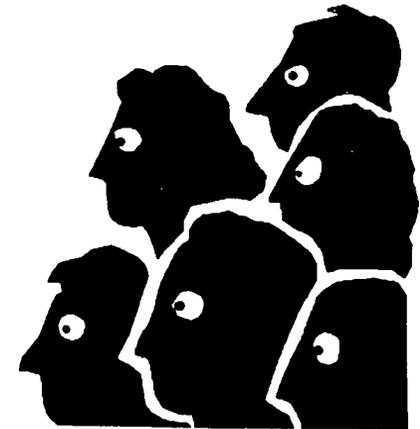
Models of Change
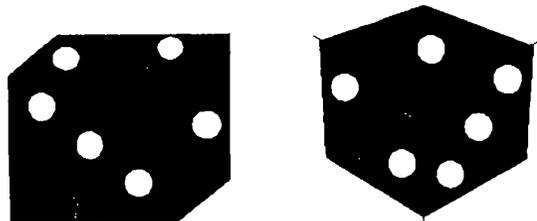
Momentum

Choice

Discontinuity

What we know about change

√ Rates of change vary over time

√ Outcomes are unpredictable

√ We rarely get the change we want

Elements of the Current Period

Origins

Trends

Choices

Outcomes
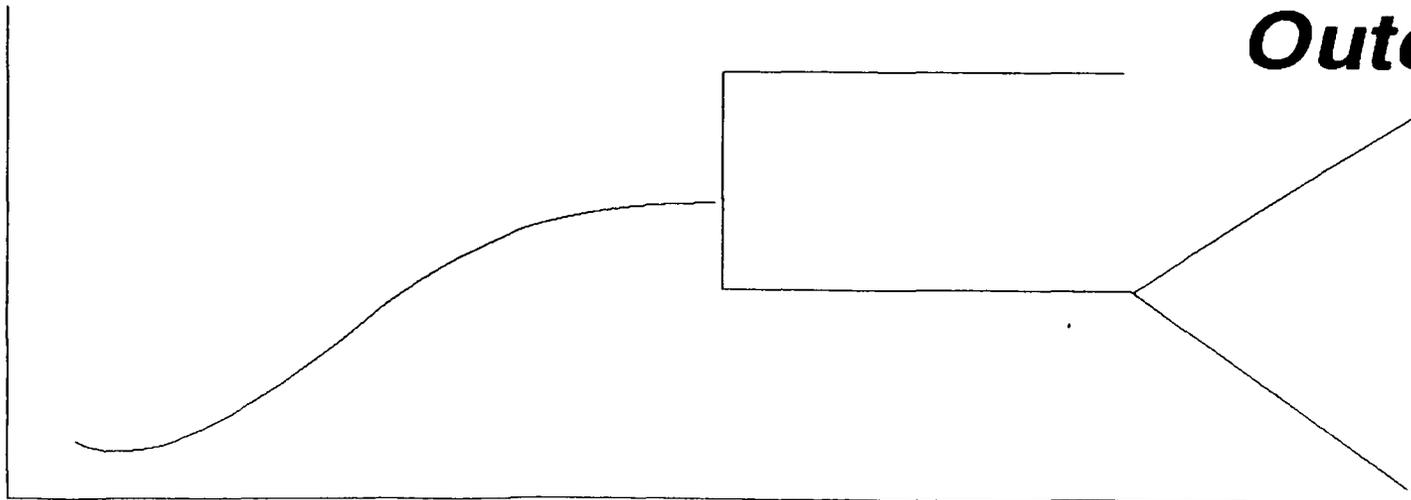
The Signs of Change

Anomolies

Strain

Contradictions

Leaders break the cycle of the resistance to change

The Leadership Role

Change $\xrightarrow{+}$ Complexity

Complexity $\xrightarrow{+}$ Visionary Leadership

Visionary Leadership $\xrightarrow{-}$ Ambiguity

Ambiguity $\xrightarrow{+}$ Threat

Threat $\xrightarrow{+}$ Change

Problem of Rapid Change

Change $\xrightarrow{+}$ Complexity

Complexity $\xrightarrow{+}$ Ambiguity

Ambiguity $\xrightarrow{+}$ Threat
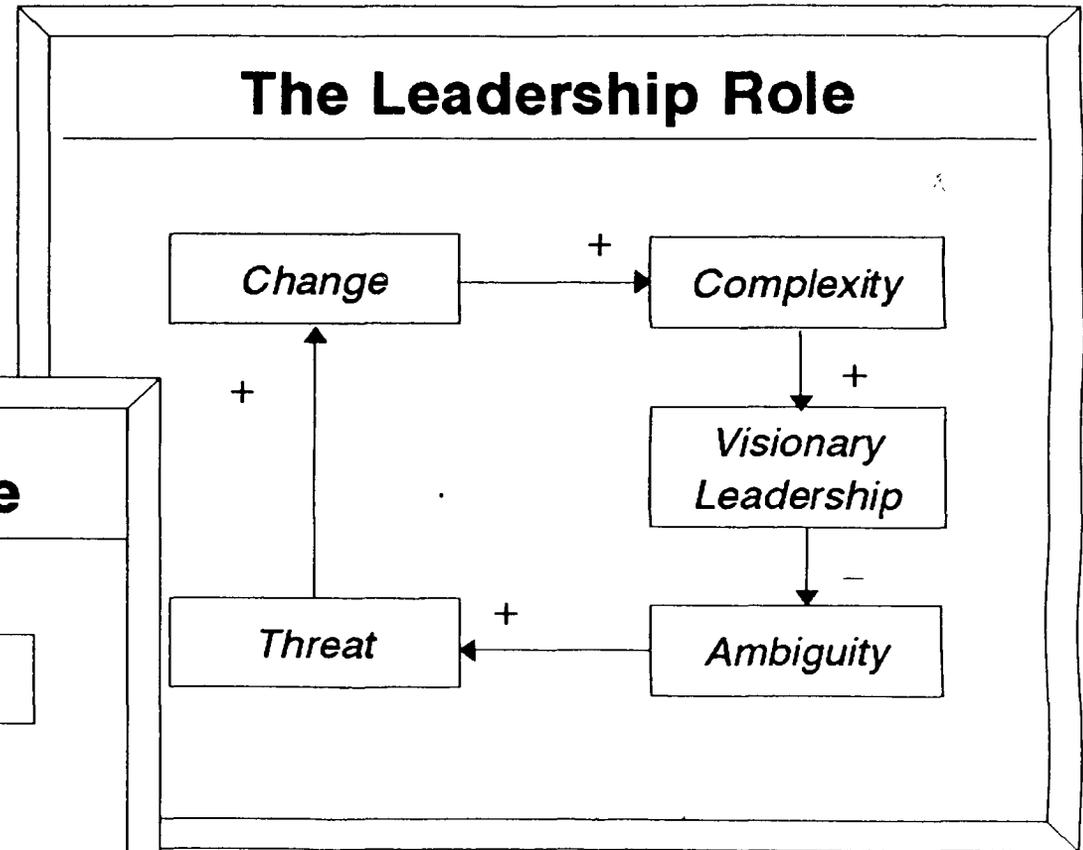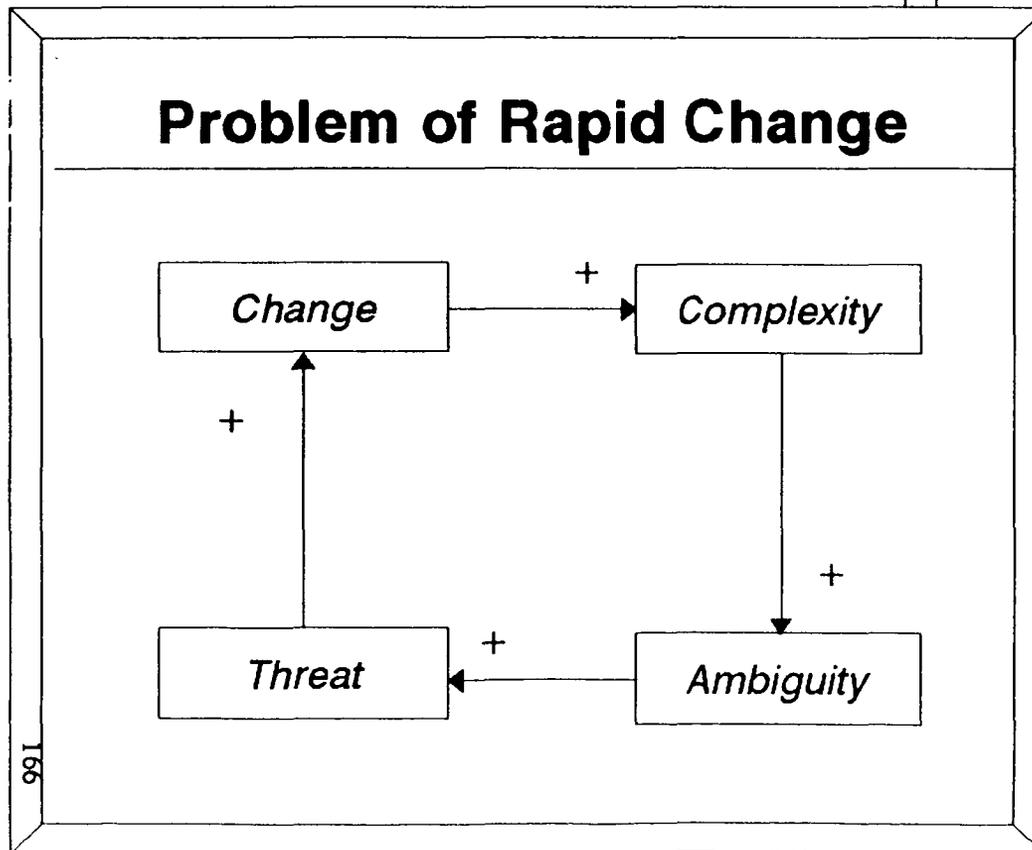
Threat $\xrightarrow{+}$ Change

by reducing ambiguity and decreasing threat.

Leader's Contribution

Values	What is important to us?
Vision	What can we become?
Direction	Which way should we go?
Boundaries	What are the rules of the road?
Adjustments	When and how do we change?

Paradigms -- Our Worldview

Rules that filter

- What we see

- What we say

- What we do

- Powerful and necessary in stable times
- Dangerously outmoded in times of change

Paradigm Change
Examples

Tradition ———————————————→ Flexibility
"We've always done it this way."

Security ———————————————→ Risk
"I didn't want to do anything wrong."

Authority ———————————————→ Motivation
"Because I said so."

Accountability ———————————→ Responsibility
"You have to meet your quota."

Competition ———————————————→ Cooperation

The Effects of Revealing Paradigms

► Making assumptions and skills explicit

► Considering their utility

► Changing them to suit the situation

Checklist for Change

1. Understand the current period
   - when did it start?
   - what is the paradigm?
   - why was that paradigm adopted?

2. Understand the sources of change
   - what has changed?
   - how soon before the system goes critical?

3. Understand the stakeholders
   - who is invested in the current paradigm?
   - who is ready for change?
   - who is confused?

4. Identify leaders and champions

5. Propose an alternative paradigm
   - new vision
   - new values
   - new rules
   - new rationale

Getting What You Want

Getting What You Want

Technology to overcome the limits of organizational control

David H. Peterson
Ventana Systems, Inc.
Harvard, Massachusetts

Technology to Overcome the Limits of Organizational Control

Technology to Overcome the Limits of Organizational Control
Outline

- Human control limits
- Organizational control limits
- Technology to overcome the limits

Technology

Technology
Definition of Technology:

- Ways to Increase Human Effectiveness
  - Machines
  - Methods
  - Models

Control

Control
Four steps to achieve control:

- Understand
- Predict
- Choose
- Act

13

Control

Four steps to achieve control:

- Understand
- Predict
- Choose
- Act

Control

Technology to Overcome the Limits of Organizational Control

Outline

- **Human control limits**
- Organizational control limits
- Technology to overcome the limits

Technology to Overcome the Limits of Organizational Control

Human control limits in unfamiliar environments

Target is clear, but:

- Required control misperceived
- Side effects unanticipated
- Control efforts missed

Human control limits in unfamiliar environments

Technology to Overcome the Limits of Organizational Control

Outline

- Human control limits
- **Organizational control limits**
- Technology to overcome the limits

Technology to Overcome the Limits of Organizational Control
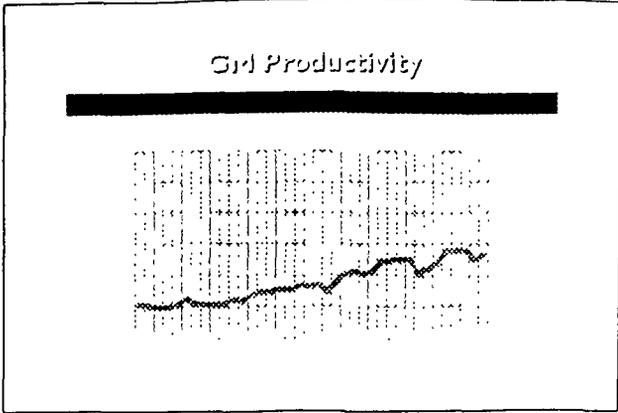
GM Productivity
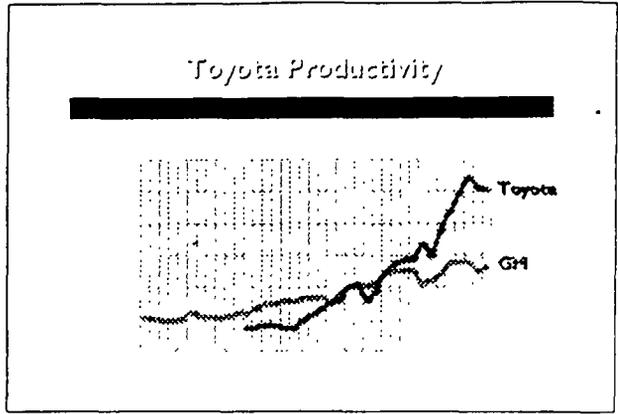
Toyota Productivity
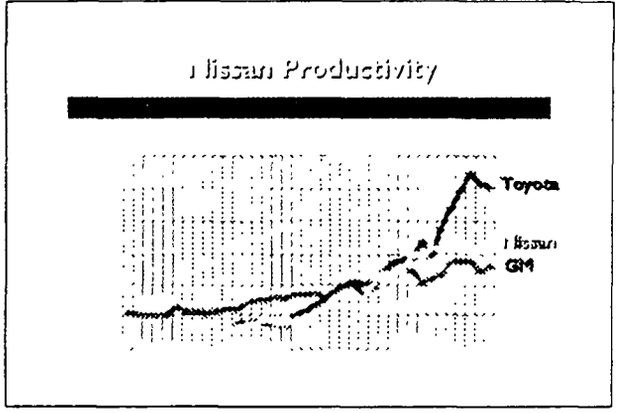
Nissan Productivity

Organizational control limits

Technology to Overcome the Limits of Organizational Control
Outline

- Human control limits
- Organizational control limits
- **Technology to overcome the limits**

Ford Productivity

Toyota
Ford
Nissan
GM

Technology for Automotive Organizational Control

- High standards
- Employee Involvement / Teamwork
- Statistical process control

Big-Project / Low-Volume Organizational Control

Target
Productivity
Automation

Technology to Overcome the Limits of Organizational Cont
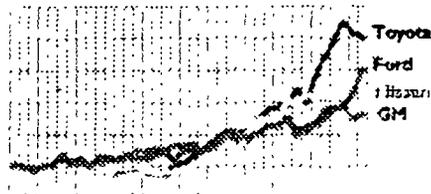
Ford Productivity

Technology for Automotive Organizational Control

Big-Project / Low-Volume Organizational Control

Big-Project / Low-Volume
Organizational Control
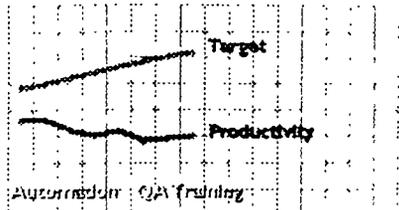
Big-Project / Low-Volume
Organizational Control

Big-Project / Low-Volume
Organizational Control
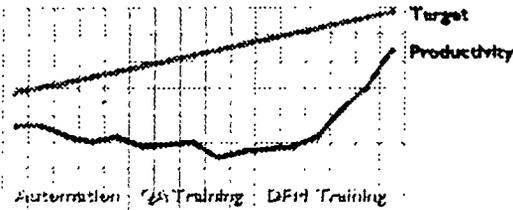
Big-Project / Low-Volume
Organizational Control

Big-Project / Low-Volume
Organizations

Additional limits:

. Small sample sizes

. Long lead times

. Expensive experiments

. Expensive data

Big-Project / Low-Volume
Organizations

Big-Project / Low-Volume
Organizations

Imperatives for control:

. Squeeze all information out of data

. Optimize for high-leverage decisions

Big-Project / Low-Volume
Organizations

Technology for Big-Project / Low-Volume Organizations

- High Standards
- Employee Involvement / Teamwork
- Model-based Statistics
- Model-based Optimization

Lessons Learned
Models must be:

- Dynamic (to predict change)
- Complete (people, project, hardware)
- Robust (realistic under all inputs)

Lessons Learned
Statistics must be:

- Model-based
- Optimal
- Compute-Intensive

Future Management

- Technology we now apply to hardware will be routinely applied to organizations

---

Technology for Big-Project / Low-Volume Organizations

Lessons Learned

Lessons Learned

Future Management

ACQUISITION STREAMLINING

-A CULTURAL CHANGE-

Jesse Stewart

DEFENSE SYSTEMS MANAGEMENT COLLEGE

May 6,1992

TOPICS

- **DEFENSE SYSTEMS MANAGEMENT COLLEGE**

- **STREAMLINING INITIATIVES**

- **CULTURAL CHANGE**

DEFENSE SYSTEMS MANAGEMENT COLLEGE

- **EDUCATIONAL INSTITUTION**
  - GRADUATE COLLEGE(PMC)
  - PROFESSIONAL DEVELOPMENT
  --SHORT COURSES

- **6,000 STUDENTS/YEAR**
  -GOVERNMENT AND INDUSTRY

- **175 FACULTY**

DEFENSE SYSTEMS MANAGEMENT COLLEGE

- **RESEARCHERS**

- **CONSULTANTS**

- **INFORMATION SOURCES**

- **ACQUISITION MANAGEMENT POLICY EXPERTS**

EDUCATIONAL PHILOSOPHY

- **COURSES FROM PROGRAM MANAGER'S PERSPECTIVE**

- **THINK FIRST, REGULATIONS SECOND**

- **REALISM AND CURRENCY**

ACQUISITION WORLD
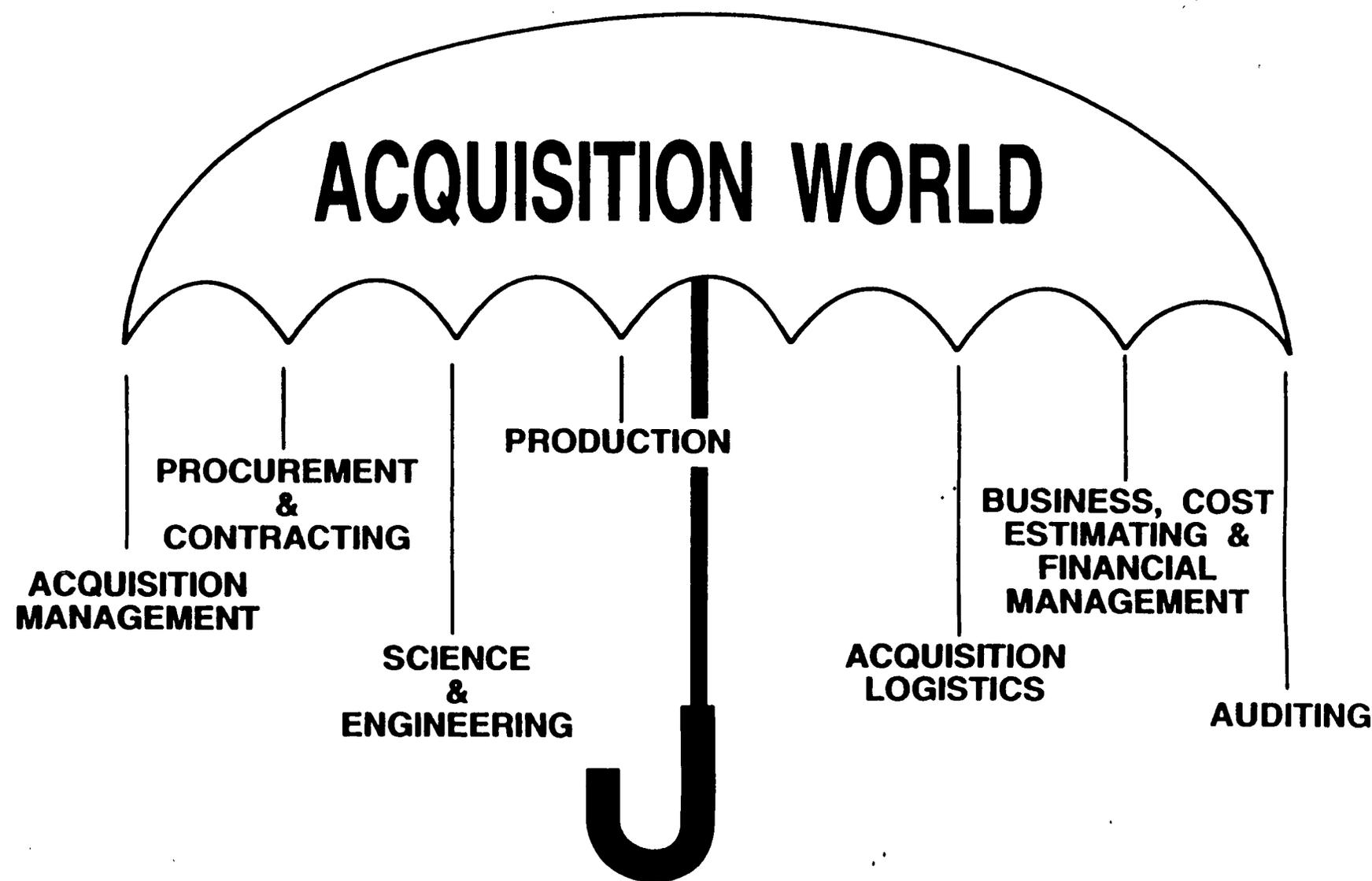
PROCUREMENT
&
CONTRACTING

PRODUCTION

ACQUISITION
MANAGEMENT

SCIENCE
&
ENGINEERING

BUSINESS, COST
ESTIMATING &
FINANCIAL
MANAGEMENT

ACQUISITION
LOGISTICS

AUDITING

DEFENSE ACQUISITION ENVIRONMENT

069.58.8106.14743
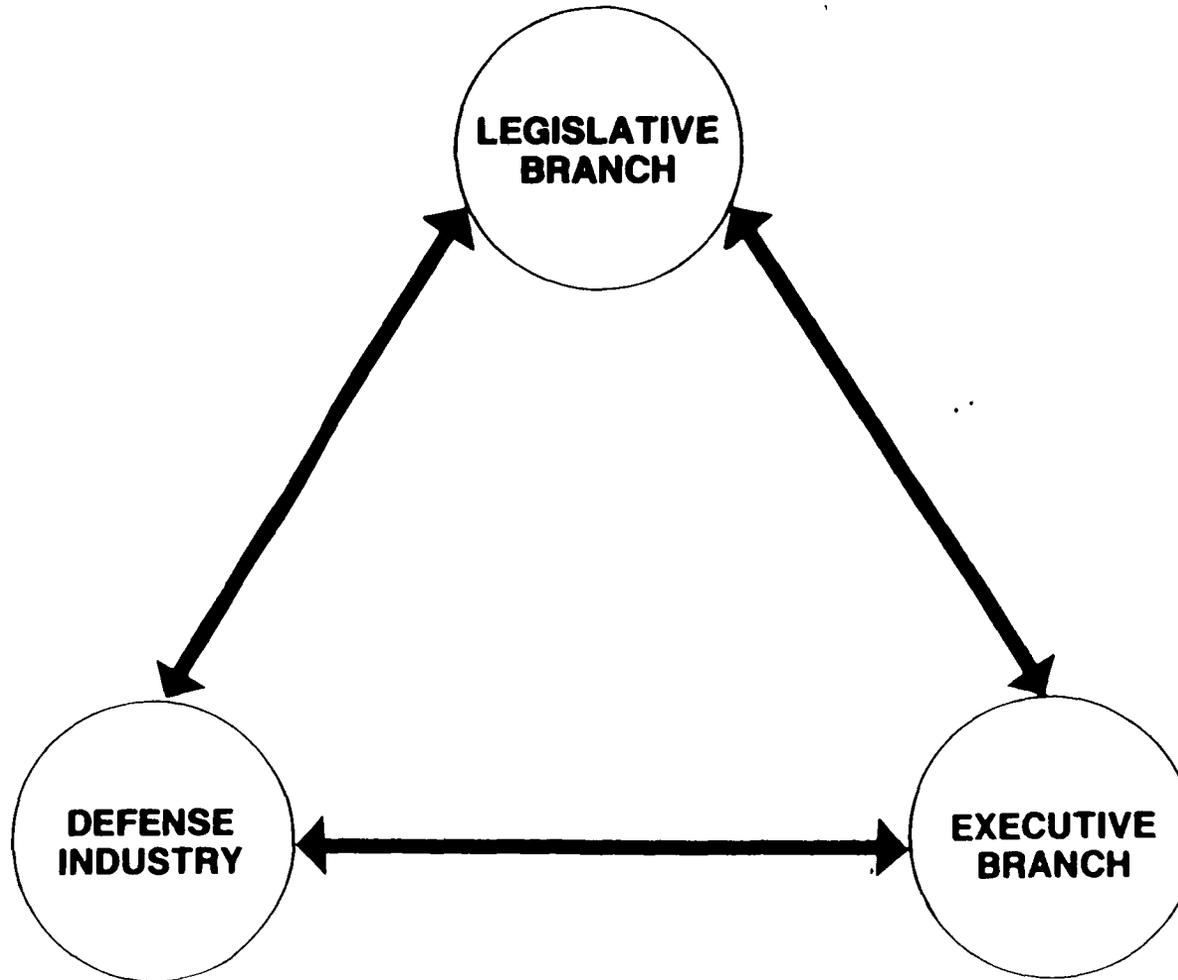
DEFENSE ACQUISITION ENVIRONMENT

- **INCREASING EMPHASIS ON ACQUISITION EDUCATION**

- **DECLINING FORCE STRUCTURE AND RESOURCES**

- **LEGISLATIVE DEMANDS AND REQUIREMENTS**

STREAMLINING INITIATIVES

- ■ **ORGANIZATIONAL OVERSIGHT RELIEF**

- ■ **LEGISLATIVE STATUE CHANGE**

- ■ **DEFENSE ACQUISITION PILOT PROGRAMS**

ORGANIZATIONAL STREAMLINING TYPES

- **WITHIN PROGRAM OR COMMAND**

- **SPECIFICATION and REGULATION ORIENTED**

- **PERSONNEL REASSIGNMENT**

DEFENSE LAW REVIEW

- **1991 DEFENSE AUTHORIZATION ACT**

- **STREAMLINE AND CODIFY ACQUISITION LAW**
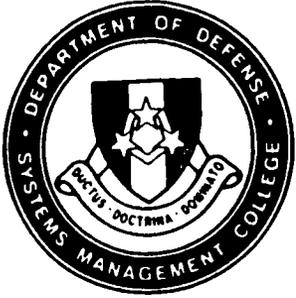
LAW REVIEW PURPOSE

- **REVIEW ACQUISITION LAWS**
  - **STREAMLINE ACQUISITION PROCESS**
  - **ELIMINATE UNNECESSARY LAWS**
  - **ENSURE FINANCIAL and**
    ETHICAL INTEGRITY
  - **PROTECT BEST INTERESTS OF DoD**

- **PREPARE RECOMMENDED CODE**

LAW REVIEW OBJECTIVES

- **FACILITATE GOVERNMENT ACCESS**
  -COMMERCIAL TECHNOLOGIES
  -SKILLS IN COMMERCIAL MARKETPLACE

- **COMMERCIAL ACCESS TO GOVERNMENT DEVELOPED TECHNOLOGIES**

LAW REVIEW OBJECTIVES

- **COMMERCIAL PRODUCTS/SERVICES PURCHASE BASED ON MARKET PRICE**

- **INTEGRATE COMMERCIAL/DEFENSE PRODUCTION WITHOUT ALTERING ACCOUNTING OR MANAGEMENT PROCEDURES**

- **MINIMIZE ADDITIONAL REPORTING**

MODEL

PANEL
GUIDANCE

Legislative
Intent

Impacts in
Practice

RESEARCH

Problem
Identification

PROPOSED
CHANGE

PREPARE
SUPPORTING
DOCUMENTS
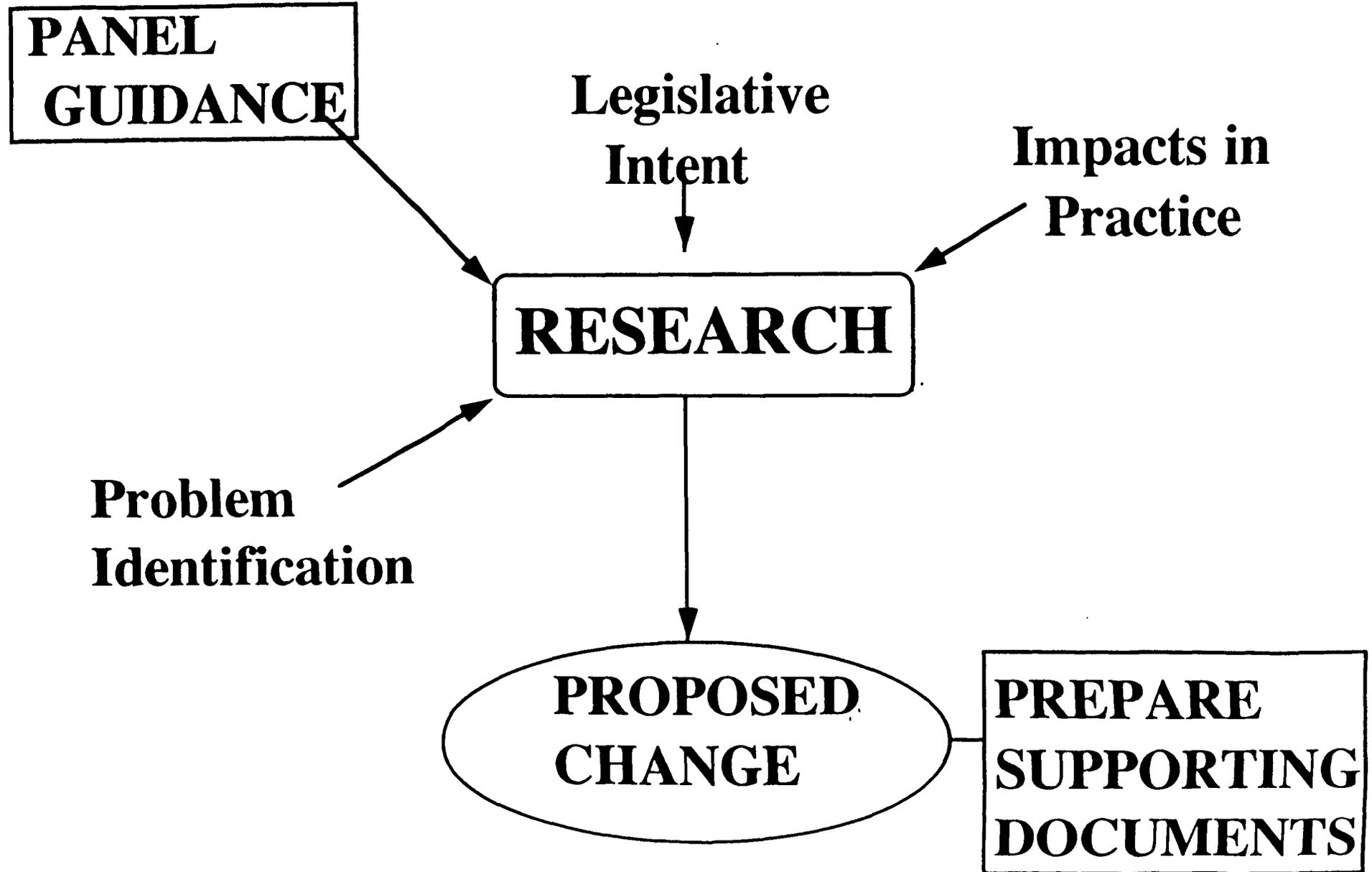
Public Law
Pilot Program

- **1991 PUBLIC LAW 101-510**

- **MAJOR DEFENSE ACQUISITION PILOT PROGRAM**

- **DoD**
  - **DESIGNATE UP TO SIX PROGRAMS**
  - **PROVIDE LEGISLATIVE PROPOSAL**

GUIDELINES

- **MAY WAIVE OR LIMIT ANY PROVISION OF LAW EXCEPT:**
  - **FINANCIAL INTEGRITY OF DESIGNATED PROGRAM**
  - **AUTHORITY OF DoD INSPECTOR GENERAL**

- **ACQUISITION RELATED LAWS ONLY**

PILOT PROGRAM GOALS

- **MANAGEMENT-BY-EXCEPTION**

- **FINANCIAL FLEXIBILITY**

- **MANAGEMENT DISCRETION**

PROGRAM ANALYSIS

- **EFFECTS OF LAW**

- **EFFECTIVENESS OF NEW PROCESS**

- **EFFICIENCIES OR SAVINGS**

PROCEDURES

- **PROGRAMS ANALYZE LAWS AND REGULATIONS -PROVIDE RECOMMENDATIONS**

- **DEPARTMENT OF DEFENSE-- REVIEW and FORWARD TO CONGRESS**

- **CONGRESS AUTHORIZES and DESIGNATES IN APPROPRIATIONS**

POTENTIAL EXAMPLES

- **MODIFY**
  - **STATUTORY REPORTING**
  - **OBLIGATION TIME FRAMES**
  - **REPROGRAMMING AUTHORITY**
  - **TEST REVIEW REQUIREMENTS**
  - **MULTIYEAR CONTRACTING**

- **WAIVE PROGRAM CATEGORY DESIGNATION**

STREAMLINING STATUS

- **SPECIFICATIONS--ONGOING**

- **LAW ADVISORY PANEL -REPORTS JANUARY, 1993**

- **PILOT PROGRAM -PREPARING CONGRESSIONAL PROPOSAL**

-CHANGE-
A CULTURAL PERSPECTIVE

- **ORGANIZATIONS HAVE A CULTURE**
  - GENERALLY RESIST CHANGE

- **JOBS AND PERSPECTIVE BUILT ON**
  LESSONS LEARNED

- **PUBLIC SECTOR DIFFERENT**
  THAN PRIVATE
  -REVISIT MANY DECISIONS

-CHANGE-
A CULTURAL PERSPECTIVE

- **LEADERSHIP**
  - **TOP DOWN VERSUS BOTTOMS UP**

- **MAJOR CHANGE ISSUES**
  - **COMPARATIVE DATA**
  - **RESOURCES**
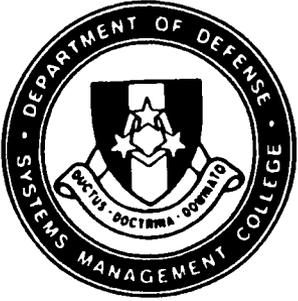
-CHANGE-
ACADEMIC VIEW

- **STAFFING DIFFICULT WITHOUT TOP LEADERSHIP SUPPORT**
  - **VISION TO BREAK PARADIGM**
  - **PROGRAM OFFICE CLOUT**

- **ANALYSIS OF CHANGES**
  - **CLEAR RATIONALE**
  - **RESOURCES AVAILABLE**

SUMMARY

- **THREE TYPES OF STREAMLINING**
  - **SPECIFICATION**
  - **LAW REVIEW**
  - **PILOT PROGRAM**

- **MAJOR OPPORTUNITY FOR ACQUISITION IMPROVEMENT**

ORGANIZATIONAL CHALLENGE

Effect of Development Organization Type on Program Development and Production Cost*

• From AMCM Database
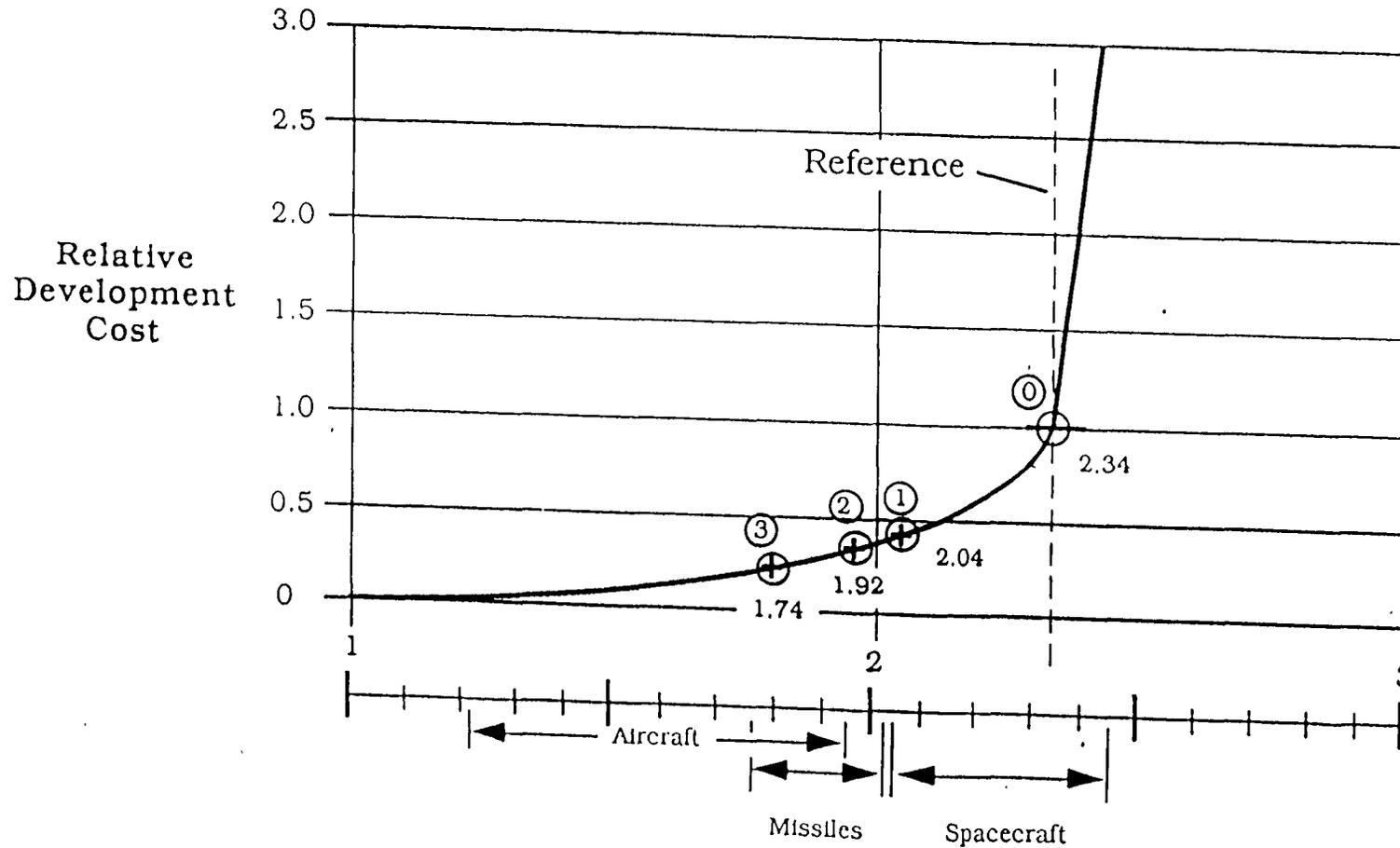
Small changes in development culture can have

- The ingredients of successful low-cost, high technology programs are well known and universally recommended by successful program managers interviewed
  - Use government only to define and verify requirements
  - Keep requirements fixed: once requirements are stated, only relax them; never add new ones
  - Place product responsibility in a competitive private sector
  - Specify end results (performance) of products, not how to achieve the results
  - Minimize government involvement (small program offices)
  - Insure that all technologies are proven prior to the end of competition
  - Utilize the private sector reporting system: reduce or eliminate specific government reports
  - Don't start a program until cost estimates and budget availability match
  - Minimize or eliminate government imposed changes
  - Reduce development time: any program development can be accomplished in 3 to 4 years once uncertainties are resolved
  - Force people off of development programs when development is complete
  - Incentivize the contractor to keep costs low (as opposed to CPAF, CPFF of NASA)
  - Use geographic proximity of contractor organizations when possible
  - To reduce the number of interfaces and keep responsibilities clean, use the major prime contractor as the integrating contractor

206

R.L. GRANT REMARKS
CQI: CULTURE CHANGE WITHIN BOEING DEFENSE & SPACE GROUP

Space Exploration Initiative Third Technical Interchange Meeting
Houston, TX
May 6, 1992

- Boeing's approach to Continuous Quality Improvement (CQI) is an evolutionary one

- While consistently stressing the importance and inevitability of fundamentally changing the way we do business, and characterizing CQI as the "cornerstone" of our business management approach, the Corporation left it to individual organizations to adopt and implement CQI as they saw fit

- The ultimate aim of any CQI activity is the delivery of quality products and services. We are in the business of satisfying our customer.

- Fundamental to the culture change around the delivery of a quality product are several key points:

  - Quality is ultimately defined by the customer

  - Redefining "customer" as internal as well as external.

  - Understanding the part suppliers play in the quality of our products, internally as well as externally.

- The shape of CQI in any organization around Boeing has been based on the workplace culture, products, customers and leadership personalities of that organization

- Our early approach focused on Quality Circles and Process Improvement Teams where we began to apply principles of problem-solving, process analysis, SPC and other tools

- However, in 1989, Boeing Aerospace began an organization-wide focus on planning - getting everyone's arrows pointed in the same direction.

- A Large-Scale System Change process brought management together for 2-3 days for strategic planning and goal-setting sessions, a process that under normal circumstances would take months to accomplish.

- The single threading of the Company vision, mission, goals, objectives and actions through every level, organization and individual is known as Policy Deployment. Performance Management, the vehicle to communicate and involve everyone in the process, will be accomplished within the Group by the end of this year

- Our CQI approach follows three principle pathways for achieving quality products the first time every time:

  - **Policy Deployment**, mentioned earlier, is the system by which goals are determined, plans to achieve the goals are established, and measures are created to assure progress toward the goals

  - **Partnership** - paying attention to the way we work together - with our customers, suppliers, both internal and external; the way we work together cross-functionally, between paycodes; and the way we use teams to get work done. Through partnering, we recognize the importance of making organizational boundaries permeable at all stages of a product or process, thereby empowering everyone to have an opportunity to influence what goes on and be satisfied with with the outcome and how it was attained.

  - **Process Improvement** - shifting the focus away from examining the quality of the end product to the quality of the process which creates the end product, thus preventing problems or poor quality from occuring and enabling true improvement and innovation

- Continuing with the spirit of evolution, the Defense & Space Group senior managers participated in a Japanese Study Mission late last year which involved a "Ground School"in Seattle followed by a visit to 10 world class Japanese companies

- Some basic observations of the Japanese management culture include:

  - Setting up an effective management system to assure the long-term viability and continuous adaptation/evolution of the enterprise is management's job (Nippon Steel: "Each day a new dawn for steel.")

  - Well-structured, documented, disciplined process for conducting all phases of business activity. System driven by facts and data and founded on the Plan-Do-Check-Act continuous learning cycle.

  - Entire organization maintains a consistent and persistent focus on full customer satisfaction (Quality, Cost, Delivery, Safety and Morale). **Customer In** as a way of life in contrast to old paradigm of Product Out

  - Design engineering must stay close to the field where the products are used and close to the shop floor where the products are made. ("Armchair engineers" who never visit the factory floor must go)

  - Manufacturing is a strategic weapon that can be used to continuously meet the changing needs of the customer and beat the competition. Manufacturing systems designed with a resiliency to assure a rapid and flexible response to changing market needs

- Everyone is involved in improving everything all the time. People seen as an asset; information on goals and performance is shared openly; team environment founded on mutual respect

- They are globally ambitious and playing for keeps

- This lead us to go beyond seeing CQI as a program. While we did not radically change our current course, we are applying these lessons and are committed to institutionalizing CQI, making it the way of life in D&SG

- We will implement CQI as The Management System for D&SG and are working to develop a common approach

- Using the Malcolm Baldrige criteria to measure our progress, we will undertake the following initiatives:

  1. Train all managers to lead and teach CQI

  2. Establish a CQI information-gathering and analysis process

  3. Improve the planning process to produce an integrated plan that encompasses product, technical, business and CQI efforts

  4. Continue toward a participative work place for all employees and prepare the work force to function effectively in a CQI environment

  5. Identify, document and improve our product and business processes

  6. Measure our performance levels and improvement trends, and identify need for change

  7. Improve satisfaction of our external customers.

- So, we learn, we implement and we evolve. Japanese as well as US companies have their ups and downs as they embrace CQI as The Management System. Often we must regroup and restart.

- But the point is they stay with it, they persevere, and so shall we.

- We don't know what the future holds in this time of tremendous change.

- And while we may currently be in the middle of the journey as we perform the transition to a CQI culture, I believe we will figure it out, we will land on our feet, and we will compete successfully as a world class organization with all comers!

Policy Deployment

Policy deployment aligns the work of our people to meet consensus-developed mission statements, goals and objectives. This is reinforced by performance management contracts with all managers.

Process Management

Processes are managed effectively and efficiently by creating process-focused teams to accomplish continuous improvements that reduce defect rates and process flow time.

Product Variability Reduction

Product variability is minimized by product development teams that create robust designs. These teams iterate the design of a product while simultaneously developing manufacturing, support and training processes.

Supplier Excellence

We develop cooperative relationships with suppliers to help them improve their processes and products. Tracking costs of working with individual suppliers will enable us to select future suppliers on the basis of highest quality and lowest total cost.

Customer Focus

Maintaining a strong customer focus is the cornerstone of our continuous quality improvement process. This involves fulfilling the requirements and expectations of both internal and external customers.

People Involvement

Empowering our people creates ownership of and individual commitment to the continuous improvement process. This provides the energy to achieve our Quality goals.

Metrics

Metrics provide the data needed for measuring trends and making fact-based decisions. CQI metrics primarily focus on defect rates and flow times.

Customer Focus

Policy Deployment

Process Management

Product Variability Reduction

Supplier Excellence

METRICS

NASA
PROGRAMS

≡🖋Lockheed
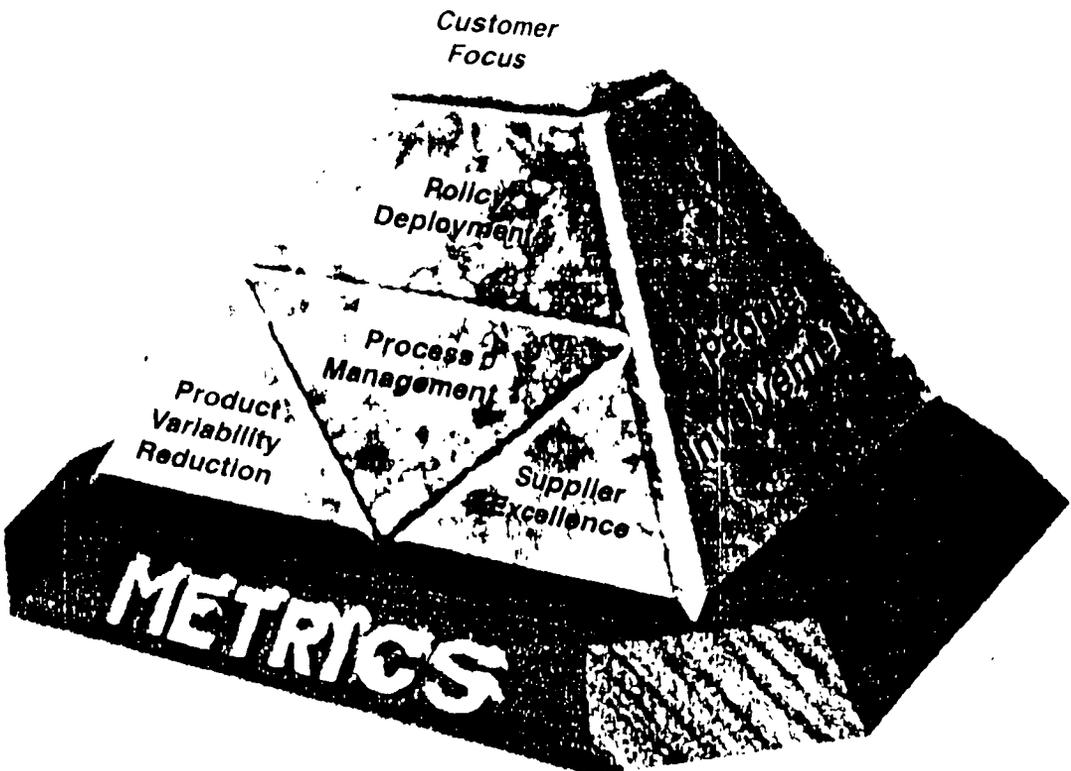
Technical Interchange Meeting
for
NASA
Space Exploration Initiative

Cooperation in Space

by

A. Guastaferro

May 6, 1992

Houston, Texas

212

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- **Personal background**

- **Objective**

- **Today's culture**
  - **Environment**
  - **Monopsony relationship**
  - **NASA decision maker**
  - **Industry participation**

- **Let's try a new approach**
  - **14 rules of Skunk Works**
  - **Analysis of rules**

- **Congressional oversight**

- **Programmatic impact**

- **Recommendations**

Personal Background

| 38 years of aerospace experience |

- 8 Air Force
- 22 NASA
- 8 industry

| Significant project management experience |

- QB-47 (aeronautics)
- Scout (launch vehicle)
- Viking (planetary)
- Rotor systems research aircraft (helicopter)
- Large space structures (technology)
- Space Station Freedom (large program)

| Other Government experience |

- Project management
- General management
- Tech rep of contracting officer
- SEB chair
- SEB member
- Launch conductor

| Served on a number of NASA, National Academy, and AIAA committees |

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To share a perspective of a cost-effective cooperative management structure of NASA and industry as we move towards the 21st century and the national commitment to continue our Exploration in Space with Humans

---

Environment

- **Space exploration has been a very focused R&D activity**

- **Uncertainty leads to cost reimbursible contracts and change as the standard**

- **Developments take a major part of a decade**

- **Estimated costs are traditionally underscoped**

- **Award fee became fashionable over fixed fee (contractors become more responsive to change)**

216

M8113F2002 1

Monopsony relationship

- **Many suppliers → one customer**

- **Demand driven**

- **High contractor investment**

- **Aggressive pricing**

- **Optimistic proposal**

- **Raised technological and programmatic expectations**

- **Atmosphere of "more for less"**

- **Contractors accept lower fees**

- **Leads to overcommitment of programs (advocacy is optimistic)**

- **Congressional funding shifts programs to the right**

Above leads to public and congressional criticism of both NASA and industry

NASA decision maker

- **Too many Phase A's**

- **Phase A's demand contractor investment**
  - **$$$**
  - **People**

- **Phase B's are started with inadequate funding**
  - **Competition will yield adequate design**
  - **Extended activities level playing field**

- **Scientific community (users) brought in late to process**

- **Phase C/D decision making is at top**
  - **Vertical assessment for approval drives NASA centers to become aggressive**

- **Non-advocacy review has been a remedy for aggressiveness**

- **For every 10 Phase A/B programs, only one will be approved**

- **Process wastes resources and creates false advocacy**

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Industry Participation

≡⟩Lockheed

- **Industry understands the technical and fiscal value of participation in civil space**

- **Military and national security programs have contributed significantly to industry's technical base and professional staff development**

- **Strong relationship with NASA centers during advocacy**

- **Industry becomes advocacy agents for NASA (selling the program through heavy investment and optimistic planning)**

- **Industry understands that a delayed program costs more investment dollars and has lower probability of a win**

Kelly Johnson (1950)

- Empowered program management in industry
- Small, effective project team
- Limited access to "reviewers"
- Simple documentation system
- Minimum reports (critical work will be documented)
- Cost reviews with performance measurement
- Empowered contractor (do not micromanage)
- Inspect at lowest level (do it right the first time)
- Contractor will test for flight (need for continued competency)
- Specifications established early (prior to contract)
- Timely funding
- Trust
- High security (keep outsiders outside)
- Pay for performance

220

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Let's Try a New Approach

Lockheed

Analysis

Industry management (rules 1, 2, and 3)

- TQM was started by Kelly Johnson
- Resources with program manager
- Complete control on people selection
- Shift from quantity to quality
- Stay "lean and mean"
- Must be customer driven

Documentation system (rules 4, 5, and 6)

- Our *paperless* programs must drive to be paperless
- MIS can be electronically controlled
- Most have more analysis and less paper
- There is "gold in the mounds of paper"
- Documentation reduction is our greatest challenge – but it is also our greatest reward

Let's Try a New Approach (Cont)   *Lockheed*

Analysis

Subcontract management (rule 7)

- Buy "off the shelf"
- Use of commercial practices
- Must take prudent risk by selection of the best with the least oversight
- Empower the subs to beat cost and schedule – reward if successful

Inspection (rule 8)

- Inspect at lowest level
- Avoid duplication
- Look for value added processes
- Reduce Government participation

Testing (rule 9)

- Try to do at contractor's facility
- Hold contractor responsible
- Eliminate Government testing

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Let's Try a New Approach (Cont)   ≡⅔*Lockheed*

Analysis

.Specifications (rule 10)

- **Firm-up early**

- **Make play not better**

- **Time is the enemy of good**

- **Be flexible to meet performance**

- **Use the best people up front**

Funding (rule 11)

- **Plan the funds**

- **Fund the plan**

- **Try to get congressional support**

Trust (rule 12)

- **Award fee process must create a trust atmosphere**

- **Government/industry working together towards a common goal**

- ***Must* improve in this area**

Let's Try a New Approach (Cont) ≡⫩Lockheed

Analysis

Security (rule 13)

- **A psuedo system may yield positive results**
- **No "bad verbs"**

Pay for performance (rule 14)

- **Pay the best to get the best**

- **NASA must recognize and reward and penalize**

- **Cost analyst should not decide that cost per hour will give you the *best***

- **More emphasis on skills**

- **NASA should not negotiate to lower the cost for fee determination**

- **Emphasis should be on getting the best at the lowest number of hours**

224

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Congressional Oversight

Lockheed

- **Responsibility for appropriation of resources should be rewarded with "shared accountability"**

- **NASA must treat their customer as a trusted and valued participant in the process**

- **Oversight by Congress should be reasonable and limited to major problems**

- **Micromanagement should be eliminated**

22

- **Must improve Government/industry relationship**

- **Need major overhaul of existing contracting methods**

- **Insufficient budget to match program**

- **Must change incentives to industry**
  – Higher reward opportunities
  – Lower fees for marginal performance

- **Government/industry must become equally accountable for successes and failures**

226

Recommendations

- NASA should assume a greater share of predevelopment funding

- NASA should reduce the number of underfunded programs and studies in Phase A/B to provide adequate funding for approved programs

- NASA should continue the nonadvocacy review process

- NASA should consider some elements of the Lockheed Skunk Works

- Congress should authorize multiyear funding for large programs

- Congressional staffers should provide better understanding of the risk associated with space exploration and get involved in the technical as well as political aspects of a program. Become an informed critic

- Industry, working with NASA, should find a way to leverage B/P and IRAD resources against underfunded predevelopment programs

- Industry must work harder to put the best and brightest on programs to achieve superior performance

- Industry must demonstrate that they are willing to provide their best people and facilities towards the civil space program for the benefit of:
  - The nation
  - The public
  - NASA
  - Congress
  - The user community
  - Their employees
  - The stockholders

SKUNK WORKS TYPE APPROACH FOR F-SAT

NASA EXPLORATION PROGRAMS OFFICE
JOHNSON SPACE CENTER
MAY 6, 1992

G. F. TURNER
MANAGER, F-SAT
LOCKHEED MISSILES & SPACE C(
(408)   756-5216

N92-33323

F-SAT PROGRAM

PROBLEM: MULTIPLE TOUGH COMPETITIONS COMING - LOW EARTH ORBIT ATLAS/DELTA CLASS MISSIONS

- DMSP, NOAA, SBR, SDI EXPERIMENTS, EOS, CLASSIFIED PGMS
- LOTS OF COMPETITION, COST INTENSIVE

NEED: DEMONSTRATED LOW COST "STANDARD" SPACECRAFT BUS

- CUSTOMERS CAN'T AFFORD TO REINVENT BUS FOR EVERY PGM.
- MUST ESTABLISH COST CREDIBILITY THROUGH PERFORMANCE.

APPROACH: DEVELOP AND PRODUCE A "STANDARD" SPACECRAFT BUS

- DEMONSTRATE LOW COST, VERSATILE DESIGN
- GOAL - FACTOR OF 4/5 COST REDUCTION
  - MAJOR IMPROVEMENT REQUIRES MAJOR CHANGES
    - IN OPERATING METHODS
    - IN PROGRAM CULTURE

THE CLASSIC PROGRAM ORGANIZATION

Procurement	Pgm Manager	Business Ops.

Quality Ass.	Manufacturing	Engineering	Systems Engineering	Test

Attitude Control System	Electronics/ Data Systems	Electrical Power System	Structure	Propulsion System

Processors ACS Electronics Safe Mode Electronics	Power supplies Processors Power Controls	Solar Arrays Battery Pkging Gimbals	Tankage Plumbing

- Separation of Engineering, Manufacturing, Test responsibility
- Overlapping hardware and software responsibilities
- No total product responsibility, no ownership

F-SAT PROGRAM

PROCUREMENT

BUSINESS OPERATIONS

MECHANICAL SYSTEMS

DESIGN
MANUFACTURING
COMPONENT TEST
    STRUCTURE
    PROPELLANT TANK
    PLUMBING
    MECHANISMS
    SOLAR ARRAY
    BATTERIES
    SPACECRAFT THERMAL CONTROL
    SYSTEM STRUCTURAL ANALYSIS
    LOADS
    WEIGHTS

SPACECRAFT OPERATIONS

REQUIREMENTS - SYSTEM/SUBSYSTEM
LIAISON - INTERNAL & EXTERNAL CUSTOMERS
SYSTEM ANALYSIS/ENGINEERING
POWER SYSTEM ARCHITECTURE/ANALYSIS
GUIDANCE & CONTROL
    HARDWARE PROCUREMENT
    ANALYSIS/ARCHITECTURE
SYSTEM SPEC
SYSTEM OPERATING SOFTWARE
SYSTEM TEST PLANNING

LEADER

ELECTRONIC SYSTEMS

DESIGN
MANUFACTURING
COMPONENT TEST
    INTEGRATED ELECTRONICS
    POWER ELECTRONICS
    BATTERY CHARGE CONTROLLER
    INSTRUMENTATION
    CABLING

Staffing

Staffing

SYSTEM TEST

DESIGN/CONFIGURATION MANAGEMENT

- **TOOLS**
  - MECHANICAL - COMBINED CIEM (ANALYSIS) AND CADAM (DRAFTING). CROSS-COMMUNICATION
  - ELECTRONICS - MENTOR GRAPHICS SUPPORTED BY SELECTIVE BREADBOARDING

- **DRAWING FORMATS**
  - DESIGNER PHONE NUMBERS ON DRAWINGS
  - ISOMETRIC VIEW ON DRAWINGS
  - CRITICAL INSPECTION DIMENSIONS MARKED ON DRAWING

- **DRAWING RELEASE/CHANGE CONTROL**
  - LIMITED CONTROLLED COPIES (3)
  - CHANGE AUTHORITY FOR EACH PART - SOLE AUTHORITY TO MAKE CHANGES
    - RESPONSIBLE FOR ASSESSING, REPORTING COST IMPACT (ONE MAN CCB)
    - HIS CALL - AMOUNT AND TYPE OF FUNCTIONAL SUPPORT
    - RESPONSIBLE FOR MARKING CHANGES ON ALL 3 CONTROLLED DRAWINGS
  - MANUFACTURING APPROVAL PRIOR TO RELEASE
  - RED LINE CHANGES ONLY. NO EO'S ALLOWED. MUST INCORPORATE AFTER 1 MONTH.
  - SAME REV LETTER - PARTS LIST AND DRAWING
  - QA AUDIT OF SYSTEM

- **DESIGN/DEVELOPMENT PHILOSOPHY**
  - "READY, FIRE, AIM, RELOAD," NOT "READY, AIM, AIM, AIM . . . . ."
  - DON'T "CONTROL" NUMBER OF CHANGES. ENCOURAGE EARLY SURFACING, CORRECTION OF PROBLEMS
  - ENCOURAGE HARDWARE/SOFTWARE/TEST MIX OF ASSIGNMENTS

PROCUREMENT/MATERIEL

- **SUPPLIER INTERFACE** - **LISTEN!**
    - **SUPPLIER COMMENTS ON DRAFT SPECS, RFPS, REVIEW OF FINAL VERSIONS**
        - **USE _THEIR_ SPECS WHEN WE CAN**
    - **UNLIMITED PRE-RFP ENGINEERING CONTACTS**
    - **DON'T JUST ASK - _BEG_ FOR BENEFICIAL EXCEPTIONS, CHANGES**
    - **MAXIMUM ACCEPTANCE TEST BY SUPPLIER**

- **SPECS/DOCUMENTATION**
    - **NO INTERNAL LOCKHEED SPECS/PROCESSES/STANDARDS - USE MIL SPECS OR GENERAL ACCEPTED STANDARDS**
    - **USE SUPPLIER'S INTERNAL FORMATS**

- **OPERATING METHODS**
    - **TREAT SUBS AS PARTNERS**
    - **DELAYS ARE MUCH MORE COSTLY THAN WASTED MATERIAL**
        - **ORDER MATERIAL BY AMR BEFORE DRAWING RELEASE IF POSSIBLE**
        - **IF YOU'RE NOT SURE WHICH MATERIAL YOU WANT, ORDER BOTH**
        - **PROVIDE EXTRA RAW MATERIAL TO THE SHOP. RETURN UNUSED MATERIAL.**
        - **USE OF PETTY CASH WHEN NECESSARY TO SAVE SCHEDULE**
    - **REJECT THE PROPOSITION THAT NON-PROCUREMENT PEOPLE ARE INCAPABLE OF SUBCONTRACTOR INTERFACING**

MANUFACTURING

- **ALL NEW PAPERWORK SYSTEM**
    - BASIC - MINIMUM DOCUMENTATION OF WHAT WE'RE <u>GOING TO DO</u> - MAXIMUM DOCUMENTATION OF WHAT WE <u>DID</u>
    - SIMPLIFIED TRAVELER - "BUILD TO PRINT" PLUS BRIEF INSTRUCTIONS, ROOM FOR S/Ns, ETC.
    - PHOTO RECORDS OF 1ST BUILD BY ENGR/TECHNICIAN - SCAN INTO COMPTR FOR 2ND BUILD
    - INSPECTION SHEET WITH TRAVELER. 100% 1ST BUILD, THEN CRITICAL DIMENSIONS ONLY.
    - NC MACHINE DATA RETAINED WITH 1ST BUILD DATA PACKAGE
    - MANHOUR ESTIMATES, UPDATES, ACTUALS BY DOERS

- **SHOP OPERATION - EMPHASIS ON SPEED, FACE-TO-FACE COMMUNICATION**
    - F-SAT MARKER ON ALL TOTES - EASY TO SPOT IN SHOP
    - BRIEFING OF ALL 1ST TIME F-SAT MANUFACTURING TECHNICIANS
        - WHAT DOES PART/ASSEMBLY DO?
        - PROGRAM DESCRIPTION, POSTER TO TAKE HOME
        - DISCUSSION ON BEST FAB APPROACH
        - TECHNICIAN CREATES AND/OR APPROVES PROCESS TRAVELER INSTRUCTIONS
    - FAST RESPONSE
        - DEDICATED STATION WAGON FOR FAST MATERIAL RETRIEVAL
        - IMMEDIATE ENGINEERING RESPONSE TO QUESTIONS/PROBLEMS - DAY OR NIGHT
            - NO LIAISON ENGINEERING
    - FULL SCALE FOAM BOARD MOCKUP BY SAME TECHNICIANS WHO WILL BUILD

- **TOOLING IN CONTROL OF USERS**
    - NEED FOR TOOL, DESIGN CONCEPT APPROVAL BY USING TECHNICIANS
    - TOOLS BUILT BY USERS

QUALITY ASSURANCE

- **BASICS**

    - · F-SAT QA MANAGER IN COMPLETE CHARGE OF **ALL** QA FUNCTIONS

        - INCLUDES INSPECTION, PROCUREMENT, SUPPLIER PRODUCT CONTROL, RECEIVING

    - MAXIMIZE ACCEPTANCE TEST BY SUPPLIER. DON'T REPEAT.

    - EMPHASIS ON AUDIT OF CONFIGURATION CONTROL, OTHER PROCEDURES - NOT SERIAL APPROVAL OF EACH ACTION

- **INSPECTION**

    - HEAVY EMPHASIS ON SELF INSPECT WITH QA AUDIT

    - LATEST POSSIBLE INSPECT POINTS

        - EXAMPLE - PC BOARDS. 1ST INSPECTION AFTER BOX TEST FOR OK TO COAT.

    - RECEIVING INSPECTION IN BLDG. 159. WE KNOW WHAT WE WANTED WHEN WE ORDERED IT. OTHERS DON'T.

- **MRB**

    - ON LOW-COST PARTS, DELEGATE "SCRAP" MRB DECISION TO MFG TECHNICIAN

        - LOG EVENT, HOURS SPENT, REPLACEMENT NEEDS ON BACK OF TRAVELER

    - MRB ACTION APPROVAL - PA AND CHANGE AUTHORITY ONLY. NO LIAISON.

FACILITY

- CO-LOCATE ENGINEERING, PROGRAM OFFICE, QA, ASSEMBLY, AND TEST IN B/159
  - DETAILS (PARTS, BOARDS) STILL BUILT IN CENTRAL MANUFACTURING
  - BUT WITH FACE-TO-FACE INTERFACE WITH DOING PERSONNEL

- SMALL ELECTRONICS AREA, MACHINE AREA FOR FAST PART FAB, REPAIR, CHECKOUT IN B/159

- NO COMPUTER "ROOMS." OPEN WORK AREA - ENCOURAGE INTERCHANGE.

"PEOPLE" MANAGEMENT

- **CULTURE** - MAKE PEOPLE FEEL **BIGGER**, NOT SMALLER
  - TOTAL ACCESS BY EVERYONE TO EVERYONE
  - DEEMPHASIZE ORGANIZATION CHARTS
  - EVERYONE GETS BUSINESS CARDS - SAME TITLE ON ALL (F-SAT)
  - NO MARKED PARKING PLACES

- **TYPE OF PEOPLE** - JUDGE ON **ATTITUDE**. RUTHLESS TOWARD "BAD APPLES."

- **SKILL MIX** - NO BAD VERBS (REVIEW, MONITOR, COORDINATE, ETC.) NEED APPLY

- **COST CONTROL** - MORE RESPONSIBILITY TO FEWER PEOPLE (THE ONLY **REAL** COST REDUCTION TECHNIQUE)

- **APPROVALS** - MINIMUM. "IF YOU CAN'T DO IT, YOU CAN'T REVIEW IT. IF YOU CAN DO IT, WHY DON'T YOU?"

- **COMMUNICATION** - SUBSTITUTE FACE-TO-FACE FOR PAPER

- **TRUST** - A MUST. IF PEOPLE LET US DOWN, DON'T USE THERAPY (LAYERS OF REVIEWS), USE SURGERY (GET THE HELL OUT OF B/159).

THE PILL

Here's an excellent plan to increase organizational efficiency. Issue each employee a small pill box with a lethal pill inside. At the end of each work day, employing a lie detector to preclude cheating, each employee must honestly answer the question "Have I accomplished more work than I have caused?"

If the answer is no ---- take the pill!

Project Management Lessons Learned on SDIO's Delta Star and Single Stage Rocket Technology Programs

6 May 1992

Paul L. Klevatt
Deputy Program Manager, SSRT
McDonnell Douglas Space Systems Co.
(714) 896-4618

N92-33324

Agenda

❑ **Delta Star (Delta 183) Program Overview**

❑ **Lessons Learned**

   - Imperatives

   - Proven Rapid Prototyping Groundrules and Axioms

   • **Management, Design, and Control**

   • **Cost and Schedule**

❑ **Rapid Prototyping and the Single Stage Rocket Technology (SSRT) Program**

❑ **Concluding Remarks**

Acknowledgement

The author is indebted to the following individuals whose experiences, ideas, writings, advice and can-do management approach have been a key factor in the successes achieved on important SDIO/MDSSC programs. The "lessons-learned" contained herein, gleaned from program achievements and setbacks, provide valuable guidelines for present and future Rapid Prototyping programs.

❑ **DELTA STAR**

- **Dr. Michael Griffin - NASA (formerly SDIO)**

- **Col. Michael Rendine - (SDIO/USAF(Ret.))**

❑ **SSRT**

- **Lt. Col. H. "Pat" Ladner - SDIO**

- **Major Jess Sponable - SDIO**

165/PLK2/Jc

# "The Difference Between a Dream and a Goal is a Time Limit"

❑ **Basic Objective of SDI Space Experiment Programs:**

- **<u>Quickly</u> reduce key uncertainties to a manageable range of parameters & solutions**

  ● **Yield results applicable to focusing subsequent research dollars on high payoff areas.**

❑ **Success is achieved via a partnership relationship amongst all participants conducted in a Rapid Prototyping environment**

COMPLEX AND DEMANDING SDIO MISSIONS
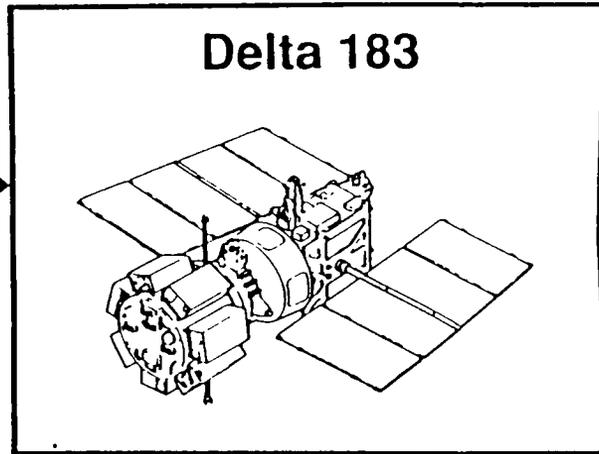# SUCCESSFULLY ACCOMPLISHED
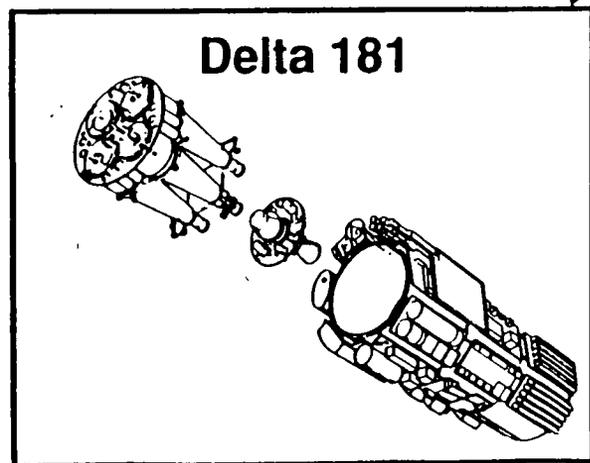# WITHIN SHORT SCHEDULES

VCZ21316 T10DA

Delta 180
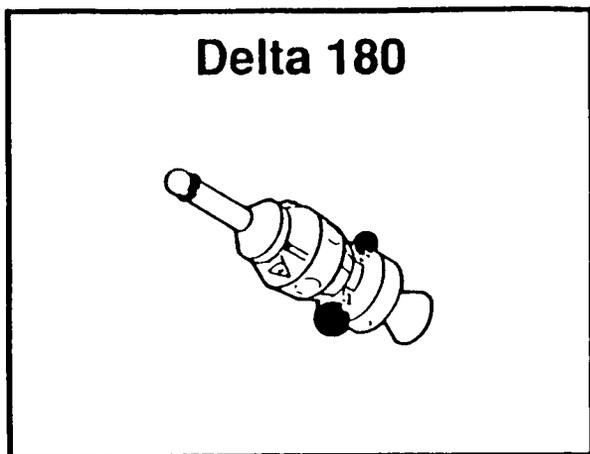
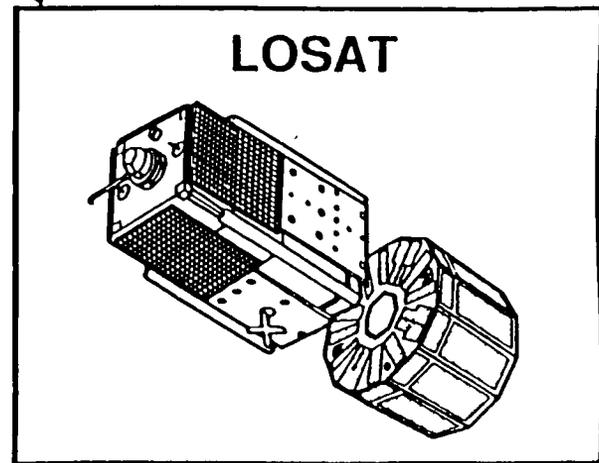**ATP to Launch
14 Months**

Delta 183

**ATP to Launch
13 Months**

Delta 181

**ATP to Launch
18 Months**

LOSAT

**ATP to Launch
13 Months**

243
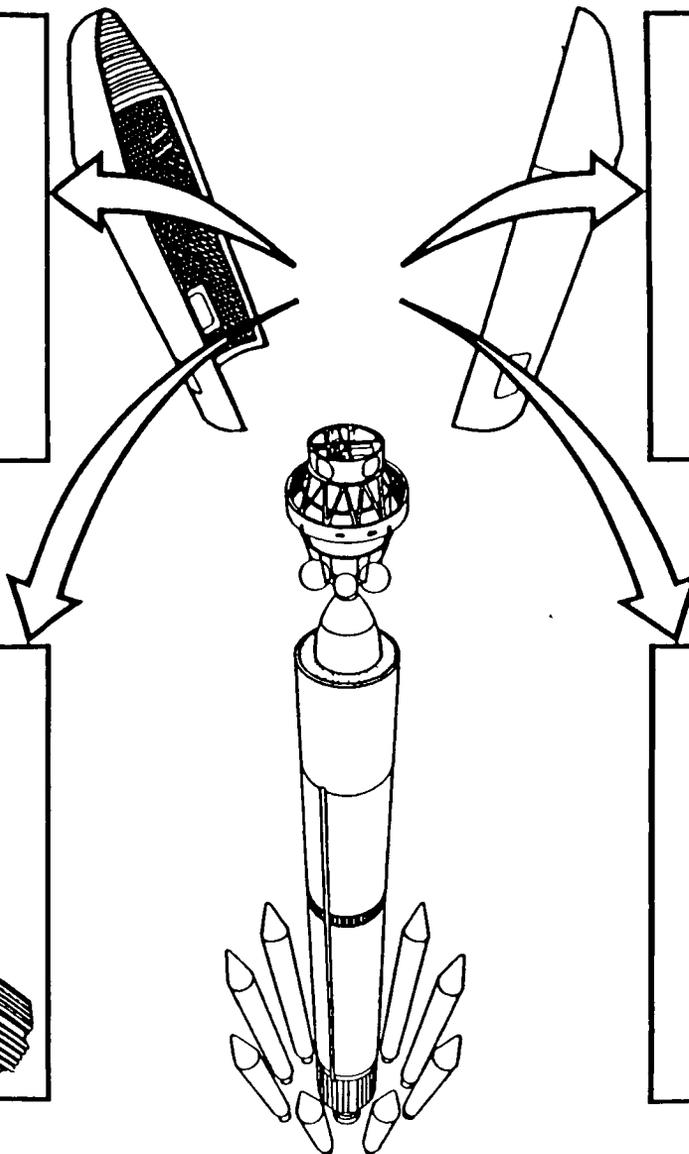
DELTA STAR PROGRAM
SCHEDULE OVERVIEW

DELTA ☆

MCDONNELL DOUGLAS

DS-VDG18953 1 M300

	1988											1989				
	J	F	M	A	M	J	J	A	S	O	N	D	J	F			D
Contract Start		▲8															
Sensor Selection		▲10															
Concept Review		▲12															
Design Review				▲2													
Spacecraft Fabrication & Assembly																	
APL Sensor Design Review				▲11, 12													
System Design Review					▲4, 5												
Revised Mission Baseline						▲18											
Spacecraft System C/O and Test at MDSSC									▨								
OOCA/SM Integrated Testing at MDSSC										▨							
OOCA Thermal-Vacuum Test											▨						
Telemetry Verification Testing											▨						
Mission Integration and Readiness Testing												▨					
Spacecraft at CCAFS											▲6						
SP-1 Mate to Spacecraft											▲14						
Spacecraft System Checkout at CCAFS											▨						
Spacecraft Mate to Delta Booster												▲18					
SDIO Directors Review												▲26					
Launch													▲24				
On-Orbit Operations														▨	▨	▲	

The Path To Program Success Begins With An Endorsement of Fundamental Needs & Priorities

McDonnell Douglas

☐ <u>IMPERATIVES</u>

A. Customer
- <u>Short,</u> concise SOW and contract - defining mission objectives, functional goals, schedule
- Agreement on funding profiles and funding timeliness
- Agreement on specifications at contract negotiation/award
- Clearly established guidelines for on-site government agency roles/participation
- Assurance of timely approvals re. test/qualification requirements and results
- Rapid access to Program Director for on-the-spot problem solving

B. Contractor
- Company President's commitment to Rapid Prototyping operations within the "system" to ensure:
  - Resources availability
  - Program priority
  - Team Program Manager's responsibility, authority and accountability

Proven Rapid Prototyping Groundrules and Axioms

A. Management, Design, and Control

1. Risk Management:

❑ Recognize differences between space <u>research</u> experiment programs and <u>operational</u> space programs.

<u>Space Research Experiment</u>	<u>Operational Space Program</u>
High Probability of Success	Minimize/Eliminate Probability of Failure

❑ Make decisions early on need for incorporating independent experiments to ensure high probability of success

Proven Rapid Prototyping Groundrules
and Axioms

A. Management, Design, and Control (Cont'd)

2. Requirements/Goals

❑ **Establish optimistic & challenging technical and schedule requirements and goals**

- **Allow for flexibility depending on hardware/software availability**

❑ **Select only those Mil-Specs & Mil-Standards that match the intent of the program.**

- **Allow flexibility & tailoring**

  • **The experiment just has to work**

❑ **Do not allow ICD's to drive the program - be flexible**

- **Change the interfaces to allow the most suitable teammate to solve the problem**

Proven Rapid Prototyping Groundrules & Axioms (Cont'd)

A. Management, Design, and Control (Cont'd)

3. Decision Making:

❑ **Delegate responsibility to those closest to the problem**
- Working groups (government & industry) produce timely "can-do's"
  - Reduces external monitoring & oversight
  - Replaces/augments the ICD
  - Tracks action items/documents with minutes
- TQMS's Concurrent Engineering principles a <u>must</u>
- All applicable functions/disciplines committed to first time quality

❑ **Use real-time problem solving (at team and management levels)**
- Come to closure rapidly
- Have meetings only when decision makers are present
  - Reinforces responsibilities

Proven Rapid Prototyping Groundrules & Axioms (Cont'd)

A. Management, Design, and Control (Cont'd)

4. Design:

❑ **Work with the concept - go rapidly to the Initial Design Review**

- Conduct timely focused systems engineering analyses & trades - keep documentation at a minimum, but thorough

- Surface problems and resolve quickly

- Maximize use of off-the-shelf hardware & software items
    - "Imagineer" their original use/intention vs. program's performance goals (performance goals are flexible!)
    - Insist on "touching" the H/W & S/W before commiting schedules

- Quickly establish hardware quantity and usage list

- Maximize use of existing embedded facilities and launch operations resources & procedures

Proven Rapid Prototyping Groundrules & Axioms (Cont'd)

A. Management, Design, and Control (Cont'd)

5. Design Reviews:

❏ Replace MIL-Spec. type PDR & CDR with participative Initial Design Review (IDR) and Final Design Review (FDR) approach

- IDR demonstrates design and development testing appropriateness

- FDR freezes overall mission/experiment design; establishes basis for configuration management/control

- Fosters teamwork & commitment to schedule; enables reviewers to be part of the solution, reinforces the benefits of cultural change

❏ Encourage executive level participation

- Reinforces their commitment

Proven Rapid Prototyping Groundrules & Axioms
(Cont'd)

A. Management, Design, and Control (Cont'd)

6. Qualification / Acceptance Testing:

❑ **Emphasize testing, especially at the subsystem and system levels**

- **Conduct timely reviews and move on; document findings**

❑ **Integrate subcontractor / supplier testing to reduce cost and schedule**

❑ **Use proto-flight approach (vs. full Mil-Spec style qualification)**

- **Reduces schedule risks, provides hardware use flexibility**

❑ **Do not subject available H/W to multiple +6dB test cycles**

- **Only flight safety, ascent, housekeeping functions and selected purpose-designed H/W need to meet a 3 sigma case**

- **Use tailored test procedures / hardware insulator / isolator systems to avoid over testing**

❑ **Use contractor procedures; Q.A. audits ensure adherence**

Proven Rapid Prototyping Groundrules & Axioms (Cont'd)

A. Management, Design, and Control (Cont'd)

7. Team Co-Location

❏ Co-locate all team functions, technologies, & disciplines

- Organize into concurrent engineering teams to develop a sense of "ownership"

- Make Liberal use of E-Mail, FAX, telecommunication with customer & teammates

- Include teammate/supplier representatives

❏ Minimize team size

- Promote team spirit/reward performance

8. Control

❏ Short, daily actions and status-oriented staff meetings with functional heads/decision makers

❏ Weekly meetings with customer to review progess issues, actions, and expenditures

Proven Rapid Prototyping Groundrules & Axioms (Cont'd)

McDonnell Douglas

B. Cost and Schedule

1. Requirements

- ❑ **Use contractor cost reporting and control processes**
  - **- If over key thresholds, use CSSR not full C-Spec**
    - • **Cost accountability to WBS Level II**
- ❑ **Reduce the number of CDRLs**
  - **- Provide visibility via government / industry working groups**
  - **- Limit submittals to only those needed to control basic requirements, cost, schedule, and system / personnel safety**
- ❑ **Make liberal but judicial use of commercial parts and processes**
- ❑ **Baseline "redlined" drawings / procedures; cleanup/ incorporation of changes not required**
- ❑ **Quickly baseline and freeze program master schedule/major milestones**
  - **- Use "work-arounds" in lieu of schedule slips**
- ❑ **Put heavy reliance on subcontractor's inspections and final buyoffs; strive to reduce contractor's receiving inspection to only parts I.D. and quantities checks**

Proven Rapid Prototyping Groundrules & Axioms (Cont'd)

B. <u>Cost and Schedule</u> (Cont'd)

2. <u>Implementation</u>

❑ **Perform all critical functions within Rapid Prototyping Dept.**

- **Contract Management**
- **Subcontractor/supplier negotiators/administrators**
- **Requirements, H/W, S/W and schedule configuration control**
- **Computer aided analysis, design, drawings, autocoding**
- **Drawing check, release & control**
- **Procurement**
- **Receiving/shipping authority**
- **Control booths/stockrooms**
- **Production control/follow-up**
- **Material reviews/dispositions**
- **Fabrication/tool orders**
- **Facility planning and design**
- **GFE acquisition**

165/PLK2/lc

DELTA Clipper

The "lessons learned" from prior SDIO/MDSSC Rapid Prototyping programs are an integral part of the Single Stage Rocket Technology program. We continue to learn by "doing" and to apply new knowledge gained to a <u>continuous</u> process of improving our management approach techniques and processes

"Hardware Flying Fulfills the Promise of our Vision"

Specific Guidelines Have BeenApproved For The SSRT Rapid Prototyping Department

SSTO

RAPID PROTOTYPING DEPARTMENT GUIDELINES

DELTA Clipper

Superior Space Transportation Through Quality

MCDONNELL DOUGLAS SPACE SYSTEMS COMPANY
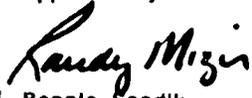ADVANCED PRODUCT DEVELOPMENT AND TECHNOLOGY

MDSSC-GUIDELINES

for

SINGLE-STAGE-TO-ORBIT

RAPID PROTOTYPING DEPARTMENT

Submitted by:
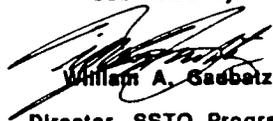
William A. Gaubatz
Director, SSTO Programs

September 1991
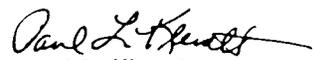
Paul L. Klevatt
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Approved by:

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DC-X Objectives Support Substantiation of Single Stage Rocket Technologies

DELTA Clipper

❑ **System operability supportability**
- ● **Airline type operation**
- ● **Support systems and procedures development**

❑ **Rapid system turnaround**
- ● **Mobile flight operation and ground support functions**
- ● **Small operations crew**

❑ **Software rapid prototyping/modularization**
- ● **Reduced development costs with increased reliability**
- ● **Flexible mission planning**

❑ **Controllable vertical take-off and landing**
- ● **Blended control**

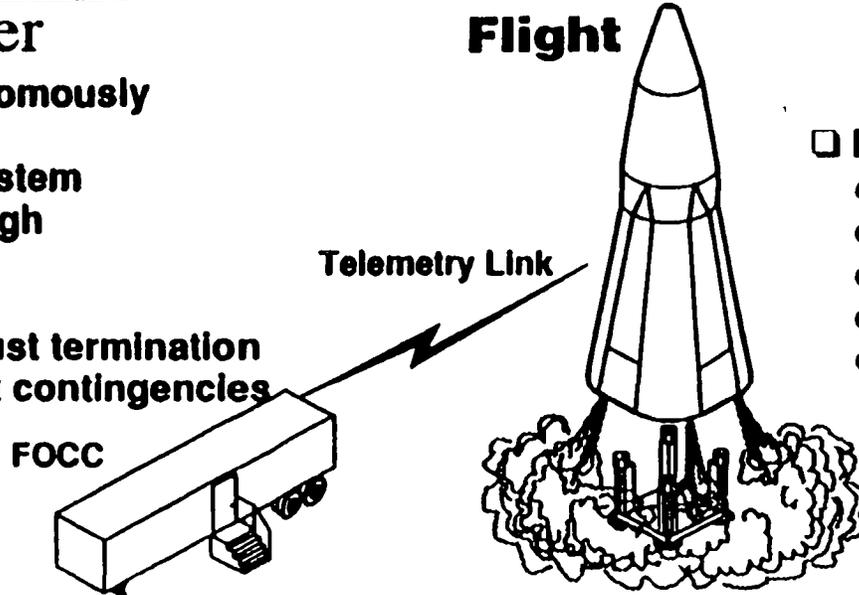❑ **Controllable rotation maneuver**
- ● **Typical of operational system**

OPERATION PROVIDE LEARNING FOR SINGLE STAGE ROCKET TECHNOLOGY

DELTA Clipper

❏ Vehicle flies autonomously

❏ FOCC monitors system performance through Telemetry link

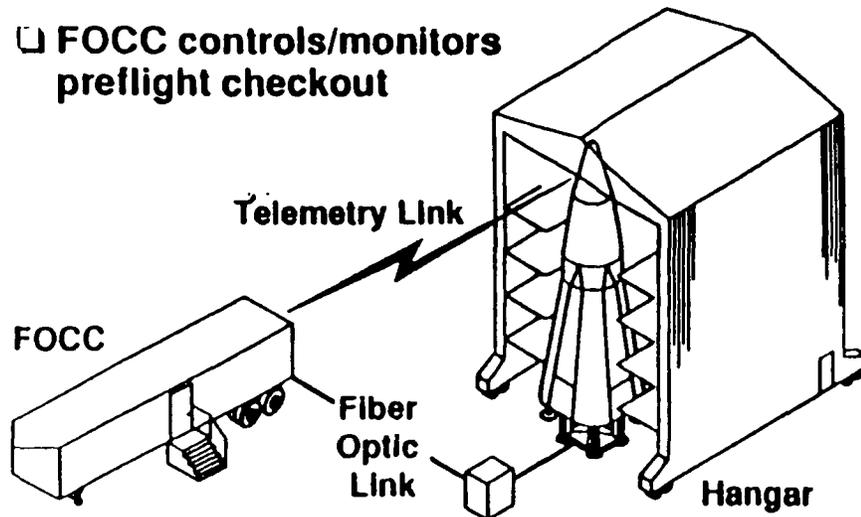❏ FOCC initiates thrust termination system for in-flight contingencies

FOCC

Telemetry Link

Flight

❏ Flight vehicle demonstrates
 ● Vertical takeoff and landing
 ● Key rotation maneuver
 ● Blended control
 ● Verification of reflight capability
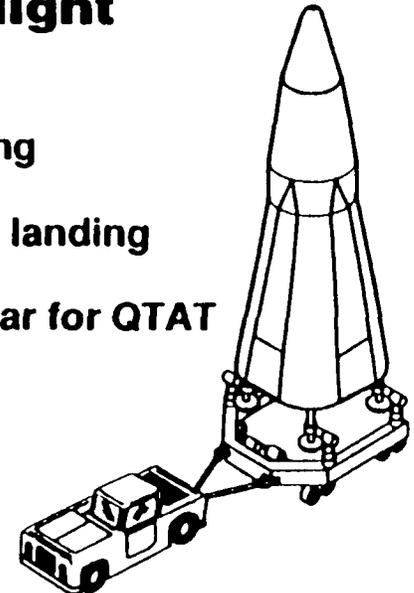 ● GPS updating

Preflight

❏ FOCC controls/monitors preflight checkout

Telemetry Link

FOCC

Fiber Optic Link

Hangar

Postflight

❏ Autonomous securing

❏ FOCC confirms safe landing

❏ Towed back to hangar for QTAT

Concluding Remarks

❏ **Lessons learned from quick, productive, cost efficient past and present SDIO/MDSSC Rapid Prototyping programs provide a basis for achieving technical & programmatic goals and objectives**

  ● **Key principles are fundamentally identical to basic TQM tenets:**

    - **Build in quality, empower the work force, build teams & commitment**

❏ **Transferring lessons learned from one organization or program to another requires:**

    - **A total team (management & work force) paradigm shift to a new way of doing business**

        ● **Expect some early "cultural" shocks, but also expect some near-in, rewarding successes!**

P-12

AIAA 92-1706
The Impact of New Business
Approaches Derived from the Manned
Transportation System Study

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AIAA Space Programs
and Technologies Conference
March 24-27, 1992 / Huntsville, AL

THE IMPACT OF NEW BUSINESS APPROACHES
DERIVED FROM
THE MANNED TRANSPORTATION SYSTEM STUDY

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Abstract

A survey of senior contractor and NASA management was conducted to determine those factors perceived as having the greatest impact on the aerospace industry's ability to do business with the Government, specifically NASA. The results, both critical and laudatory, are categorized and discussed herein. It is anticipated that a follow-on paper will address specific means of and agencies for alleviation of the greatest impediments.

Background

The Manned Transportation System (MTS) study contracts are being conducted by the New Initiatives Office of the NASA Johnson Space Center. The objective is to use past work and current data to provide a framework for determining the right path to follow for an integrated manned transportation system using a logical, measurable, and repeatable process. A NASA-Industry Team (NIT) was formed to obtain consensus on the needs, attributes, and top-level requirements for manned transportation and to provide technical data to aid NASA in determining the right path to follow. The NIT consists of the following NASA participants - Johnson Space Center, Headquarters, Marshall Space Flight Center, Langley Research Center, and Kennedy Space Center; and the following industry participants - Lockheed, Boeing, Martin Marietta, Rockwell, General Dynamics, and McDonnell Douglas.

One of the Study objectives is to identify "better" ways of doing business with the Government that would aid in developing and operating a more affordable, reliable, safe, and routine manned transportation system. The business areas identified to meet the Manned Transportation System objectives were: procurement, management, organization, policy/procedures, budget, personnel, and operations.

A survey was conducted among senior managers within the government and participating companies to obtain information in these selected business areas (see Appendix A). The main goal of the survey was the identification of items that could improve industry's way of doing business with the Government. Over one hundred suggestions were received; the categorized responses of the survey are depicted in figure 1. Each suggestion was assigned to one of the selected business areas (categories) so that those with the most concern could be easily determined. The summary of the responses are presented by category in the following sections. Most responses pertain to what is wrong with the current way of doing business, rather than improvements to the system. But where suggested improvements or solutions to specific problems were identified, they wereincluded in

the summation. *The NIT did not necessarily agree with all of the responses received, but the summation reflects the unchanged context thereof.* This has also resulted in the occasional occurrence of somewhat stilted language herein,

for which the reader's understanding is requested. Before the study is finished, the NIT will develop and promulgate a strategy for the attempted removal of the impediments.
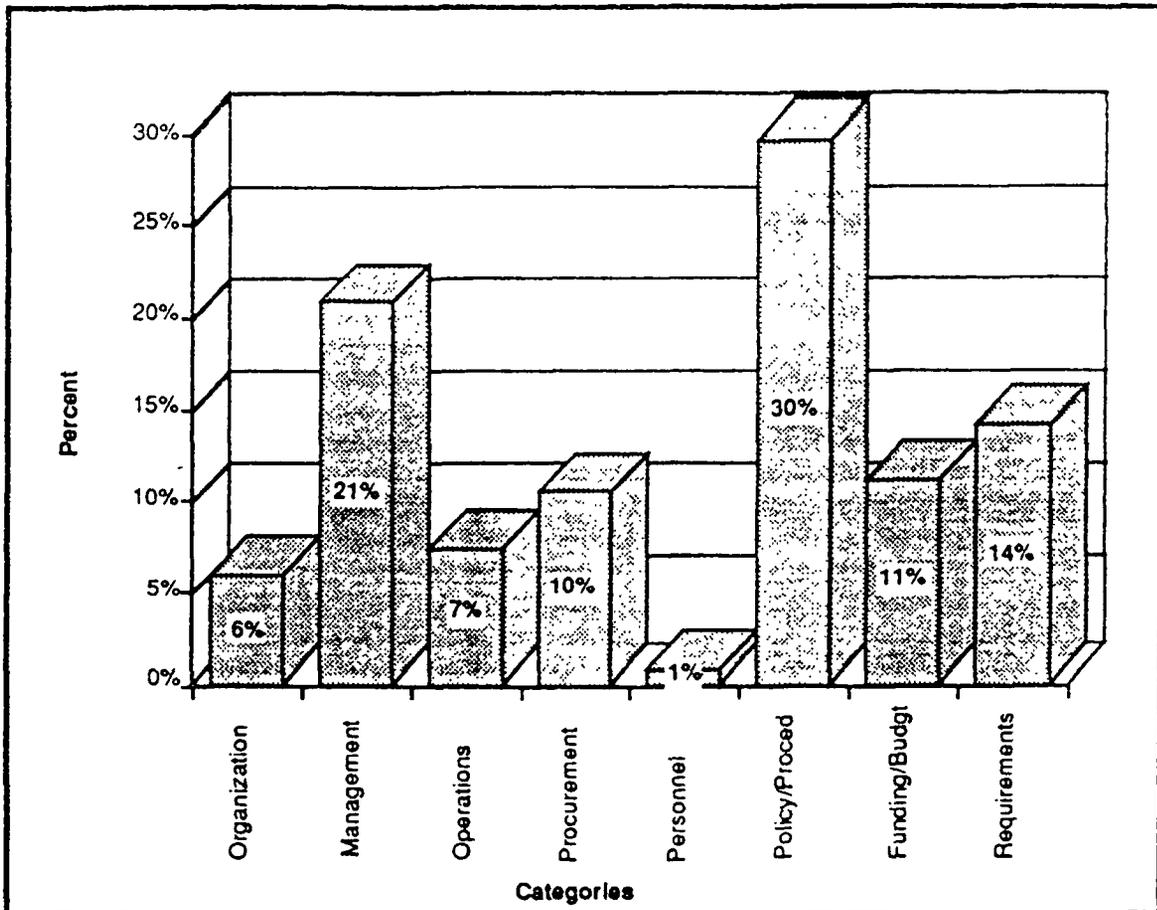

Figure 1 - % of categorized survey responses on better ways of doing business with the Government

Budget

Program funding constraints can cause several things to happen. For example, test hardware may be forced to be deleted and designs may be changed resulting in much higher operational costs. Emphasis on low cost is perceived to be at the expense of on-time schedules and technology advancement. Cost and budget estimates have a significant influence on program stability and outcome. The lack of multi-year funding inhibits planning for orderly and efficient development of operational capability. Annualized funding is so variable that contractors expect to cost share in order to get around the uncertainties of the Government. Programs become longer and longer due to such

constraints, which makes them more costly overall. The detailed involvement of Congress in the budgeting process (*e.g.*, redesigning Space Station Freedom {SSF}) and the resultant contractor response to reduced budget levels cause early program inefficiencies. Political constraints affect the budget of NASA acquisitions and cause many restructuring problems.

Timely funding of fiscal year options is hindered because of tendencies within the appropriation and authorization processes to transfer NASA-budgeted funds to other agencies. This often results in work stoppages, delays of scheduled launches, and increased overall costs.

There is enormous pressure at the onset

2

of a program to assume high levels of cost risk without adequate reserves to cover contingencies or growth. One recommendation is to delay the start of a program until cost estimates and budget availability match. Program budgeting should recognize program dynamics from the outset and reflect "looking back" costs. Reserves should be budgeted after the originally predicted peak cost point.

Management

New management practices must be introduced. To reduce costs and meet tighter end-item delivery schedules, oversight and review of projects must be sharply reduced, and authority must be delegated to those closest to the problems to allow them to effect the solutions. There is a need to streamline and reduce the number of customer reviews and meetings. Top management time is consumed by lack of delegation and excessively broad program reviews which do not concentrate on key issues. Meetings for information only that do not address any specific problems should be minimized. When meetings are held, the decision makers should maintain open lines of communications, and maximize productive time. To save costs, telecommunications should be used to reduce travel and facilitate participation by those closest to the technical problems.

Management needs to assign clear responsibility, goals, and commensurate authority to each job assignment so that the responsible person(s) can see that the job gets done. Clear goals will focus the efforts to adhere to schedule and avoid lost time. Government management needs to specify the deliverables of the program, rather than how to achieve those deliverables. Mission objectives should be defined and the technical solutions should evolve as technical problems arise. This allows people the creative flexibility in their approach to problems which leads to the most cost-effective solutions.

The lines of communication should be open between Government and the contractor. Contractors should be treated as team members in open discussions. If continuity can be maintained within the program team (NASA and contractor) the following will happen: the team will be well-informed; time will be saved on training new team members; increased

cooperation and enthusiasm for the program will be generated; and team members will have recognition for their individual efforts. A sense of trust among Government, industry, and team members must be established to allow the members to push ahead decisively and to reduce barriers. Each member must be able to rely on support from the others. The high degree of interaction between NASA and its contractor, while technically productive, also tends to place upward pressure on the cost outcomes.

Government management should select the contractor and then let it perform design development. The contractor should have more up-front responsibility, using clearly defined requirements and goals set by management, to perform its assigned role of design development. Program direction should emphasize the project accomplishment rather than reporting, documentation, justification, etc. Abortive procurements that continue to the point where the Request for Proposal (RFP) is expected any day and are then dropped, should be avoided to reduce the wastage of contractor resources which ultimately are paid for by the Government. A level of risk should be established to enable the Government to communicate the likelihood that funding will be available to consummate any given procurement.

Operations

There are outdated design/integration processes used today that concurrent design, systems engineering, and integrated product development teams should improve. Establishment of concurrent engineering teams to evaluate candidate designs and system architectures should reduce the complexity of interfaces during the design phase. These teams need to be established early in the program. A "skunk works" activity may be one way to effectively formulate the concepts and system definition on which the overall program development effort relies: production (logical manufacturing processes), operations (reduced manpower and documentation), specialty engineering (safety, quality, reliability, maintainability, etc.), and design.

Having a "Design-for-Operations" philosophy in the front end of a program can reduce overall acquisition and support costs. This is substantiated by quantitative modeling

techniques and by experience. The F-117A has shown a reduction of over 25% in operations costs based on this concept. The F-117A Program also used commonality of hardware and saved over $60M in DDT&E costs for avionic systems.

The Japanese approach to reliable product development is to engineer, in the product definition phase, both the design and the manufacturing process to provide a stable production approach and a product that is highly reliable. This "concurrent engineering" process produces a basic product design that will accommodate the normal statistical variance that can be expected from the manufacturing process. If the design and manufacturing process are properly developed together, a quality product can be built and statistical process control utilized rather than 100% inspection.

If design, fabrication, and operational processes for space hardware are put together using the following suggestion (e.g., launch vehicle), the results could possibly be a system with lower costs and greater reliability than any existing element of space hardware. The development team must establish an approach for the concurrent engineering of the element that will assure, to the maximum extent possible, a producible and reliable design. Before the hardware design of the element is started, an extensive analysis should be conducted of the functional operation of the total system to determine the design limits that must be placed upon all the critical subsystems/components to assure acceptable system functionality.

This effort first requires a functional flow analysis of all the subsystems that make up the total system. This analysis should flow the operational requirements down to the major component or Line Replaceable Unit (LRU) level. Next a consistent computerized systems simulation model should be developed and utilized to apply Taguchi's techniques to establish acceptable operational limits on the subsystems down to the same LRU level.

When these limits are known and an assessment of the operational environment has been made, concurrent engineering design studies for the LRU's can begin. These studies must include considerations for all elements of the launch system's life cycle. The product and process designs must result in LRU's that can be repeatedly built and operated reliably within the specification limits and with the only inspection being to assure that there is no human error in putting them together. The Mean Time Between Failure (MTBF) of the LRU's must be high enough that operational testing is not required to assure systems reliability.

A suggestion to minimize long term operating costs was to consider the impact and influence of logistics requirements on the system design early in the design phase of a program. The "blind spot" associated with inadequate front end analysis of logistics requirements results in an incomplete concurrent engineering process. The Department of Defense (DOD) major systems managers demand logistics assessments as a part of the concurrent engineering process, knowing the impact on long term operating costs. One obstacle encountered in implementing this suggestion was that funding constraints continued to reduce or cancel the logistics engineering analysis tasks.

It would greatly improve the implementation of the NASA management information data system if there was compatibility of computer hardware and software between NASA Centers. There could be an imposed standard of hardware/software requirements so that NASA computer systems can be compatible. The computers would be better utilized if there were more commonality.

Organization

An understanding of the division of authority between NASA Centers is often not clear. Multiple Center roles and responsibilities need to be complementary rather than overlapping. Standardization of business practices between Centers would greatly improve the efficiency of doing business. Paperwork is sometimes required by one Center for another Center that, in turn, actually demands something different.

Another area for improvement is when Level II wants all changes coordinated with the element for feasibility of concept approval

before a Level II Program Change Identification Number (PCIN) is processed. The Level III projects do not appear to want to listen to improvements or changes that are not within their current funding structure. After the PCINs were processed, Level II had to direct Level III to assess the changes, which took over a year to complete. Time is costly. To reduce the time, one suggestion might be for Level III to consider sponsoring the change if they become involved. It would also allow an independent evaluation of the element data.

Use the major prime contractor as the integrating contractor. Contract design through launch with no second or third parties involved (e.g., Shuttle Processing Contractor).

Within an organization, establish separate work centers focusing on one function or product with all supporting elements under the direction of the work center. There may be obstacles to overcome when co-locating some of the functional elements in the work centers due to the perception of where their traditional place is in the organization.

Procurement

The procurement processes are fundamental to how a program succeeds. A procurement approach is needed that; 1) is applicable even with "international partners," (2) can get work going within a few months, (3) expends only a small percentage of the resources on the effort of the procurement process itself, and (4) has a way to continue to utilize the capability that has been built-up during a competition. The process needs to find the best combination of capability, motivation, and low cost, and also to leave the losing competitors with somewhere to go and something to do.

The procurement system needs to be simplified and kept honest. One suggestion on how to keep it honest was to establish a type of referee system whereby all procurement decisions are made by people who are precluded from subsequent involvement with the companies involved. The policy should be made simpler with no contractor involvement in the development of the statements of work. This includes ₊support contractors -- competitive procurements should be fair to all.

The procurement "boilerplate" needs to be streamlined. A large amount of effort is spent answering untailored specifications. It takes too long to get through all the steps to receive approval on procurements, both initial and modifications. Reduction in reporting requirements would simplify and keep costs down within the program. The Government could take advantage of the contractors' reporting systems to reduce or eliminate specific government reports.

The NASA Research Announcement (NRA) is a good approach for small studies and is a step in the right direction for larger contracts. The use of the NRA has resulted in less than a 30-day turnaround of award from proposal receipt from the contractor and streamlined the process of getting the contractor on board earlier. Level of Effort (LOE) contract types are good for increased flexibility. In all contracts, there needs to be an easier change mechanism because it takes too long and involves too many people.

Development of new systems should not be competitively priced. In fixed priced developments, the contractor is forced to throw out things that can be significant (e.g., testing).

The imposition of a Performance Measurement System (PMS) on a one-of-a-kind type of DDT&E program (e.g., SSF) is not wise. PMS does well with a production program and products that are well defined.

The cost of complicated procurement regulations unnecessarily raises the costs of launch services. Standardizing the planning system to reduce acquisition complexity may help keep the costs down. The current acquisition process forces unrealistic cost schedule submission. Suggested solutions to improve the acquisition processes are to: (1) develop new cost estimation methodologies, (2) establish requirements early and conservatively, then avoid changes; (3) utilize multi-year authorizations and appropriations, (4) allow more flexible/realistic contract type selection, and (5) promote total quality management (TQM) at all levels.

Incentives for the contractors to meet or exceed the program objectives would help keep

costs low. For example, Rockwell International earned 20% of every dollar it saved NASA on building the Endeavor. Incentives could include: direct grants to develop new technology for systems specifically directed toward saving costs rather than increasing performance; cash incentives to firms for reducing the manufacturing costs of specific items procured by the Government; and encouragement of industrial teaming arrangements in focused technology areas such as the National Aerospace Plane Materials Consortium. In addition, the U.S. Government could stimulate the private sector's innovative creativity by issuing a request for proposal for space transportation services, and having industry bid on the end product (e.g., 4 seats to/from SSF every 90 days). Such an approach assumes minimum Government oversight over the design and manufacturing processes. It would also require the aerospace community to assume much greater financial risk than it has taken on in the past. In order to offset that risk, the Government would likely have to agree to a minimum purchase that would allow the companies involved to earn a profit on their investments.

Financial incentives passed through to the individuals working on a program would increase enthusiasm and motivation for working on the program. The individuals would be more personally responsible for the quality of their own efforts, and there would be less peer-tolerance of poor performers, who would otherwise dilute the financial incentives.

Personnel

The only suggestion that was received explicitly regarding personnel was to off-load people supporting development programs when the development is complete. This is an ingredient of a successful low-cost, high technology program, but should be coupled with a plan to retain or otherwise utilize the people within the company so that their expertise is available "on-call" as required.

Policy/Procedures

Lack of programmatic stability results in the wastage of resources to replan and in the loss of credibility of current schedules (caused by funding constraints, new requirements, etc.). The program planning process, in particular the cost and budget estimation processes, have a significant influence on the program's stability, and hence its outcome. The essential problem is that there is currently no process which formally connects policy and the budget. At the top level, there is a space program policy. That policy should be broken down into particular pieces of the space program, and then further broken down into Level 1 requirements. Eventually, the Level 1 requirements would get decomposed into lower level derived or implementing requirements. The policy and top level requirements would tell NASA what it has to do. On the other side, there is the budget, which reflects the monetary constraints on the job NASA has to do, as defined by the policy and top level requirements.

The solution is to develop and implement a process which links the budgets and the requirements. The link is especially important very early in the life of a program, but is required throughout.

NASA should start at the top: identify and prioritize what it wants to accomplish; what the "mission need" is; and what it would cost. These must be in harmony before proceeding further. Just as the generation activity of the technical requirements is recognized as being iterative, with the product improving with the number of iterations, the policy/requirements versus budget process should also be iterated until the desired quality of product and agreements are achieved.

The risk of not doing this is a vicious cycle of undesirable events: (1) people in control of the budgets don't trust us; (2) those who don't trust us tend to micro-manage us; (3) as they get into micro-management, they squeeze the resources or add their technical requirements to replace those we didn't have or didn't clearly enunciate; (4) as we get squeezed, we tend to take what we can get, since we find it difficult to stand fast to requirements which weren't clearly enunciated or which had poorly- or un-defined mission needs; (5) taking what we can get, instead of what we should have written down, further damages our credibility.

NASA needs to prove to the administration and to Congress that it can run multi-year programs in a cost-effective manner, particularly such programs as the Space Shuttle which

presently operate at levels of more than four billion dollars per year. Once NASA has reduced these costs and demonstrated this management capability and before it inaugurates new programs, it must make sure that it understands the top level needs, and that backers are available to support them with cash. Otherwise, these programs will be prey to multiple analyses and external micro-management.

While concept definition is fun for the participants, usually not enough work is done on accurate program planning and costing which should include supportability and even phaseout. Structured, recognizable, processes should be established which are consistent across the NASA and engineering contractor community.

It is felt that any program development can be accomplished in 3 to 4 years once uncertainties are resolved. The government should allow for more flexible contractual arrangements (less rigorous procedures and documentation). It was also recommended that the quantity of pre-phase A and B contractors be minimized.

Contractors complain that the cost of continued excessive government oversight and complicated procurement regulations unnecessarily raises the costs of launch services and/or programs. In the commercial sector, products or services are procured by the customer. The oversight in the production of those products or services is held to a minimum. In government contracting, the contractor engineering force is unnecessarily duplicated by the Government.

Purchasing launch services competitively from private firms, rather than managing launches from within NASA or the armed services, might well save money. The intent of purchasing launch services is to remove the Government as much as possible from setting detailed engineering specifications for that launch system and to reduce the burden of excessive oversight by Government managers. NASA could adopt the way the Federal Aviation Agency (FAA) does business in that they set the "airworthiness standards" and then let the industry design, develop, and qualify products to meet those standards while filling a need.

In streamlining the policy/procedure processes, a commitment to total quality management needs to be made. Some of the suggestions for the policy to incorporate are: (1) use statistical design and manufacturing process development to produce parts within the specification limits and to establish expected failure rates/modes; (2) have a "Design for Operations" philosophy in the front end of a program that would reduce overall acquisition and support costs; (3) minimize the levels of approval required for simple changes; (4) minimize formal contract deliverables; (5) decrease the time of the evaluation/definitization cycles for change orders; (6) confine review item discrepancies (RIDs) at preliminary and critical design reviews to design topics -- not requirements -- and do not change them between reviews; (7) automate the flight and mission planning systems and standardize vehicle loads to specific weights and centers-of-mass, which would save large amounts of manpower intensive planning; (8) establish documentation structures which accommodate the total program requirements definition.

NASA is perceived to hold too much work in-house. It appears that they prefer to do the conceptual and preliminary design work themselves, competing with the contractors for business. In this process, they change system requirements, the program objectives become cloudy, and the program frequently loses support. If the NASA Center's mission is to be the design center, then it should perform the design function and contract only for manufacturing, assembly, and testing where there is no in-house capability to accomplish these functions. The alternative is for NASA to hand the contractor a set of requirements and to allow it to design and provide a system that satisfies those requirements.

Low cost innovation can be encouraged by providing contractors with an incentive and giving them the autonomy to implement changes without a lot of red tape. By providing incentives to change, a culture of constant improvement can be created. The Government should consider technology transfer to the ones developing the product and providing more of the technology work effort. They should also insure that the technology is proven prior to the end of the program.

As manpower reductions on the contractor side take place as a result of implementing new ways of doing business, it is imperative that the Government reduce personnel proportionally. This would maximize the savings that result from such changes and also help the contractors to see that their efforts are matched and appreciated by their partner, the Government, in pursuing space goals. Positive accomplishments should be the primary determinants of new business and continued employment. If an area is cut, then the government employment should go down at least in proportion with the contractor's.

Government should consider entering into longer-term commitments with suppliers to purchase larger lot sizes. That could reduce the component unit cost substantially, which would directly benefit the competitive position and increase sales and profitability for the supplier. It would require some risk on either the prime contractor or the government. The Government would have to commit future budget funds which would reduce their budget flexibility. The contractor would have to take title to unsold goods with the expectation of adding value and reselling at a profit.

Requirements

NASA programs need to have a multi-tiered requirement system in order to unfold successfully. Starting with an objective from the President or upper management, each tier needs to come up with appropriate requirements, working on down to the smallest elements of the program. For example, a broad brush objective may be a permanent base on the moon, a goal set by upper management. This implies requirements for a transportation system, habitat, and other support elements. In turn, these elements must be defined for the number of people they transport or support on the surface, resulting in further requirements for lower tiers.

Such a functional decomposition has long been employed by military programs, and could be adopted more widely and consistently by NASA. With the broad top-level requirements determined, configuration control could be employed early to make sure that concepts for program elements address upper level

requirements Specifications must not drift off. once program elements start to drift away from the requirements, "You've lost the game." In the case of the military, a new system User Command will require a weapon system to counter a threat; the weapon system might be a fighter aircraft to counter ground-based weapons. The user group goes to the Systems Command to establish the definition and requirements for the weapon system, determining how it has to perform within a certain envelope. Specifications are then based on trades.

In the case of NASA development programs, in the Phase A portion of a program, contractors for one reason or another provide upper level designs instead of requirements. In the case of SSF, the requirements were set in Phase A studies, but they were set too broadly, or else disregarded to such a great degree that the Phase A contributed little substance to subsequent development of the project. When requirements for micro-"g" laboratory operations were imposed on the program, it was after the Phase A studies were complete, and without the needed configuration control. On the other hand, in the case of Apollo, the successful system engineering procedure was performed intuitively rather than formally.

It has been difficult to integrate payload or scientific requirements into the NASA engineering process, often because a multi-purpose vehicle attempts to integrate mutually exclusive requirements and because managers are not ready to say "no" to what users want.

Since requirements are the first products in any potential program, and since they are very important to the life of that program, NASA should spend more quality effort on this product. Ways to accomplish this are to include certifying requirement writers before they are allowed to write any and requirements "stamping" for certification -- much akin to the SR&QA stamps of approval -- to insure they are true requirements and not "desirements". There should be at least a center-wide, if not agency-wide requirements tracking and control "tool," and perhaps even a requirements organization to insure uniformity of the requirements within a program and across programs. The technical organization within a program should develop the parameters that

need to be controlled, and rationale for why they need to be controlled, so that discussions can be held involving all parties before actually discussing any quantified parameters.

The Government should define what it wants in a mission statement and establish the requirements. All requirements must be identified so that efforts are not wasted trying to satisfy unidentified requirements. It should let the contractor formulate the concepts and designs that meet the requirements, while providing the technology support required. The Government should review the concepts and designs (validating them against the requirements), advise, approve, and let the contractor implement the program. Once established, requirements should be changed only when absolutely necessary. All parties must stay focused on the mission statement instead of trying to meet excessive, sometimes conflicting, utopian requirements.

Summary

There is still much to be done on this topic. By the close of the study, the survey responses will have been prioritized by the manner in which they affect the way the new manned transportation system will be initiated, developed, and operated. Elements that have the authority to make improvements in the way we do business (*e.g.*, NASA centers, NASA Headquarters, Congress, *etc.*) will be identified and correlated with the required actions. A strategy will be developed to attack the major barriers to improvement and to implement the new ways of doing business into the next manned transportation system.

Solicitation

The survey form used for the solicitation of inputs in conjunction with this study is attached as Appendix A. Readers are encouraged to submit additional inputs for consideration by the team as it prepares the final study report. "White space" within the form proper has been somewhat compressed in order to stay within the page limits for reproduction in this format, so feel free to use additional sheets if required. Responses may be mailed to:

Bruce McCandless II
Mail Stop DC8001
Martin Marietta Astronautics Group
P. O. Box 179
Denver, CO 80201-0179

or FAX-ed to: (303) 971-5021. All inputs must be received prior to June 1st, 1992, for consideration.

"THE IMPACT OF NEW BUSINESS APPROACHES"
Task #4 of the
MANNED TRANSPORTATION SYSTEM STUDY

Contracts NAS-9-185xx {Point of Contact}
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...72: Lockheed {J. Kerwin: (713) 282-6204}
...73: Martin Marietta {B. McCandless: (303) 971-6308}
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...75: Rockwell {D. Bienhof: (310) 922-4918}
Sponsor: NASA Johnson Space Center {N. Lance: (713) 283-5508}

The MTS Study contract is being conducted by the New Initiatives Office of the NASA Johnson Space Center with the six industry participants indicated above. We are looking for a list of key impediments or new ways of doing business that you have encountered or are currently encountering in your experiences with Government contracts. Your input(s) will be combined with similar comments from other programs and functional areas across several contractors to focus efforts on how to improve our collective programmatic efficiency. A final NASA-Industry Team report, embodying the results of this survey, will be prepared, presented at appropriate levels within the NASA, and placed in the public domain.

Areas of interest include, but are not limited to, Organization, Management, Operations, Procurement, Personnel, Policy/Procedures, and Funding/Budgetary topics. Specific examples are useful for improving the readability of the report, but we are looking for broadly applicable material. Negative examples are acceptable, but the emphasis is on how to do more with what we have in the context of NASA-related business. Anonymity of organizations will be maintained in the final report(s) if such a desire is indicated above, but any information supplied will be available at the working level to all MTS Study contractors and participating Government elements. Additional pages may be added to this questionnaire at your discretion.

1.   Please identify the top three to five things that would (have) result(ed) in the greatest improvements in your way of doing business with the Government.

2.   Your Company/Organization:
3.   Program/Project/Functional Area:
4.   Point(s) of Contact for further info: Tel: (   )

5a. Is it O.K. to identify your Company? Yes/No   5b. - - your Program?   Yes/No
6.   Were you able to actually implement the above improvement(s)?   What obstacles were encountered?   How were these overcome?

7. What risks are involved in the foregoing? Do you have any suggestions for miti-gation?
8. Can you quantify the savings/level of improvement?

9. Approximately how large (dollars, man-months, or peak number of personnel) is/was your area of responsibility?

10. Was this a prime contract or a subcontracted effort? Were you teamed with any other aerospace contractor?

11. How would you assess the PLANNED schedule duration vs. the magnitude of the task and the length of time ACTUALLY required?

12. Can you compare or contrast your way of doing business with the Government with practices in the U.S. commercial or International sectors?

* *

INTERVIEW/DISCUSSION POINTS

* What gives you the most "heartburn" In dealing with NASA?

* Are documentation requirements:
    Excessive?
    Conflicting?
    Duplicative?
    Restricting Innovation?

* What can you say about procurement policies/regulations?

* How is the Interface with your customer(s)?

* Is your test program:
    About right?
    Duplicative as hardware progresses towards launch?
    Still addressing obsolete requirements?
    A great burden to your program?

* Are there any personnel/human resources policies/practices that are causing you difficulties?

FIRST LUNAR OUTPOST CONSTRUCTION ANALYSIS

Team:

Chris Grasso (EE)
John Happel (CE)
Brent Helleckson (Aero)
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Jane Pavlich (EE)
Dr. Renjeng Su (Prof., EE)
Rob Taylor (Aero)

N92-33325

CONSTRUCTION PROBLEM: RADIATION SHIELDING

❑ **Solar Flares:**
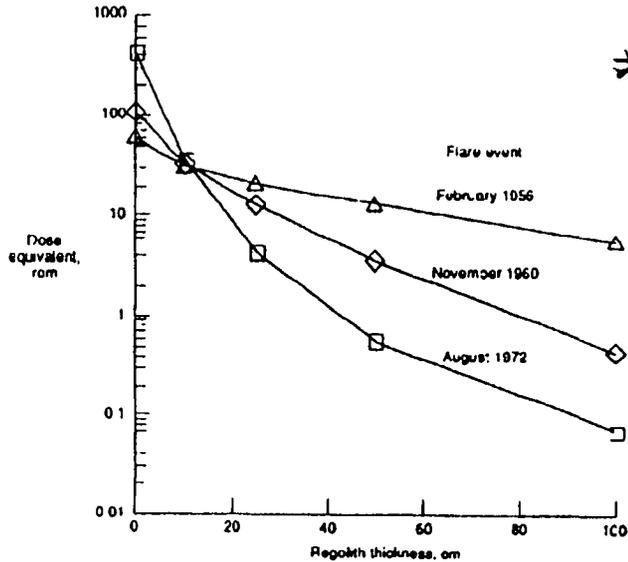
* From NASA
Technical Paper
2869
(Dec 1988)
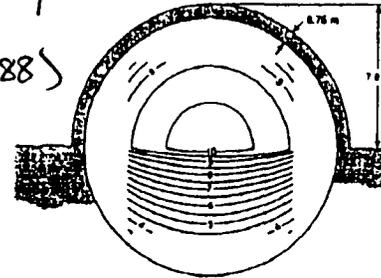
Pig 14 - Annual BFO (5-cm depth) dose-equivalent
variation within a half-buried sphere shielded overhead
with 75-cm regolith layer (Dose-equivalent values are in
rem/yr with contour intervals of 0.5 rem/yr.)

Figure 8 Predicted 5-cm-tissue dose equivalents for skin thicknesses between 0 and 100 cm in simulated lunar
regolith

❑ **But What About GCR?**

**Some Studies Indicate That Approx. 10cm of Aluminum is Required
(for a 4.4m x 12.2m habitat this would be approx. 56 MT in shielding)**

❑ **How About Long Durations (GCR & Solar Flare)?**

Some Studies Suggest $\dfrac{*785 \text{ g}/ \text{ cm}^2}{1.66 \text{ g}/ \text{ cm}^3} = 473 \text{ cm of Regolith}$

PRELIMINARY CONSTRUCTION ANALYSIS
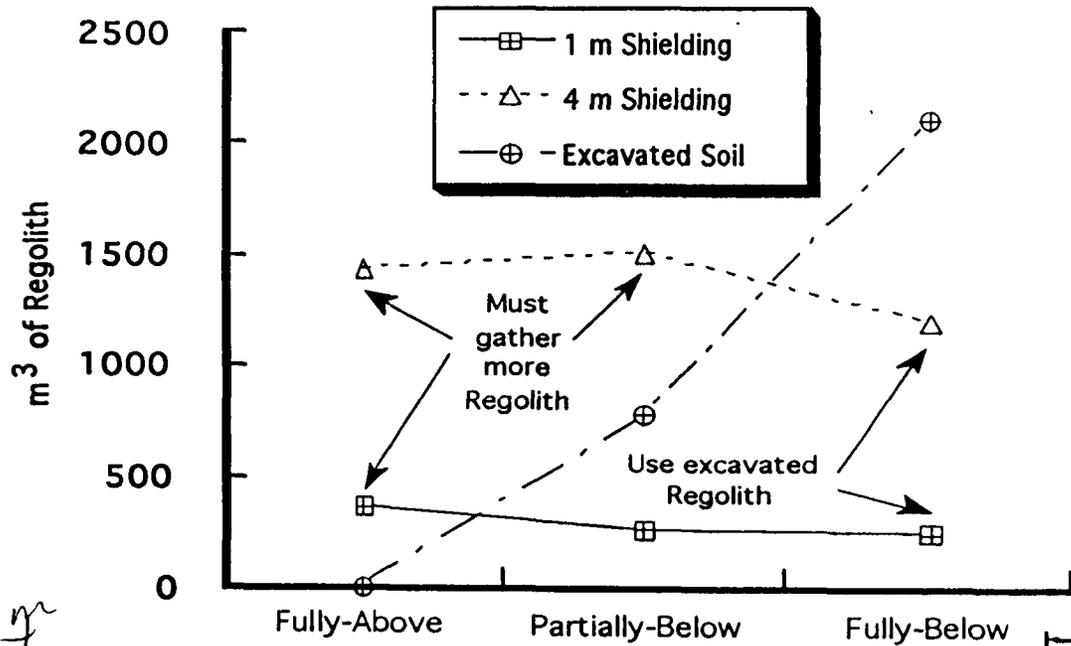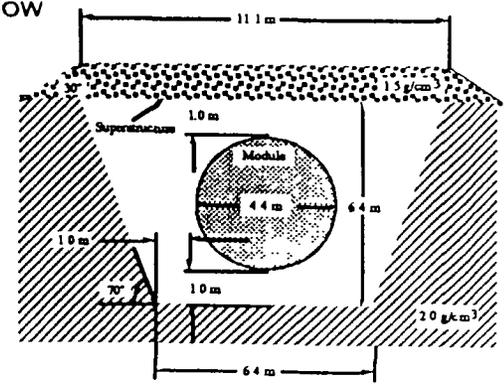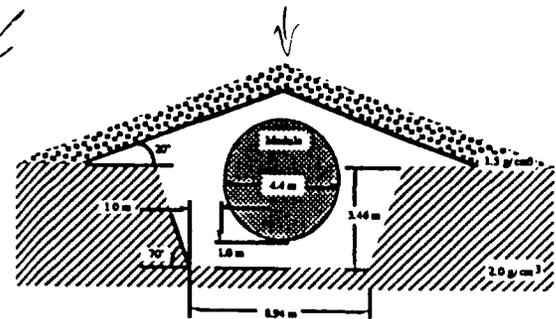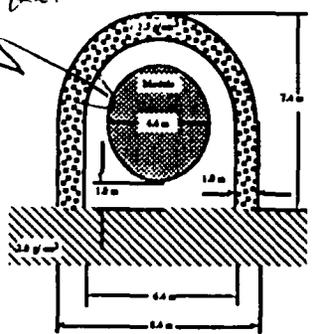
ALTERNATIVE REGOLITH SHIELDING CONCEPTS
(USING SUPERSTRUCTURES WITH 1m CLEARANCE)

A NASA Space Engineering Research Center at the University of Colorado

3 SDJ_5-13-92

A FIGURE-OF-MERIT: CUMULATIVE CONSTRUCTION KM

Legend:
- ▦ Fully-Above
- ▨ Partially-Below
- ■ Fully-Below

Chart y-axis: km (0 to 1200)

x-axis categories:
- 1 Meter of Shielding
- 4 Meters of Shielding

(assuming .25 m^3 collection per cycle)

150m avg trip

20° Ramp

Shielding

scrape

Dump

20° Ramp
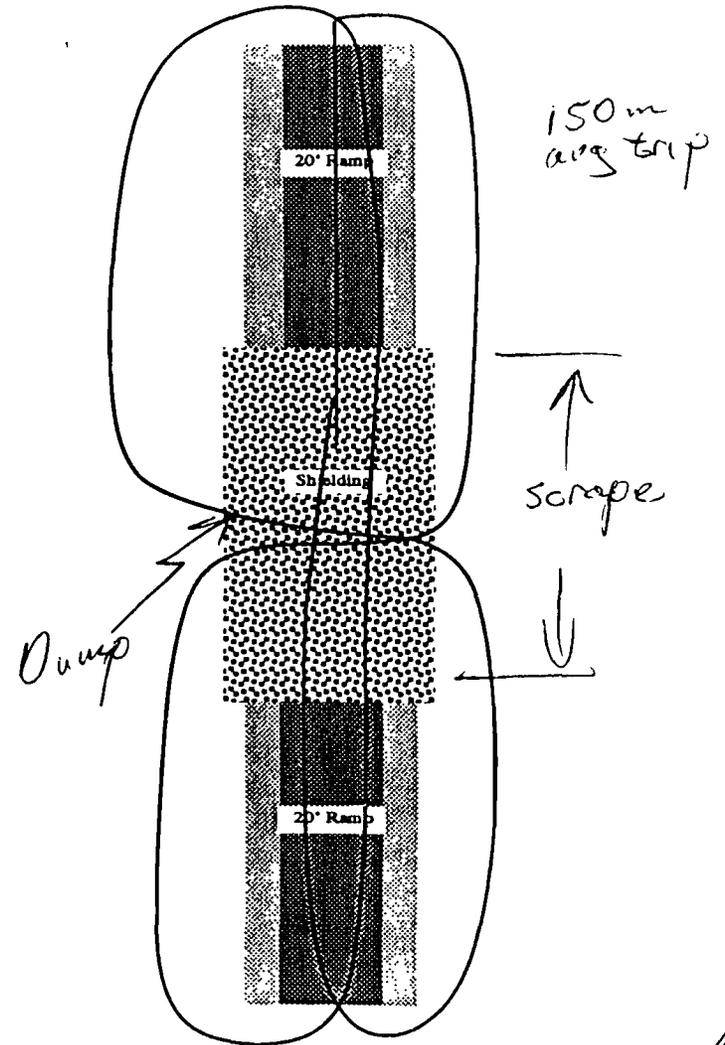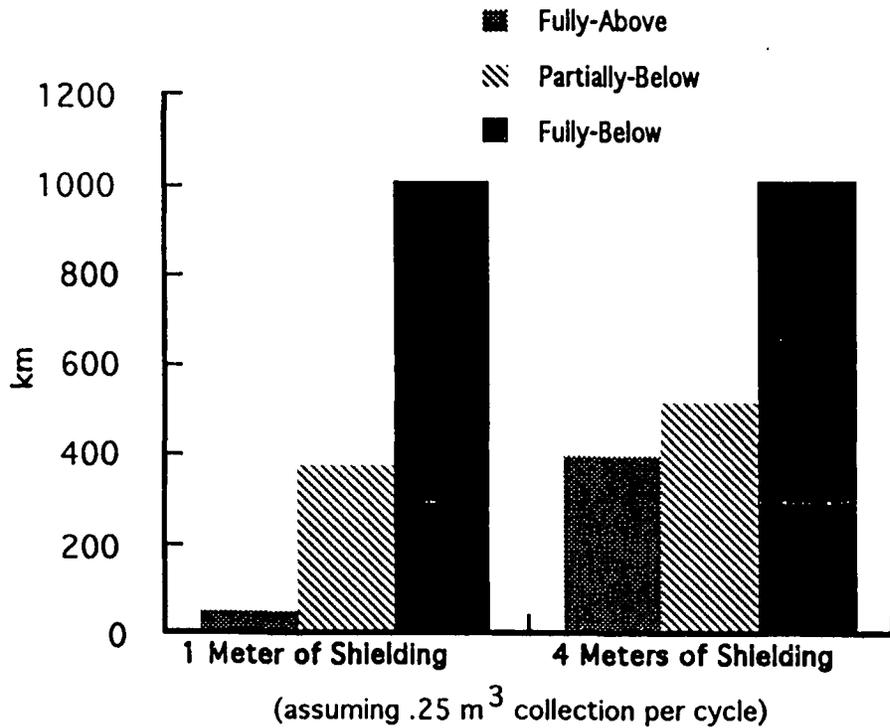
Feasibility Analysis: A SMALL LUNAR TRACTOR-SCRAPER (LTSV)
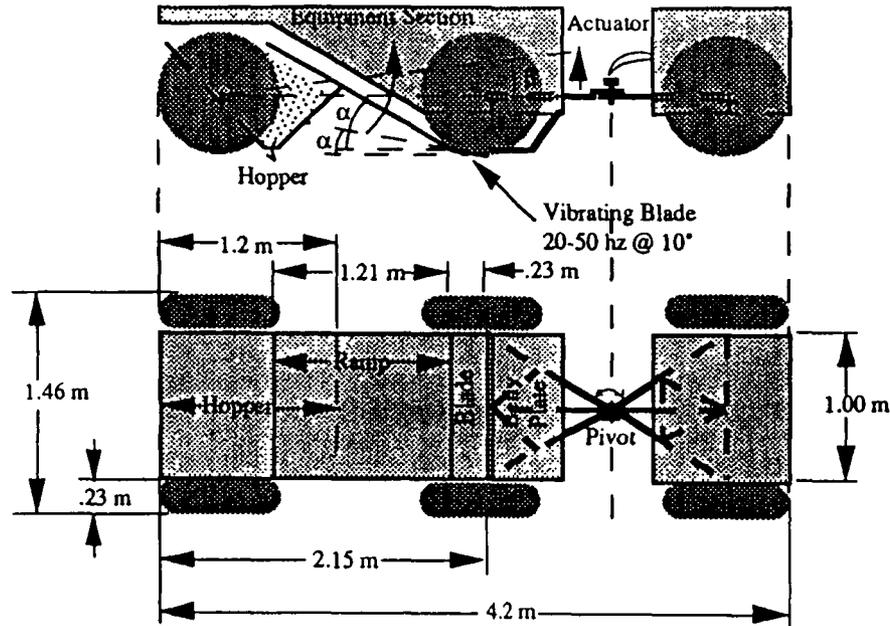
Gross Mass	1300 kg
Empty Mass	925 kg
Payload	.25 m**3
Velocity	.305 m/s
Time-To-Fill	58.97 s
Cutting Depth	1.4 cm
Apollo LRV Wheels	
Power:	275 kg NiH2
Duty:	12 hours

- **Estimate Power Requirements**
- **Perform Reliability Analysis**
- **Try Dragline Crane, Bulldozer, Front-End Loader**

A NASA Space Engineering Research Center at the University of Colorado   5 SDJ_5-13-92

SCAPER PRELIMINARY POWER ANALYSIS

LTSV Duty Cycle
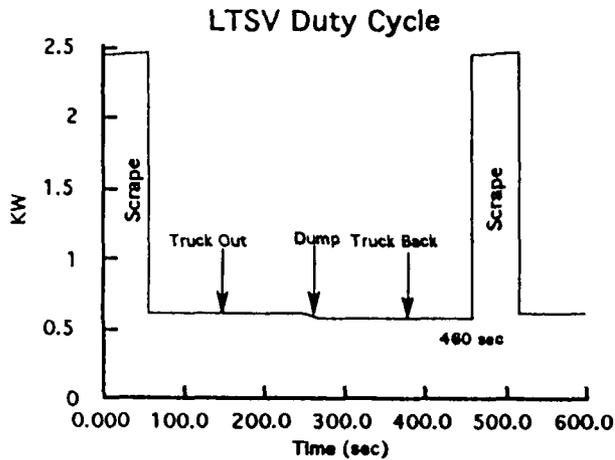
Peak Power (est.):

$$P_p = (P_{vibration} + P_{scrape} + P_{move} + P_{standby})\ FOS$$
$$= (.536\ kw +.082\ kw +.114\ kw +.500\ kw)\ 2$$
$$= 2.460\ kw$$

$$P_{scrape} = .067 + .015 = .082\ kw$$

Time-to-Fill $\cong$ 60 sec
@ $V = 1$ ft/sec

RAMPS:

$w = 1.64 \times 10^{-3}$ N

$P_m = 1.08 \times 10^{-3}$ N ($\alpha = 10°$)
$= 1.52 \times 10^{-3}$ N ($\alpha = 30°$)

$\alpha = 30°$

$\emptyset_k$ / 30°

Hopper 30° 10°
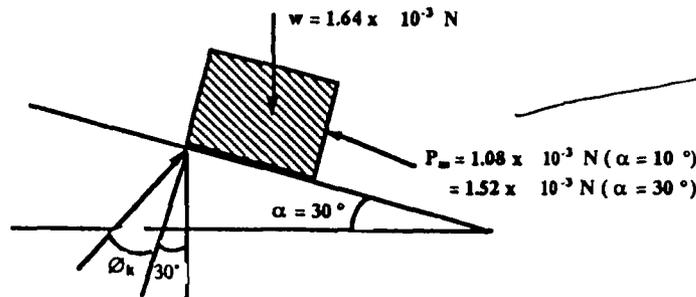
==> .015 kw
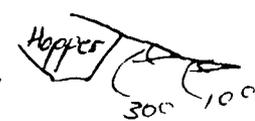
BLADE:

$$F = w(\frac{\rho g H^2}{2}\tan^2(\frac{\pi}{4} + \frac{\emptyset}{2}) + 2cH\tan(\frac{\pi}{4} + \frac{\emptyset}{2})) = 220\ N \quad ==> .067\ kw$$

$$(\text{vibration, } \emptyset = 35°)$$

$F \longrightarrow$ $\frac{\pi}{4} - \frac{\emptyset}{2}$ $H = 1.4$ cm

LTSV FEASIBILITY

Productivity of LSV R3

Fully-Below (handwritten)

- —— Cycles
- ----- km
- – ◇ –Energy (Mj)
- --✕-- Hours

.200 10^4
.000 10^4
8000
6000
4000
2000
0

4000
3500
3000
2500
2000
1500
1000
500
0

km, hours, Mj (vertical axis label)

0.000 500.0 1000 1500 2000 2500 3000
Excavation (cubic meters)

Comparison of Mass Ratio of Design (Lunar Vehicles)

2.5
2
1.5
1
0.5
0

Payload Mass (kg)/ Vehicle Mass (kg)

■ Payload/Vehicle

CSC's LSV   90-Day (unpress'd)   LOTRAN (Eagles)   Apollo

Conceptual (handwritten)

Existing (Real) (handwritten)

Fully-Below (handwritten)

1 Scraper ⇒ ~1100 hours

2 Scrapers ⇒ ~550 hours

3 Scrapers ⇒ ~370 hours
, (15 days Continuous)

☐ **Conservative Design**

☐ **Few Moving Parts =>**
High Reliability

☐ **Deterministic Unit Ops**
=> Semi-Autonomous

OTHER CONCEPTUAL POINT DESIGNS:

A SMALL LUNAR DRAGLINE CRANE (LDC)

Rover Mass   1300 kg
Crane Mass   1500 kg
Payload   .25 m°*3
Apollo LRV Wheels
3-Segment Frame

T Boom

Six-link
Suspension

Vibrating
Excavator

15 m

Counterweight
Bucket
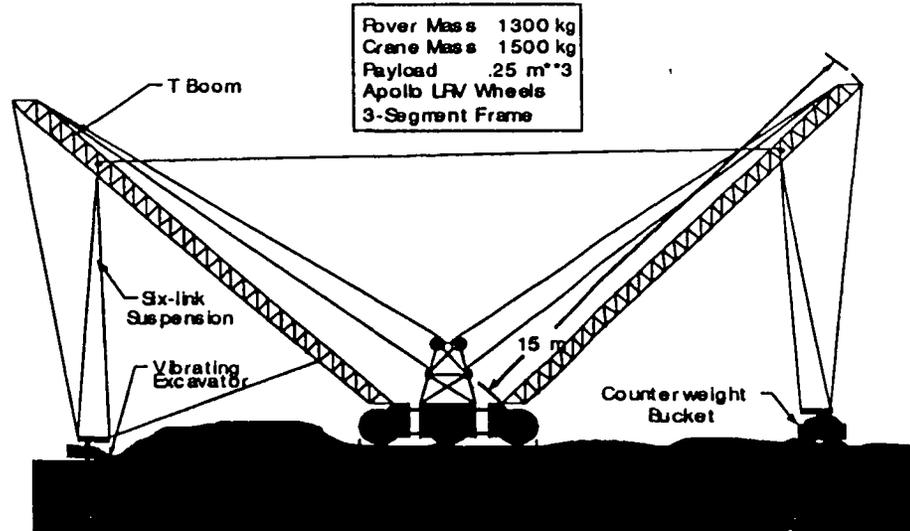
A LUNAR SUPERSTRUCTURE ARCH (LSA)

20° Ramps
(Both ends?)

20 m

6.4 m

Tied
Arch

Preliminary Mass Estimates

Carbon Composite w/ Aluminum Ties
≅ 1.7 mt

Aluminum w/ Aluminum Ties
≅ 2.8 mt

(Factor of Safety) = 2.5
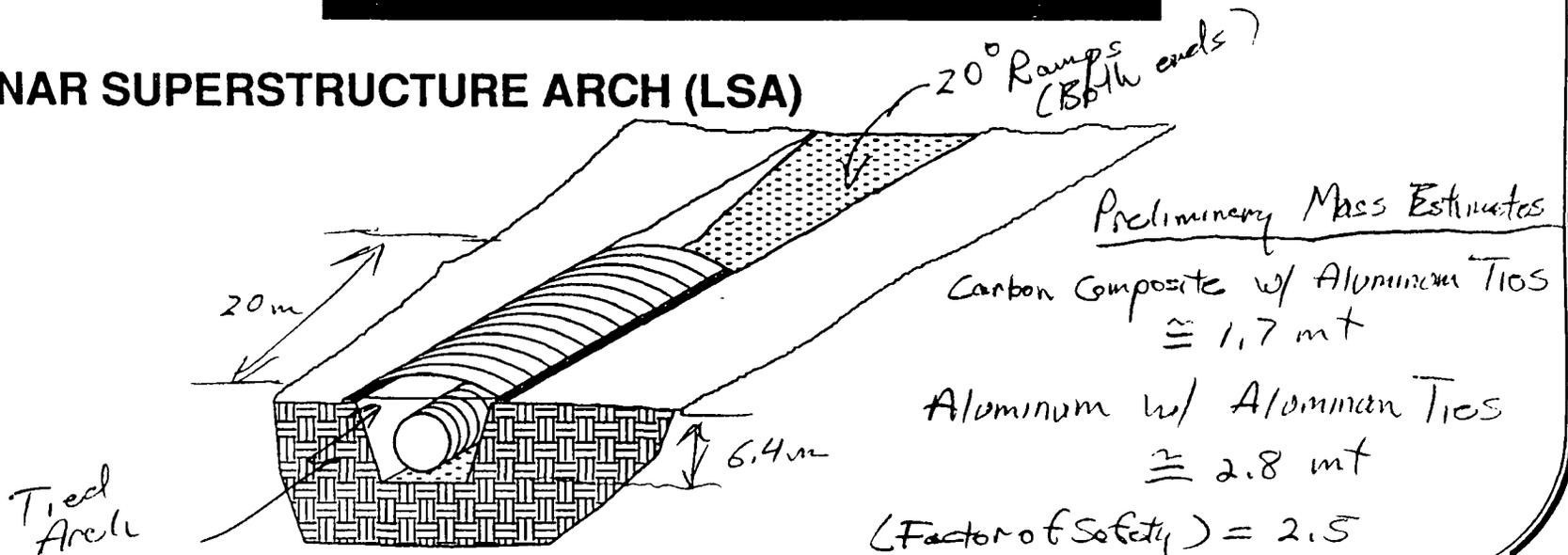
Feasibility Study: A LUNAR SELF-OFFLOADING LANDER CRANE

Lunar Lander-Crane Concept
Preparation to Lower Module

Module Lowering Cable

Boom Tie Cable

Boom Support Cables

Six Cable
Suspension

Extendable Arm

Regolith Auger
or Bucket
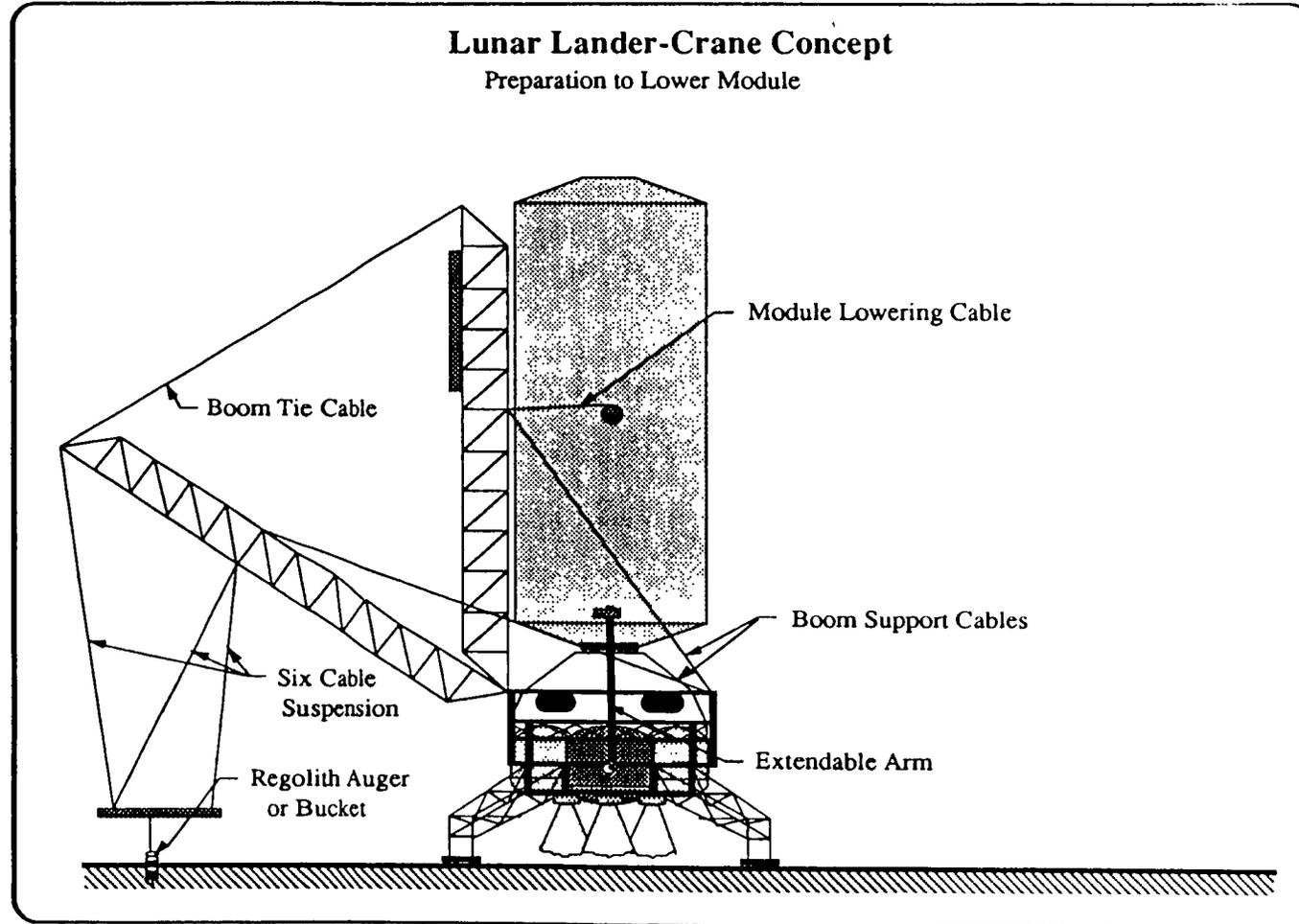
A SINGLE LAUNCH BASE POSSIBLE MANIFEST?

20 MT   SSF Derived Habitat Module with Airlock

06 MT   Lunar Excursion Vehicle (LEV) - Dry, e.g. 90-Day

01 MT (est) Lunar Self-Offloading Lander Crane

04 MT (est) 3 Lunar Tractor-Scraper Vehicles

02 MT (est) Fully-Below Superstructure Arch (composite)

---

33 MT Surface Delivery
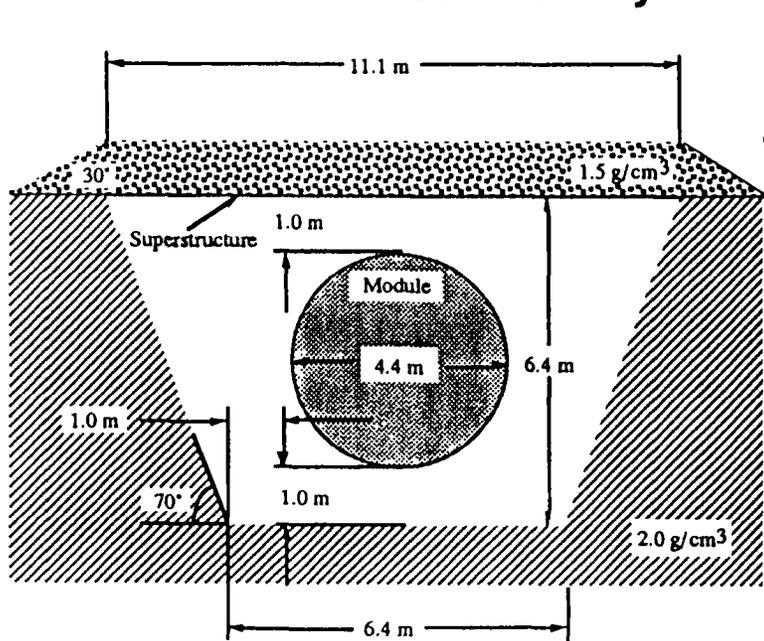
YIELDS ?

❑ **LLC Offloads Hab, Arch, LTSVs**

❑ **LTSVs Excavate Hole, Emplace Arch, Deposit Regolith to 4m**

❑ **LTSVs Drag Module on Skids into hole, umbilicals pay out**

11.1 m

30°   1.5 g/cm³

Superstructure   1.0 m

Module

4.4 m   6.4 m

1.0 m

70°   1.0 m

2.0 g/cm³

6.4 m

Dome   Cities   for   Extreme   Environments

Raymond S. Leonard[1] and Milton Schwartz[2]

Abstract

Extreme environments whether they be the frigid nights of the poles, the burning sands of the desert or the harsh environment of space pose interesting challenges to the Architect, the Engineer and the Constructor in their efforts to create habitats for mankind. Current or modern approaches seem almost primitive in that they seek minimums which confine and stifle the spirit rather than draw on technology and heritage to create environments for the human spirit which will allow it to grow. This paper is a discussion of the potential of separating some or all of the environmental protection functions from the structures providing privacy. The result is a graded environment for human habitation.

On Earth the major issue is thermal management. For arctic like environments the issue is to minimize the heat loss while providing an environment that not only maintains or merely sustains life but enriches. In the desert regions of the world the issue to minimize heat gain while preserving water. In both cases the goal is to create an oasis for human life. On Earth large domes offer a different approach. Large domes allow us to provide a buffer between the offices, living areas and recreational areas of small communities and the extreme environments in which they are located.

In space the goals are to protect from radiation while providing an aesthetic living environment for long duration missions. The need to provide both radiation protection and options for expansion of base facilities led the authors to create an unique structural system which separates the radiation protection systems from the pressure envelope of the habitats. The system uses cable networks in a tensioned structural system, which supports the lunar regolith used for shielding above the facilities. The system is modular, easily expandable and simple to construct. Additional innovations include the use of rock melting perpetrators for piles and anchoring deadmen and various sized craters to provide side shielding. The reflective properties of the fabric used in the membrane is utilized to provide diffuse illumination. The use of craters along with the suspended shielding allows dome to be utilized in fashions similar to those proposed by various designers unaware of the Moon's hostile radiation environment.

Additional topics addressed deal with construction techniques for large domes, i.e. on the order of 100's to 1000's of meters, thermal control, the integration of tertiary water treatment schemes with architectural design, human factors and its implications for the design of habitats for long term use in extreme environments

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A-2

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Engineers and Builders of the Future

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Dome   Cities   for   Extreme   Environments

Raymond S. Leonard, PE[1] and Milton Schwartz, AIA, PE[2]

> Note that the concepts of using tensioned structures for creating a suspensed radiation shield and the use of rock melting technology and microwaving regolith into structural columns are consider innovative and have been disclosed to Los Alamos National Laboratory per consulting agreements for evaluation as to patentability.

Introduction

Extreme environments whether they be the frigid nights of the poles, the burning sands of the desert or the harsh environment of space pose interesting challenges to the Architect, the Engineer and the Constructor in their efforts to create habitats for mankind.   On Earth the major issue is thermal management. and the current approach is to confine mankind in boxes.   Proposals for habitats in space range from tubes and spheres to fanciful artist concepts of large domes.

The challenge is to create or construct habitable environments which speak to the human spirit as well as to the minimums needed for survival.   In the desert regions of the world the issue to minimize heat gain while preserving water.   In both cases the goal is to create an oasis for human life.   Malls, air bridges, and underground shopping areas in northern cities point to potential design solutions for Earth based habitats.   Construction technology and economics have to develop. hand in hand in order for domed habitats to become a cost effective solution as well as aesthetic solution.

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For long duration missions or assignments to other planets the minimum goals are to protect from radiation and provide a habitable environment. If only minimums are to be meet and there will be no permanent settlements then tubes and boxes are the most cost effective. The Navy has shown that it is possible to keep over a hundred and thirty men working effectively in a confined and isolated tube, i.e. submarine, for up to ninety days. On the other hand experience in the arctic oil fields has shown that for assignments which approach being permanent more space and amenities are needed.

For arctic commercial operations the current approach consists of building large modules in the lower 48, shipping them to North Slope Oil Fields and installing them. The environmental hazards are cold and wind. Each structure wears a covering of thick insulation and requires considerable energy to counter both the cold. An alternative approach would be to erect large domes with a double skin for both strength and thermal insulation. The dome creates a volume of still air which is slightly heated by heat escaping from the structures enclosed within the volume of the dome. The inner surface of the transparent panes are coated to reflect back the infrared radiation being radiated by the buildings. Within the dome there would be the normal structures and landscaping. Waste heat from refrigerators, cooking, and normal activities would be rejected to the dome's interior atmosphere. The advantages are the creation of open space where people can walk without fear of the arctic winter winds and the energy savings which comes from not having to heat the buildings inside the dome against the wind.

While domed cities in the arctic attempt to keep the heat in, the equivalent of the lunar night, the desert environment imposes the design requirement is to minimize thermal gain, the equivalent of the lunar day, and water loss. In this case the outer layer of transparent panels are coated to reflect both UV and IR radiation thus minimizing the heat gain. On the north side of the dome the shaded radiators reject the heat load from the human activities inside the dome. Part of the tertiary water treatment system would be to run the water through irrigation channels or artificial stream beds inside the dome. This would help humidify the air and provide a pleasing background. This approach would emulate traditional practices of people living in oasis in the Sahara.

In the Sahara water is collected from the aquifers by long tunnels known as foggaras. The collected water runs through irrigation channels to the date palms. Close to the source the people draw their drinking water and in the village they add sewage or nutrients to the stream. Although the water is put to multiple uses it is in the end still lost.

Biosphere 2 is an experiment to see if a natural closed loop system can be created. It is ambitious and may or may succeed. Domed villages in the desert don't have to be as ambitious. They can allow for air exchange and if they minimize water lost they have gained a significant economic advantage.

Tertiary water treatment developed by civil engineers and filtration systems being developed for space exploration will allow use to create gardens in the desert complete with water falls and gardens. In the case of the semi-closed environment of the dome the water would be re-cycled. Some treated and some captured from the air after it had been transpired by the plants inside the dome.

Ignoring the question of economics of large domes, which is not unreasonable given the reasoning city governments use to justify domed sports stadiums, the next major question is constructibility. The proposed approach is to use a series of tower cranes, which after construction form the cores of high rise buildings and communication towers which penetrate the skin of the dome. The towers can also be used to support helipads, viewing rooms and other functions. Figure 1 shows a spherical type dome. The type of dome chosen will depend in part on whether the desire is to cover acreage or to provide vertical space for aesthetic purposes.

Design Considerations for Terrestrial Domes

The large spans and the need for thermal management lead the
authors to a double skin or a layered approach.   For the desert
habitat the outer skin would be composed of a combination of
opaque panels of solar cells and transparent panels treated with
films to reject infrared radiation.   The secondary skin would be
composed of transparent panels coated with films to filter more of
the thermal radiation.   Consequently the amount of thermal gain or
lost experience by the atmosphere and structures inside the dome
would be controlled.

Design Considerations for Lunar Systems

Having looked at two extreme environments on Earth we
considered the problem of creating livable space on the Moon.   The
basic or major factors in the design environment are: vacuum,
thermal cycling and radiation.   The basic concepts, which have been
suggested for lunar structures, are shown in the next set of figures.
They range from tubes on the ground to sand bagged spheres.

Our first thought was to incorporate the required radiation
protection into the envelope of each structure within the dome.   In
this case the residents would have windows three feet think for
radiation shielding and could look out on gardens and open spaces.
In some cases they might even be able to walk around outside if the
solar radiation levels were low.

We considered putting the necessary shielding on the dome itself
but this greatly increased both the strength needed in the structural
system and the construction complexity.   Constructibility is a
concept often overlooked by many designers who extrapolate
designs that are workable in a highly industrialized society to either
developing countries or the lunar surface.

At this point one of the authors, Milton Schwartz, combined our
layered approach with both cable roofing systems and the use of
small craters.   The resulting concept is one where a cable network
roofing system is used to create a suspend radiation ~ shield.
Underneath the shield the construction camp can gradually be
expanded until the construction expertise and resources are
available to enclose the individual habitats in a dome.

Ad Astra, Ltd.   **Milton Schwartz Architect**

The system is adaptable to both the lunar plain and to covering craters. In the latter case the crater walls provide the necessary side shielding. The approach is feasible because while weight changes with gravity the strength of materials doesn't. In addition we don't have wind loads to worry about on the moon. This sort of structure, tensioned cables, has been used around the world in applications as varied as sports arenas and in Saudi Arabia.

Tension   Structures

Tensioned structures are load adaptive in that the structural
system changes shape to accommodate changes in load rather than
increase stress levels (Leonard, 1988).   They transmit their loads to
the support system though tensile stresses.   Quoting further from
Leonard, 1988, tension systems can be comprised of membranes,
cables or combinations of both.

Tensioned structures can be grouped into two broad categories:
uniaxially stressed cable systems and biaxially stressed systems such
as membranes and nets.   Suspension bridges such as the Golden
Gate Bridge are an example of the first type.

There are four major categories of cable structures, Leonard,
1988:

   1. Single cable systems such as guy lines for towers or
      tents.
   2. Cable trusses, e.g. cable stayed bridges and double layer
      cable supported roofs.
   3. Cable nets which are multiply connected in a curved
      surface and loaded normal to the surface, e.g. hanging
      roofs
   4. Cable networks forming a three-dimensional framework.

There are also four major categories of membrane structures,
Leonard, 1988:

   1. Air-supported structures where an enclosing membrane
      is supported by a small differential pressure..
   2. Inflated structures which use highly pressurized tubes or
      dual walled mats as structural members.
   3. Prestressed membranes where fabric is stretched over
      rigid frameworks to form diaphragms such as tents,
      masted roofs.
   4. Hybird systems in which membrane panels span between
      primary load carrying members such as prestressed
      cables.

Ad Astra, Ltd.   Milton Schwartz Architect

Why consider tension structures? The following reasons where given the 1979 ASCE special publication on Air-Supported Structures:

1. They are lightweight, collapsible and easy to transport.
2. They can be prefabricated in a factory.
3. They have low installation costs (low labor component)
4. The environmental loads are carried by direct stress without bending.
5. For air supported structures the primary load carrying mechanism is the habitable enviroment itself..

Hybird tension structures could be used as initial shelters in hostile environments, Leonard, 1988. A double wall system could first be inflated with air and later the wall foamed with a hydrated boron compound which would absorb the secondary thermal neutrons produced from the collision, i.e. stopping, of high energy particles with the primary radiation shield.

Dead Loads and Radiation

The radiation environment drives the loading condition.
Discussions with radiation transport specialists (MacFarlane, 1991)
indicated the need for a shielding thickness of between two and
three meters.   One meter is often sufficient to stop the primary
radiation.   However the stopping of the primary radiation results in
the production of secondary neutrons which require the extra
thickness in order to attenuate them.   Otherwise the thin shield
actually becomes a radiation generator radiating neutrons at an
energy level which is fairly harmful to humans.

Discussions with George Augenpaugh of SST-8, Space Physics,
indicates that an astronaut on the surface of the Moon would be in a
radiation field equivalent to that of free space due to the production
of secondary neutrons and their refelction or bouncing upward
from the luanr surface.   These findings are preliminary and based on
3-D radiation transport codes as opposed to the standard one
dimensional codes used in most studies to date.   Once the
calculations are checked a peer reviewed publication will be issued
which can be cited.

Ad Astra, Ltd.   **Milton   Schwartz   Architect**

Description   of   Concept

The basic system is simply a network of cables with a membrane laid over the interconnected cables.   If the system is to be erected on a plain a series of pylons have to placed along with deadmen for anchoring the cable ends..   If the system is to br placed over a crater which is deep enough then only the anchors for the cable ends have to be placed.   Depending on the spans a three-dimensional system may be needed.   The tension structural system proposed can take on a number of different shapes.   The system can be rectangular and expandable or it can be designed to take advantage of a small crater. The advantages besides those listed above are:

1. An attenuated radiation field under the structure, which can be handled by light weight, high hydrogen content insulating materials.
2. Controlled thermal environment, i.e. a constant, known environment as opposed to the large thermal swings between the lunar day and night.
3. Large open spaces for base expansion without the need to move shielding material around whenever an expansion or facility repair is needed.

Materials   of   Construction

The   basic   materials   used   are   cables,   membranes,   columns   or   poles   and   lunar   regolith.

Weight   considerations   rule   out   steel   cables.   That   leaves   kevlar   and   some   sort   of   fiberglass.   For   terrestrial   applications   there   are   many   other   choices.   One   long   range   possibility   is   to   fabricate   ultra   high   strength   glass   fibers   from   lunar   resources.   Jim   Blacic   of   Los   Alamos   has   been   working   on   producing   and   testing   glass   fibers   in   an   anhydrous   environment.   Steve   Howe,   also   of   Los   Alamos,   suggested   coating   the   fibers   with   titanium   obtained   from   processing   lunar   soils.

Membrane   materials   will   probably   be   limited   to   kelvar,   fiberglass   or   beta   cloth.   The   fabric   should   have   a   fairly   high   resistance   to   accidental   damage.   It   should   be   easily   seamed   or   jointed.

For   the   pylons   and   deadmen   we   propose   to   adopt   rock   melting   technology   developed   at   Los   Alamos.   This   is   the   expensive   part   of   the   system   since   because   you   leave   the   tip   in   place   it   is   a   consumable.   Most   of   the   pylon   would   be   aluminum   with   only   the   tip   made   out   of   more   exotic   materials.   An   alternative   would   be   to   use   a   thin   wall   fiberglass   cylinder   which   after   it   was   placed   could   be   filled   with   regolith.   The   regolith   would   be   sintered   into   a   solid   mass   using   a   variable   frequency   microwave   generator.   This   would   minimize   the   amount   of   mass   that   had   to   be   landed.

The   tip   of   the   pile   (pylon)   would   contain   a   melting   system   which   could   be   either   electrically   driven   or   heated   through   the   use   of   heat   pipes   driven   by   concentrated   sunlight.

LUNAR CONCEPTS: CABLE-MEMBRANE TECHNOLO[G]

BERM-LATERAL SHIELD

° OFFSET SHIELDED
   ACCESS

INTERMEDIATE CABLE
MEMBRANE REINFORCING
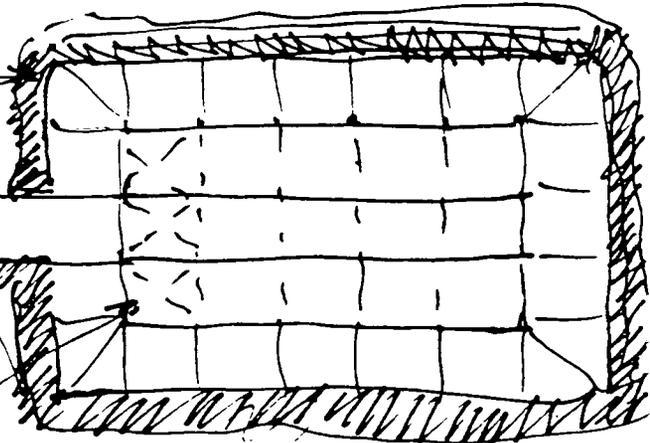
← 20 TO 50 METERS —

MODULAR-EXPANDAB[LE]
OVERHEAD SHIELD

BETA CLOTH COVER TO MITIGATE "BLAST-EFFECT" TURBULENCE

3 METER REGOLITH [S]

OFFSET ENTRY
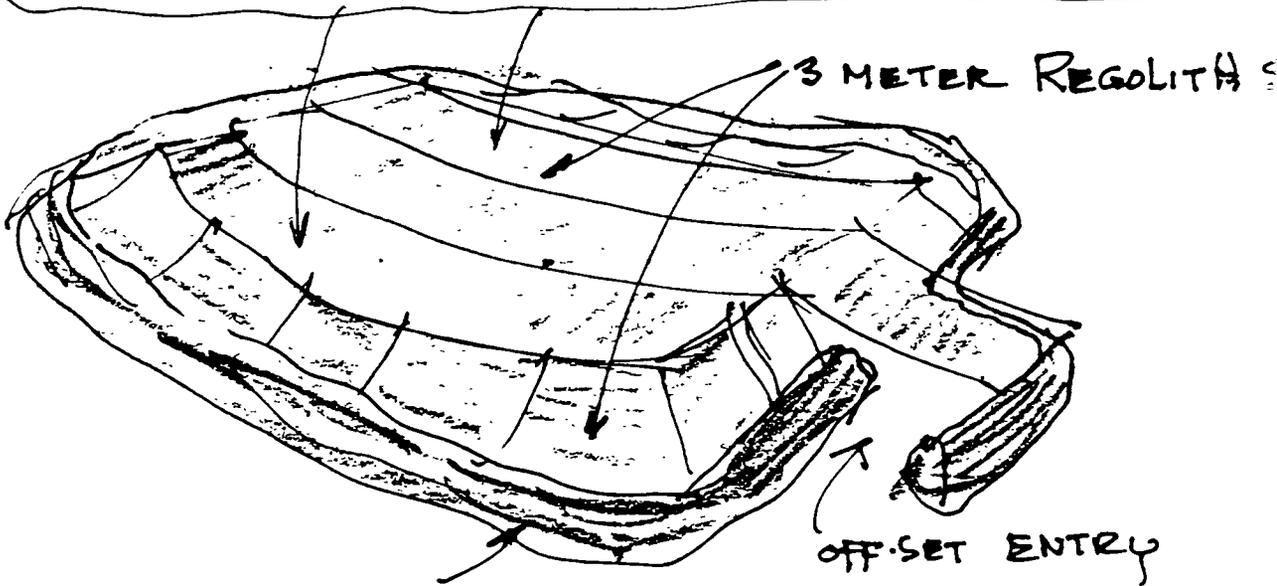
4-5 METER BERM
LATERAL SHIELD

LUNAR SURFACE STRUCTURE

LUNAR CONCEPTS: MILTON SCHWARTZ - AIA·PE
                   RAYMOND S. LEONARD - PE
          1   NOVEMBER 1991

LUNAR SURFACE STRUCTURE

3 METER REGOLITH SHIELD (BETA CLOTH COVER)

4.5 METER BERM

ADVANCED DOME HABITATS
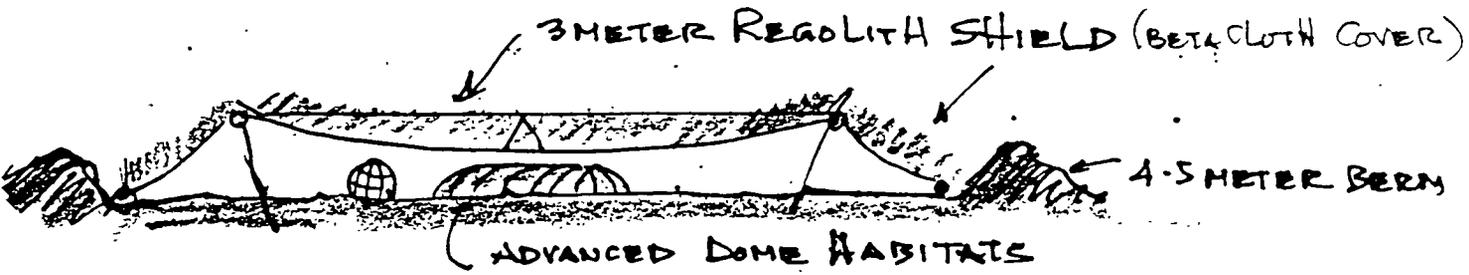
MODULAR - EXPANDABLE CABLE · MEMBRANE
RADIATION · SHIELD & MICRO-METEORITE SHIELD

CABLE ANCHOR

COLUMN SUPPORT

"ROCK · MELTER" HOLE
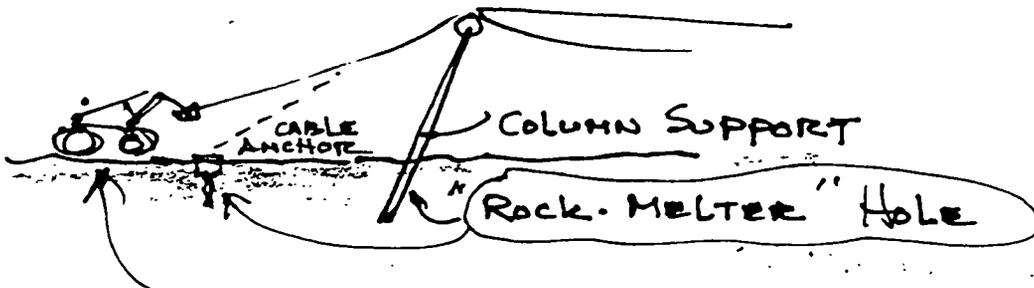
SURFACE - GRADER / LOADER CABLE - TENSION UNIT
INITIAL PHASE SHIELD STRUCTURES · HIGH ENERGY
POWER SOURCE

COLUMN

SOFT DUST LAYER

TRI-POD LEGS ROTATE TO BASE PLATE
LOCK POSITIONS

DENSE SOIL LAYER

BASE PLATE - BEARING
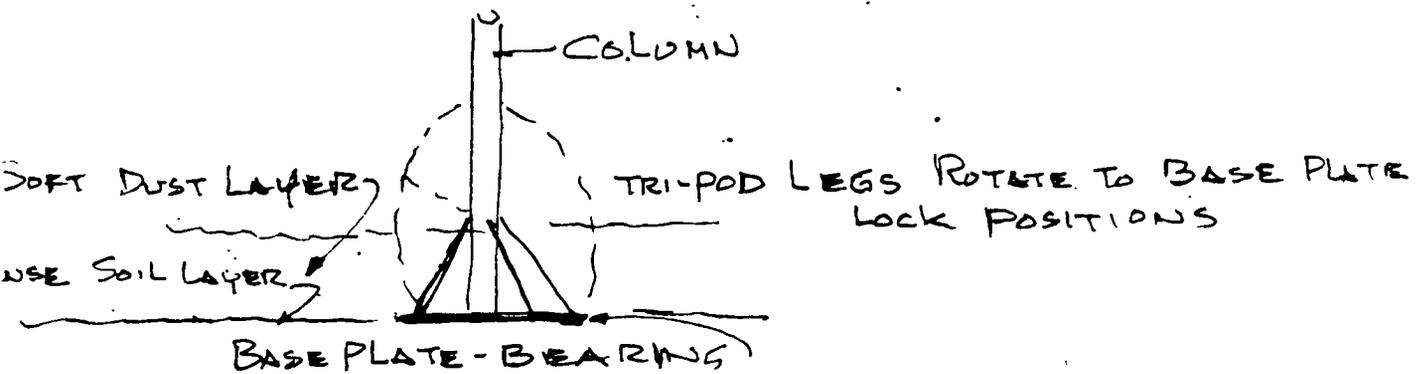
INITIAL PHASE SHIELD STRUCTURES - LOW ENERGY POWER SOURCE

LUNAR CONCEPTS: MILTON SCHWARTZ - AIA · PE
RAYMOND. S. LEONARD PE
NOVEMBER 1991

296
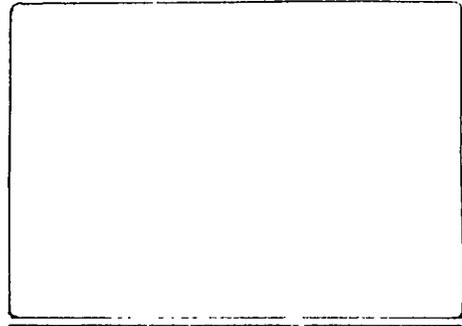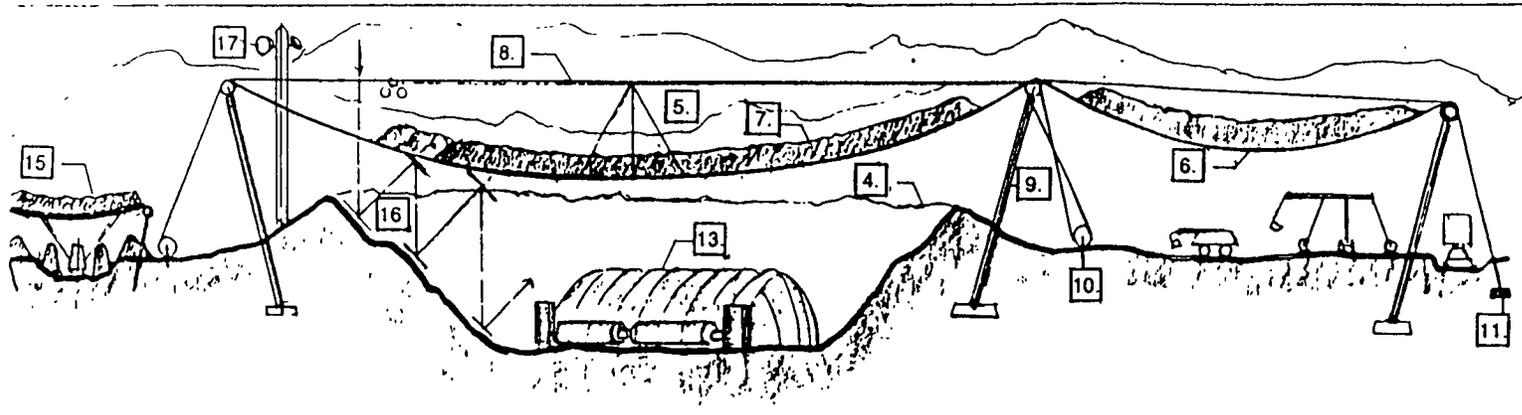
Cross Section | Scale: 1" = 20 Meters
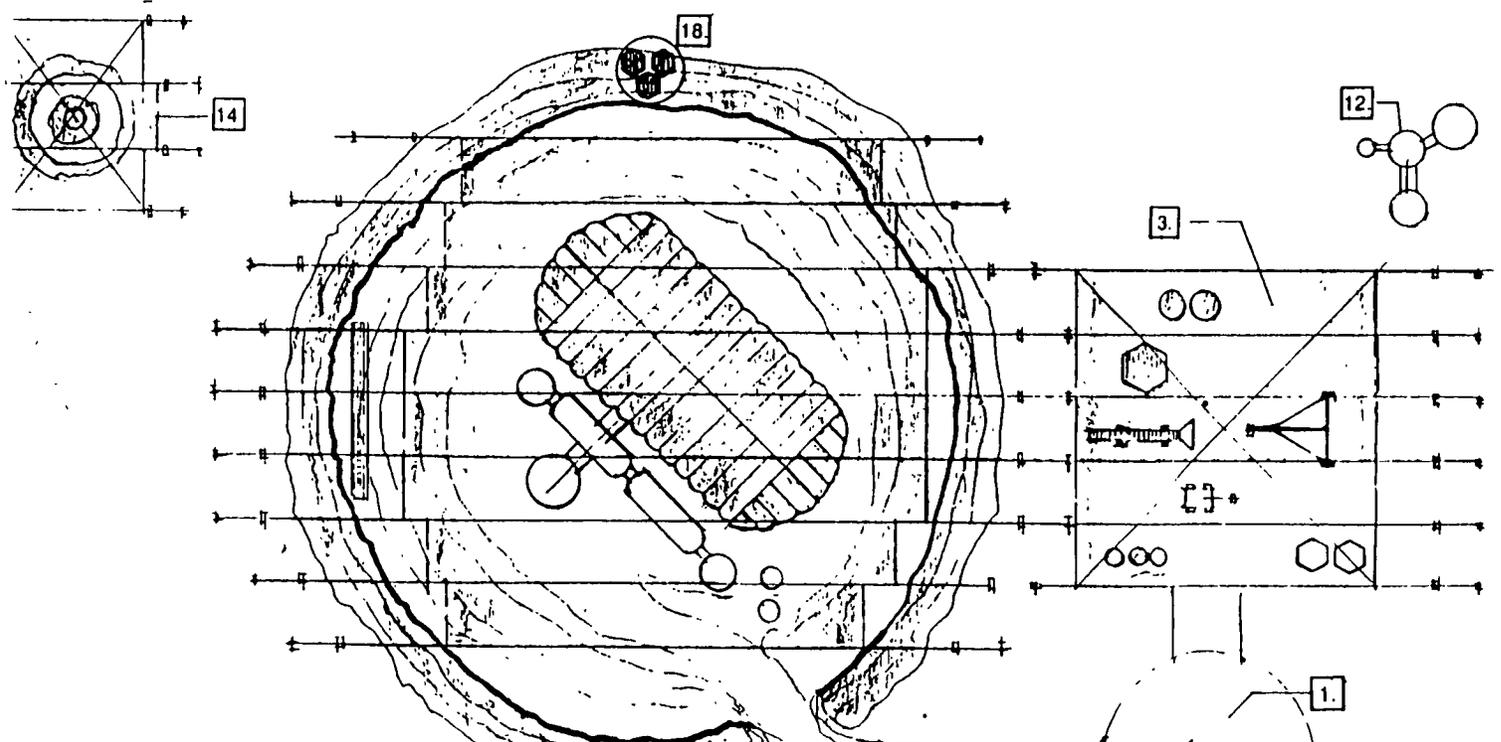
1. Landing Area
.2. Access Roadway
· 3. Level Surface Work Area
4. Crater Rim — Simple Crater
5. Caternary Cable Suspension System
6. Reinforced Beta Fabric Membrane
7. 3 Meter Lunar Regolith Radiation Shield
8. Top Cable
9. Perimeter Column and Pulley
10. Ground Winch
11. Ground Anchor
12. Stage One Habitat
13. Advanced Stage Home
 • Inflatable — Thermal Insulated
14. SP - 100 Nuclear Power Generator
15. SP - 100 Cable-Tent
 Neutron Radiation Shield
16 Helio Stat
17. Communication Tower
18. Solar Powered Regolith Fusers

LUNAR CONCEPTS
MILTON SCHWARTZ - AIA - PE
RAYMOND S. LEONARD - PE

13. Stage One Habitat

6. Perimeter Column - Pulley

7. Ground Winch

— — BERM

3. Catenary Cable Suspension System

4. Reinforced Beta Fabric Membrane

8. 3 Meter Lunar Regolith Radiation Shield

2. Access Roadway

Plan View

Scale: 1" = 20 Meters

LUNAR CONCEPTS
MILTON SCHWARTZ - AIA - PE
RAYMOND S. LEONARD - PE
OCTOBER 1991

10. Tension Ring
11. Communications
12. Helio Slat
5. Center Tower - Structural Column

10 Communications Tower

11. Helio Stat

9   Top Cable

7. 3 Meter Lunar Regolith Radiation Shield

Tension Ring

3. Catenary Cable Suspension System

4   Reinforced Beta Fabric Membrane

2. Perimeter Column Pulley

5   Center Tower - Structural Column

14 Advanced Stage Home
· Inflatable — Thermal Insulated
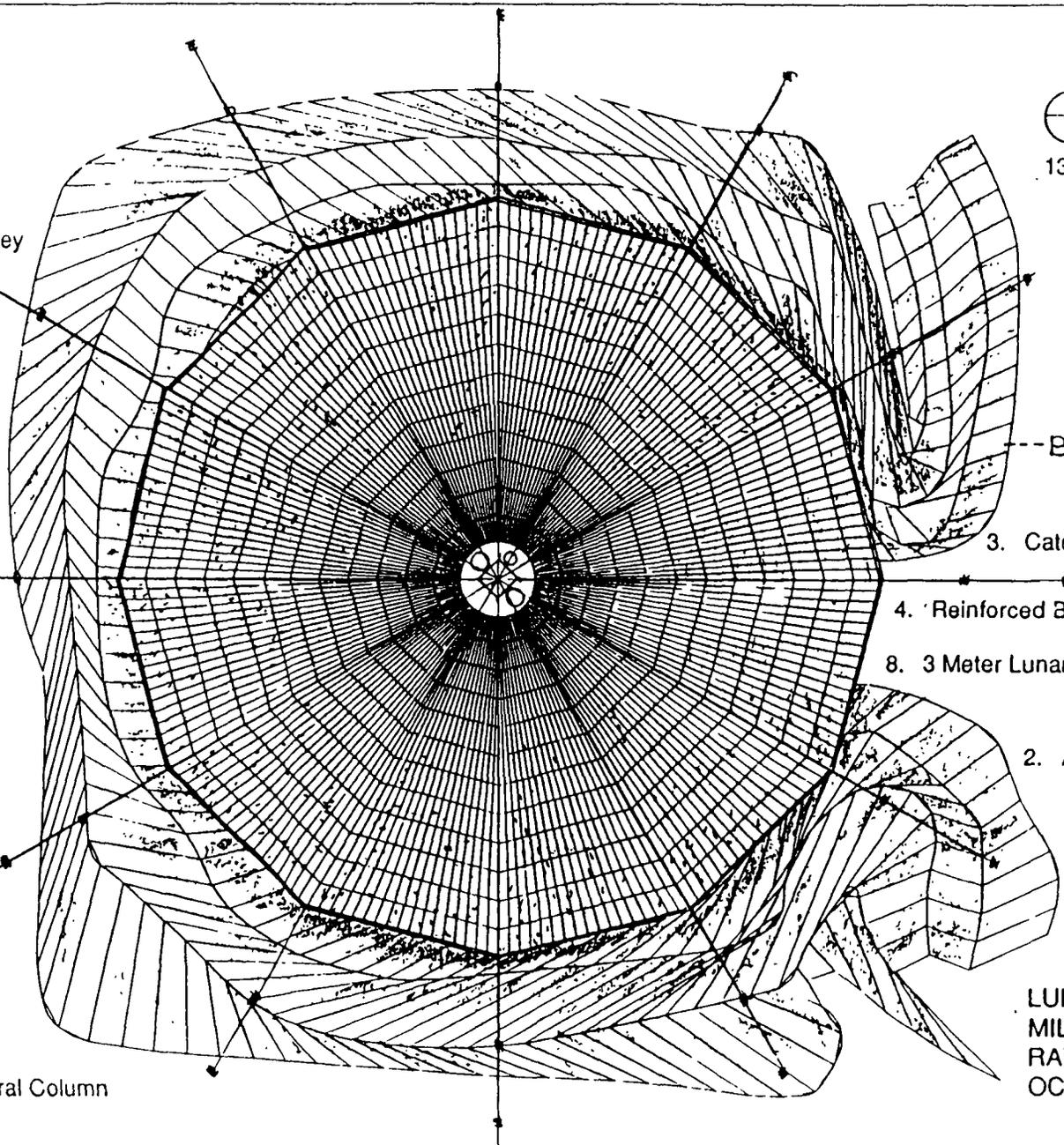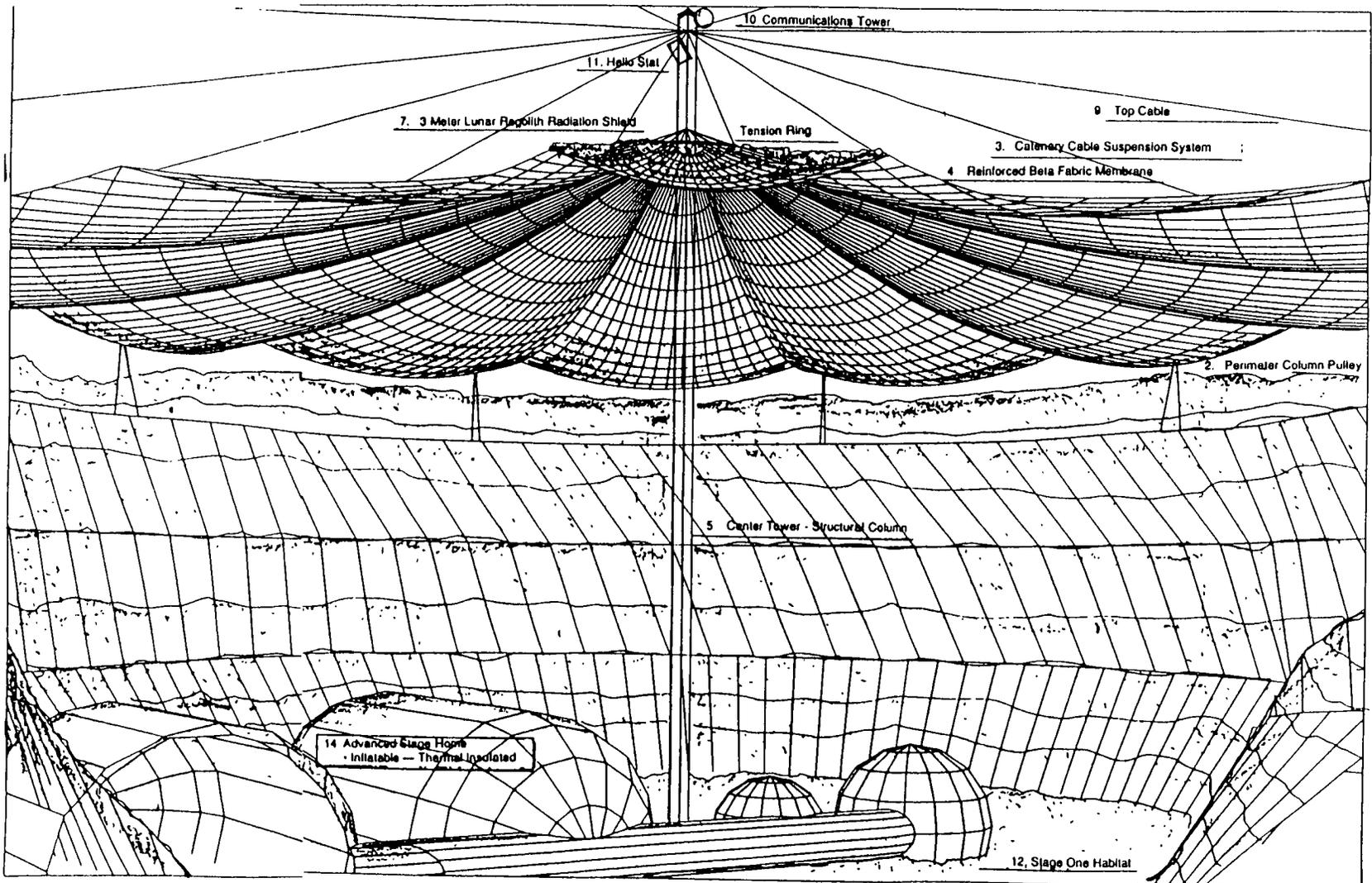
· Completed Assembly

·Cross Section

12. Stage One Habitat

LUNAR CONCEPTS
MILTON SCHWARTZ · AIA · PE
RAYMOND S. LEONARD · PE
OCTOBER 1991

LUNAR SURFACE STRUCTURE.

CABLE / MEMBRANE RADIATION SHIELD

VARIABLE ANGLE DISTRIBUTION HEAD ± 10 METER RANGE

TOP CABLE

END SUPPORT

EXTENDING CONVEYOR APPROX 70 METER RANGE

BATCH FEED 2 CM / SCREEN
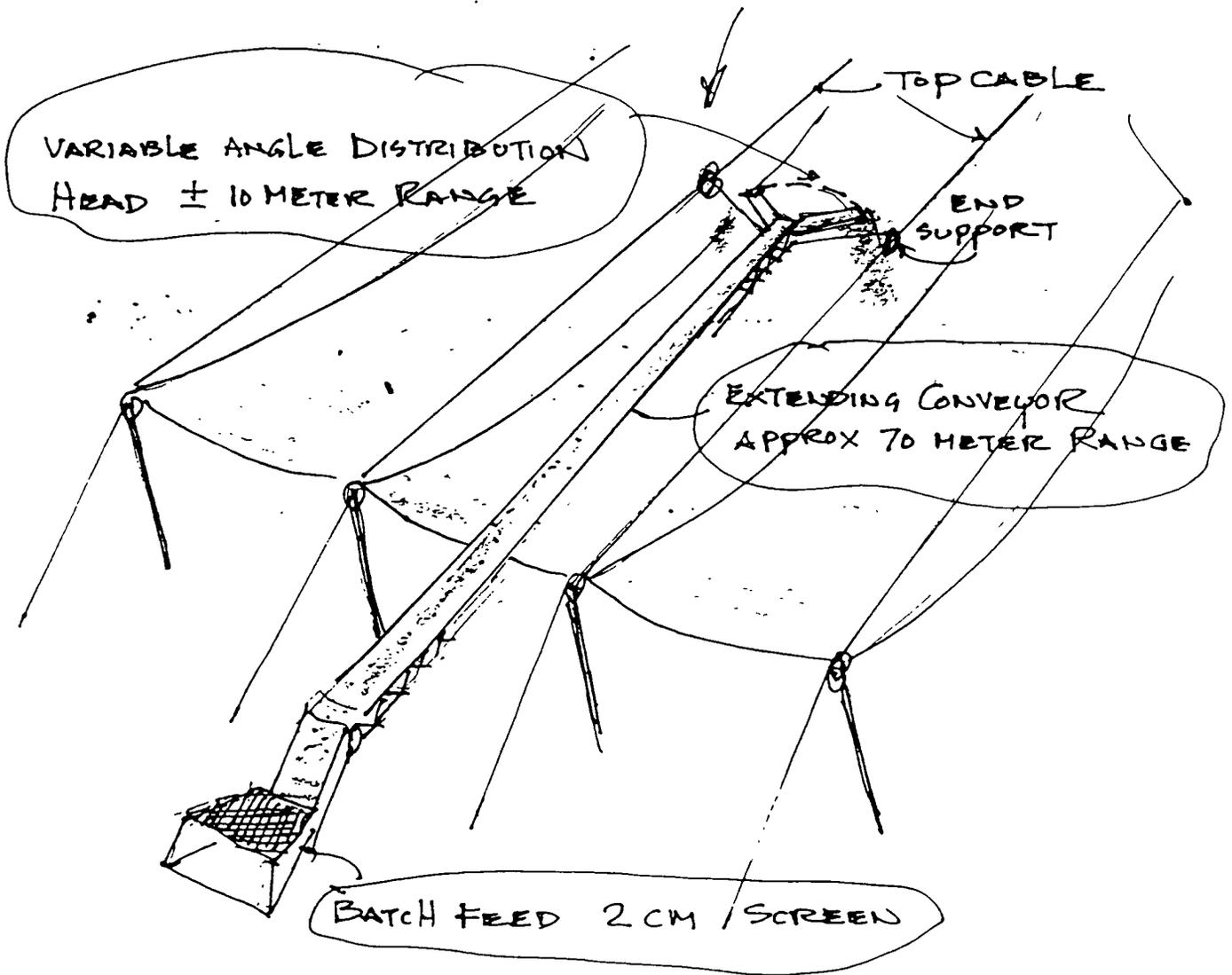
SOIL DEPOSIT CONVEYOR MOVES FROM SECTOR TO SECTOR

LUNAR CONCEPTS: MILTON SCHWARTZ · AIA PE
RAYMOND S. LEONARD PE

300

Site Preparation Studies - Dust Control - Solar Power - Robotic

FROM SOLAR CONCENTRATOR

REMOTE ROBOTIC SOLAR REGOLITH SOIL FUSER

16" DIAMETER
U. of C. DESIGN
(UNIV OF CHICAGO)
FERMI INSTITUTE

SAPHIRE CONE NON-OPTICAL LENS

FROM SOLAR COLLECTOR
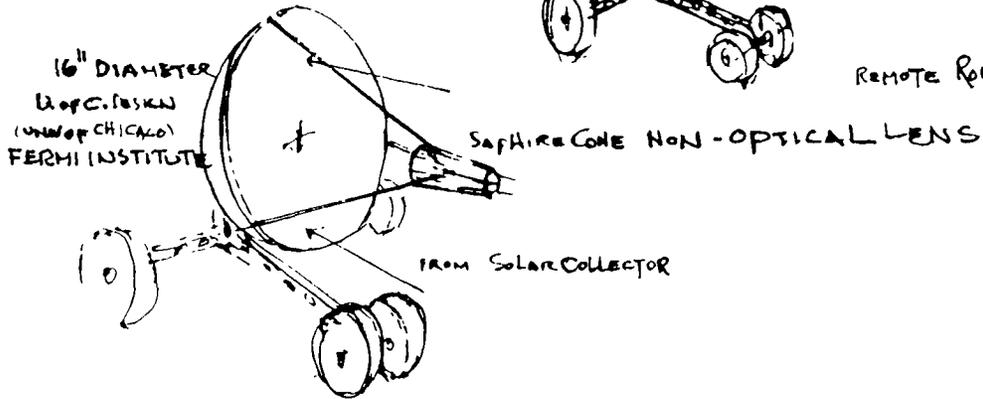
REMOTE ROBOTIC SOLAR CRYSTAL ROCK MELTER

SOLAR RAYS

McDONNELL DOUGLAS SPACE SYSTEM

APPROXIMATELY
18 MIRRORS AT
1 SQUARE METER EACH

LUNAR CONCEPTS: SOIL DUST CONTROL - ROCK MELTING UNITS
MILTON SCHWARTZ AIA·PE RAYMOND S. LEONARD - PE
OCTOBER 1991
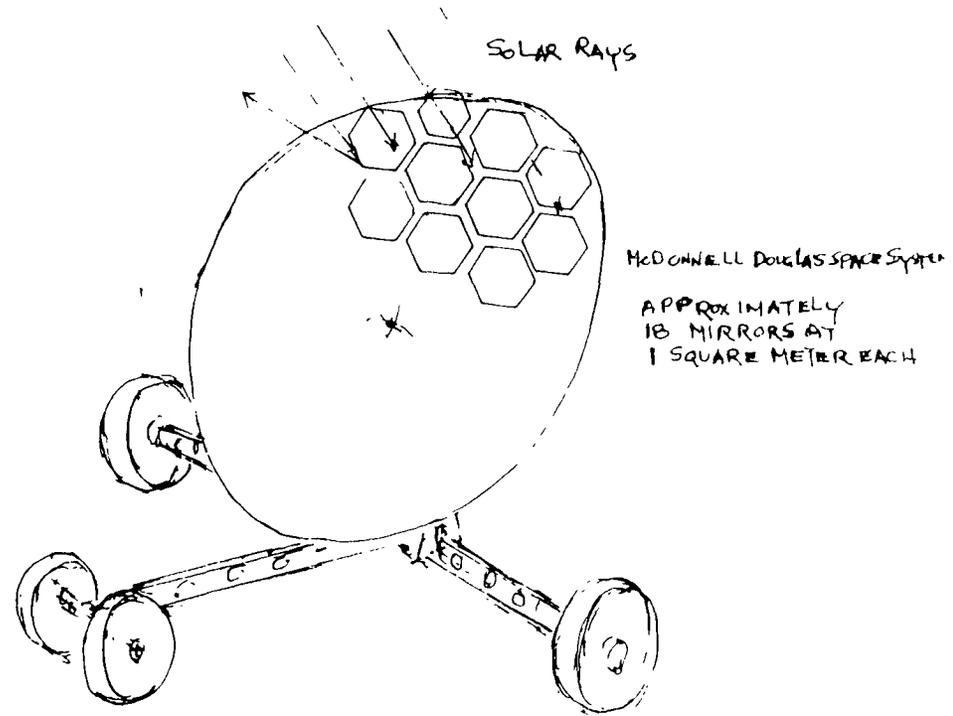
ROBOTIC SOLAR COLLECTOR - CONCENTRATOR

Rock   Melting   Technology

There are uncertainties associated with how well mechanical systems will hold up under the combination of heavy useage and hostile environment of lunar construction work.   In looking for alternatives we re-evaluated the work done at Los Alamos National Laboratory in the area of rock melting penetrators for geological work.   The next set of figures, taken from Los Alamos publications, describe some of the designs that were developed in the early 70's The work was conceptually scaled up for use in creating subway tunnels.

For our work we would look at modifying an extruding penetrator such that a solid core was left.   This would be similar to the coring with consolidation penetrator shown in the figures.   Our initial thoughts are to abandon the tip in place in order to simplify construction operations.   However a trade off study of weight vs complexity of operations needs to be made since the penetrators are very heavy.

For anchoring the cables we envision using the penetrators to melt a chamber or cavity into which the anchorage could be placed. Then using a solar concentrator lunar fines could be quickly melted and casted around the baseplates.   An alternative woud be to use sulfur concrete made for lunar resources.

A

CENTER FOR EXTRATERRESTRIAL ENGINEERING AND CONSTRUCTION

CETEC

GERALD G. LEIGH, PHD

New Mexico Engineering Research Institute
University of New Mexico
Albuquerque, New Mexico

As the Lead Agency for

The CETEC Development Team

A3

CENTER FOR EXTRATERRESTRIAL ENGINEERING AND CONSTRUCTION (CETEC)

The urge for humans to return to and explore beyond the moon appears to be increasing, both throughout our country and around the world. On 20 July 1989, the twentieth anniversary of man's first visit to the moon, President Bush announced the national Space Exploration Initiative (SEI), which calls for the construction of a manned orbital space station, the establishment of a lunar base of operations, and the manned exploration of Mars (by 2019). Japan has placed a satellite in orbit around the moon and declared its intention to be involved in efforts to colonize the moon. National defense has diminished as the primary national priority and the manned exploration of space is emerging as a new national imperative.

The establishment of an extended manned presence on the moon will require the development and application of many new enabling technologies that are not available today. Instead of predominantly aeronautical related technologies used in orbital space, great emphasis must be placed on the research, development, testing, and evaluation of technologies for construction, mining, and chemical processing of lunar materials to form habitats and provide resources for sustaining human life. As needed processes are identified and associated equipment and procedures are developed, it will be necessary to operate and test both equipment and procedures here on earth in a realistically simulated lunar environment. A large test facility where prototype equipment can be subjected to the harsh environments of vacuum, lunar soil, dust, extreme heating and cooling, and partial gravity is needed within the next five years, if currently projected schedules for space exploration are to be achieved.

A group of knowledgeable scientists and engineers in New Mexico has recognized the need for such a testing capability and has proposed a project to develop an extraterrestrial surface simulation facility. A group of universities, national laboratories, and private industrial firms is proposing to establish a Center for Extraterrestrial Engineering and Construction (CETEC) and to develop large extraterrestrial surface simulation facilities in which this needed testing can be realistically performed.

The Center for Extraterrestrial Engineering and Construction is envisioned to be both a center of knowledge and data regarding engineering, construction, mining, and material process operations on extraterrestrial bodies and a set of extraterrestrial surface simulation facilities. The primary CETEC facility is proposed to be a large domed building made of steel reinforced concrete with more than one acre of test floor area covered with several feet of simulated lunar soil and dust. The entire building would be pumped down to partial vacuum (10^{-4} to 10^{-6} Torr) to provide a realistic simulation environment. Extreme heating, cryogenic cooling, and partial gravity suspension systems would be included in the facility to further enhance simulation fidelity. Large steel cylindrical tanks could be placed inside the vacuum facility and repressurized to atmospheric pressure to simulate habitat, workshop, and

laboratory modules transported to the moon. Life support systems and elaborate safety procedures would be employed to permit researchers and test operations personnel to work safely inside the simulation facility. A lobby around the perimeter of the facility would permit tourists and other observers to look in through view-ports and observe test operations in the simulated extraterrestrial environment.

The development team proposing the CETEC includes the University of New Mexico (UNM), Los Alamos National Laboratory (LANL), the BDM Corporation, Ad Astra Corporation, and several large private engineering and construction firms. The CETEC is planned to be located in Albuquerque, New Mexico, on land provided by the University of New Mexico. The large simulation test facility is expected to cost between $35M and $50M to construct. The entire CETEC development project could be completed in five years, once funding is approved.

CETEC
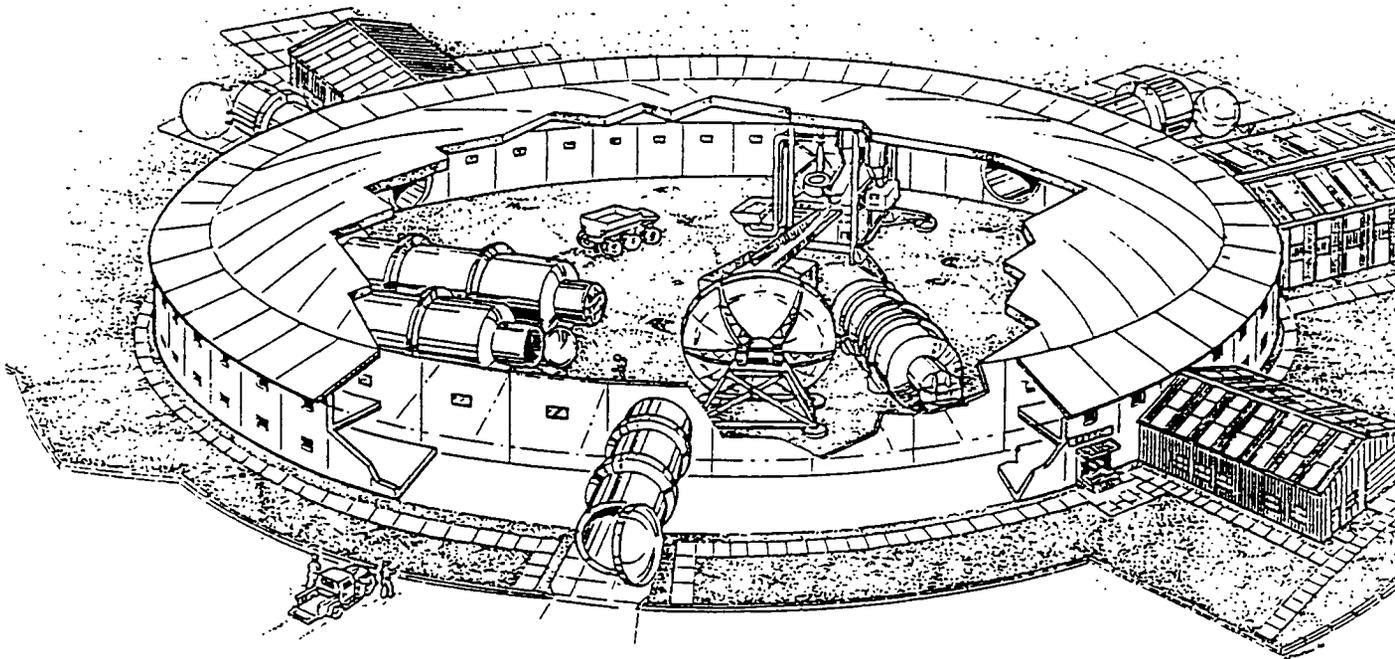
CETEC
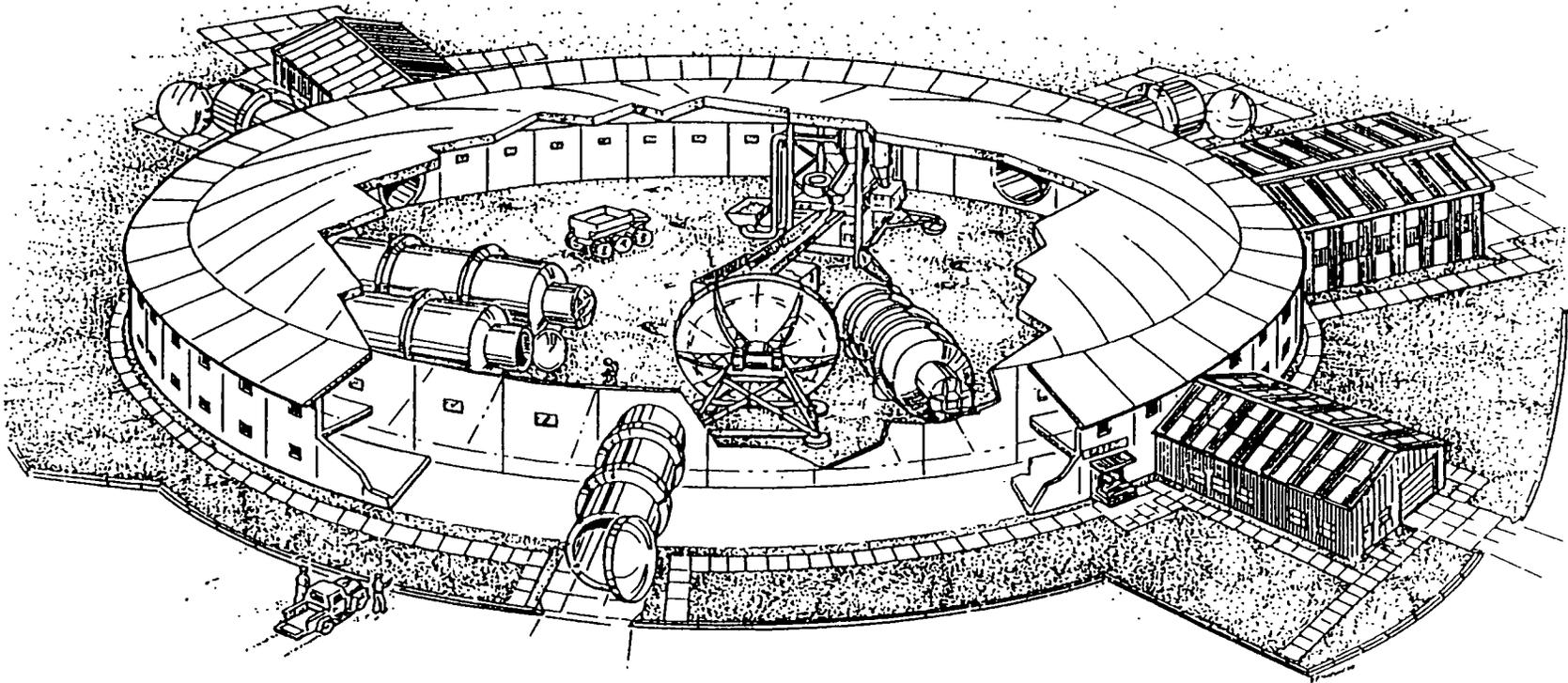
A CENTER FOR
EXTRATERRESTRIAL ENGINEERING AND CONSTRUCTION

THE UNIVERSITY
OF NEW MEXICO

INDUSTRY
SMALL BUSINESS
OTHER UNIVERSITIES

LOS ALAMOS
NATIONAL LABORATORY

WHY CETEC ?

- SIGNIFICANT CHANGE IN ENABLING TECHNOLOGIES FROM AERONAUTICAL ORIENTATION TO CONSTRUCTION, ENGINEERING, MINING, AND CHEMICAL PROCESSES AND EQUIPMENT

- SUCCESSFUL DEVELOPMENT OF A LUNAR OR MARTIAN BASE WILL REQUIRE SIMULATION OF EXTRATERRESTRIAL OPERATIONS ON EARTH

- A NEED EXISTS FOR A CENTER OF KNOWLEDGE AND FOR LARGE SIZED, REALISTIC SIMULATION OF EXTRATERRESTRIAL ENGINEERING, CONSTRUCTION, AND OPERATIONS

    - TO SUPPORT A WIDE RANGE OF CUSTOMERS

    - NO SUCH FACILITY CURRENTLY EXISTS

WHAT IS CETEC ?

- A CENTER OF KNOWLEDGE AND EXPERTISE

    - ENGINEERING, CONSTRUCTION, MINING, CHEMICAL PROCESSING

    - ON EXTRATERRESTRIAL SURFACES

- A SET OF EXPERIMENTAL TEST FACILITIES

    - RESEARCH ON EXTRATERRESTRIAL ENGINEERING AND CONSTRUCTION

    - TEST LARGE PIECES OF EQUIPMENT

    - REALISTICALLY SIMULATED EXTRATERRESTRIAL ENVIRONMENT

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3

308

CETEC
ONE ACRE LUNAR SOIL VACUUM FACILITY

LABORATORY
MODULE

WORKSHOP

ROVER

PROCESSING
PLANT

HABITAT

16 m LUNAR
TELESCOPE

240' DIAMETER

50'

20'

INDUSTRY

STEEL REINFORCED CONCRETE DOME

SIMULATED LUNAR SOIL

ENTRY / EXIT AIRLOCK

EXTERIOR SURFACE AND LOWER SURFACE OF FLOOR COVERED WITH STAINLESS STEEL OR ALUMINUM LINER

STEEL REINFORCED CONCRETE ARCHES FOR FLOOR STIFFENING

SUBSURFACE TANK FOR DRILLING EXPERIMENTS

EXTERIOR
STAINLESS OR
ALUMINUM LINER

THIN INTERIOR LINER

3-0"

LUNAR SOIL SIMULANT

CONCRETE ARCH FLOOR
STIFFENERS

SIMULATED ENVIRONMENT GOALS

- VACUUM (10^{-4} to 10^{-6} TORR)

- REDUCED GRAVITY (1/6 FOR LUNAR)

- HIGH TEMPERATURE

- LOW TEMPERATURE

- LUNAR SOIL SIMULANT

- ILMENITE MINERAL

- DUST

- MARTIAN SURFACE AND ATMOSPHERE
  (7 - 10 MILLIBARS)
    - CO_2 ATMOSPHERE
    - ICE
    - CRYOGENIC TEMPERATURES

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BRIDGE CRANE RAILS
SUSPENDED FROM
DOMED CEILING

"SMART TETHER"
SUSPENSION SYSTEM

MULTIPLE BRIDGE
CRANE DOLLYS

EACH MOVING COMPONENT
COUNTERBALANCED AT
CENTER OF GRAVITY

SUSPENSION CABLES

LUNAR EXCAVATOR

LUNAR SOIL SIMULANT
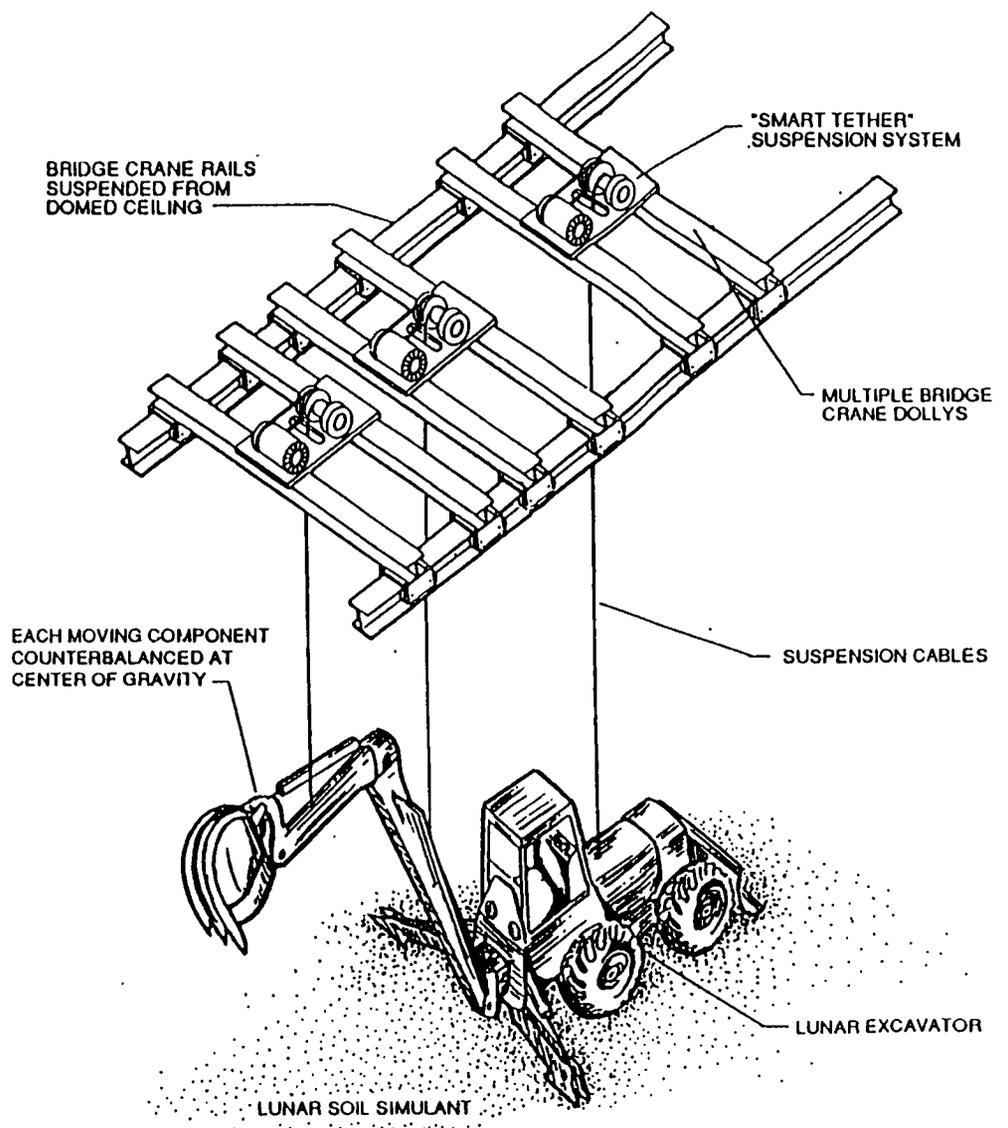
"SMART TETHER" PARTIAL GRAVITY SUSPENSION SYSTEM

PROTOTYPE EQUIPMENT TO BE TESTED IN CETEC

FROM NASA
90-DAY STUDY

CONSTRUCTIBLE
HABITAT CONCEPT

INITIAL HABITAT AND
LABORATORY MODULES
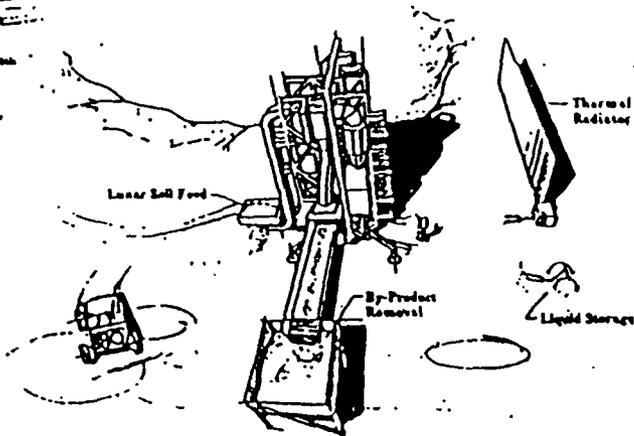
FROM
BATELLE STUDY

LUNAR LIQUID OXYGEN
PRODUCTION PLANT
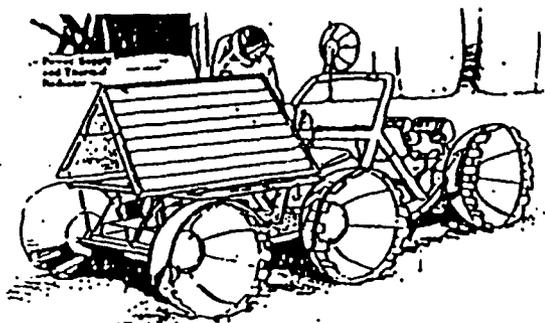
UNPRESSURIZED
MANNED/ROBOTIC ROVER

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OF NEW MEXICO
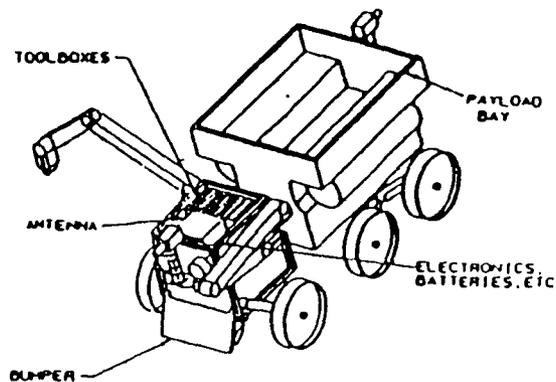
POTENTIAL CETEC
PROJECTS AND EXPERIMENTS

CONSTRUCTION

SOIL EXCAVATION
FOUNDATION CONSTRUCTION
STRUCTURAL ASSEMBLY/JOINING
MASONRY BLOCK CONSTRUCTION
EQUIPMENT OPERATION/HEAT REJECTION
MACHINERY LUBRICATION
WELDING
DUST STABILIZATION
DUST ABRASION

MATERIALS PROCESSING

RECOVERY OF ELEMENTS (O2, S, Ti, He3,)
PRODUCTION OF ULTRA-HIGH STRENGTH GLASS
PRODUCTION OF CONCRETE-LIKE MATERIALS
FABRICATION OF PRODUCTS & COMPONENTS
PLASMA SPRAYED STRUCTURAL COATINGS
MICROWAVE SINTERING

MINING & EXCAVATION

DRILLING
TRENCHING & EXCAVATION
AGGREGATE PRODUCTION
MATERIAL HANDLING

CONSTRUCTION AUTOMATION

ROBOTIC LAYOUT OF LONG BASELINE
    INTERFEROMETER
PRECISION STRUCTURAL ASSEMBLY &
    CONTROL

LIFE SCIENCES

IMPROVED EVA WORK-SUITS
IMPROVED MAN-OPERATED TOOLS

ACCIDENT CONTROL

RAPID DE-PRESSURIZATION
SPACE SUIT DAMAGE & REPAIR
TESTING OF EMERGENCY EQUIPMENT

A BALANCED CONCEPT OF OPERATIONS SUPPORTING A WIDE RANGE OF CUSTOMER NEEDS

GOVERNMENT AGENCIES

RESEARCH
DEVELOPMENT
TESTING
EVALUATION

CETEC

PEOPLE
KNOWLEDGE
FACILITIES

ACADEMIA

BASIC AND LABORATORY RESEARCH

EDUCATION
STUDENT SUPPORT

INDUSTRY

RESEARCH
DEVELOPMENT
TESTING
EVALUATION

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316
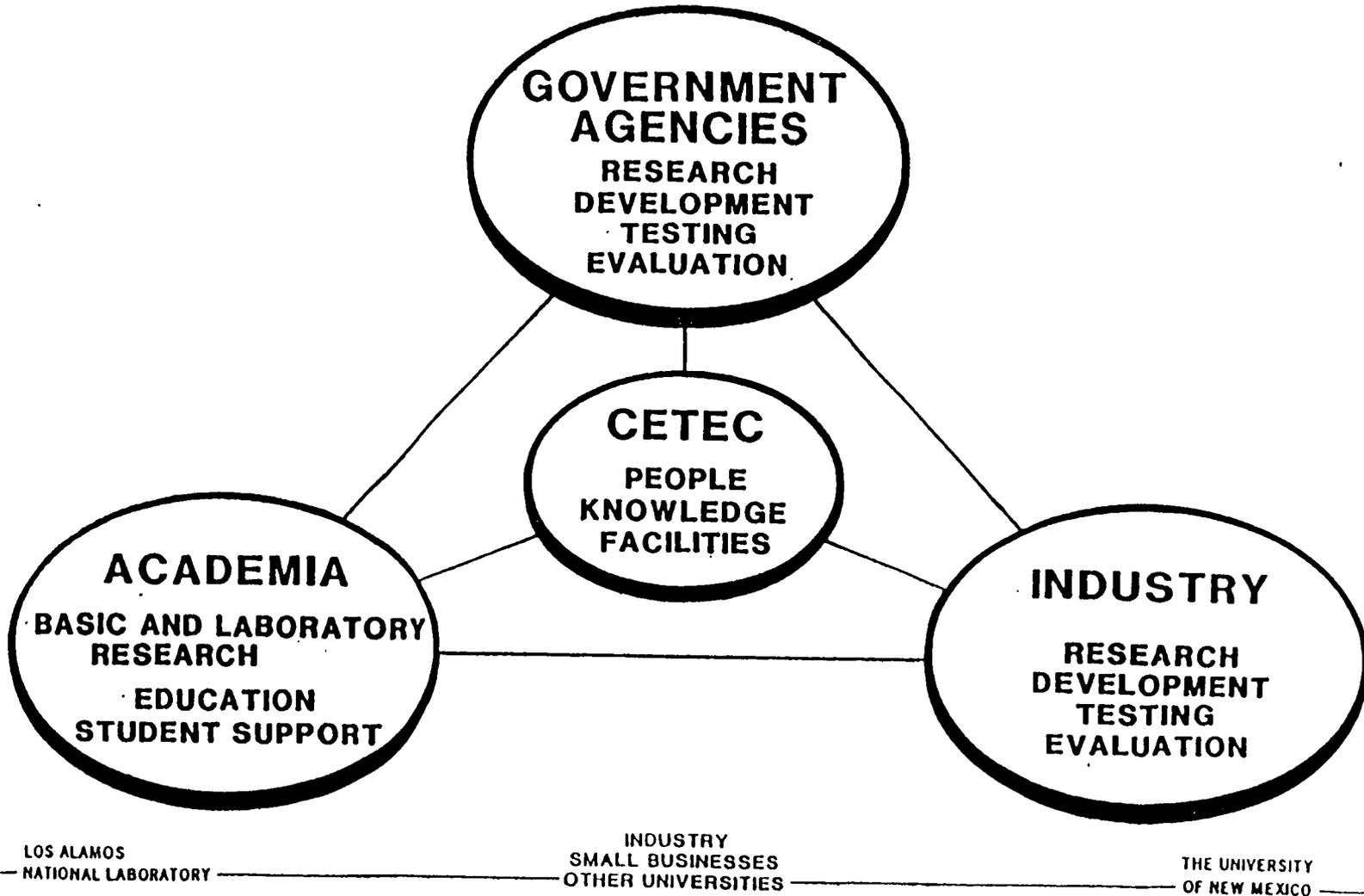
CRYOGENIC STORAGE TECHNOLOGY READINESS
FOR
FIRST LUNAR OUTPOST

John R. Schuster

3rd Space Exploration Initiative Technical Interchange
May 5-6, 1992
Houston, Texas

CRYOGENIC STORAGE TECHNOLOGY IS READY FOR FIRST LUNAR OUTPOST MISSION

- Storable propellant currently baselined for lunar ascent and Earth return

- Cryogen boiloff estimates for lunar surface indicate modest insulation requirements

- Cryogenic propulsion for lunar ascent and Earth return will reduce IMLEO by 20% to 30%

- The required cryogenic storage technology is developed and ready for mission application
    - Foam insulation to limit Earth launch boiloff

    - Thick multilayer insulation for low space and lunar boiloff

    - Thermodynamic vent system for tank pressure control in zero-g and gravity environments

Cryogen Boiloff Predictions

LH2

4840 Lb LH2 in 2 10 Ft. Dia. Spherical Tanks
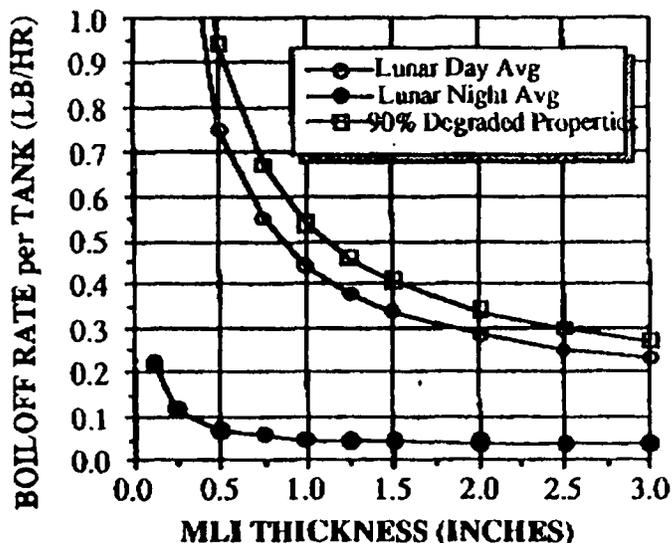
LOX

26,660 Lb LOX in 2 7 Ft. Dia. Spherical Tanks
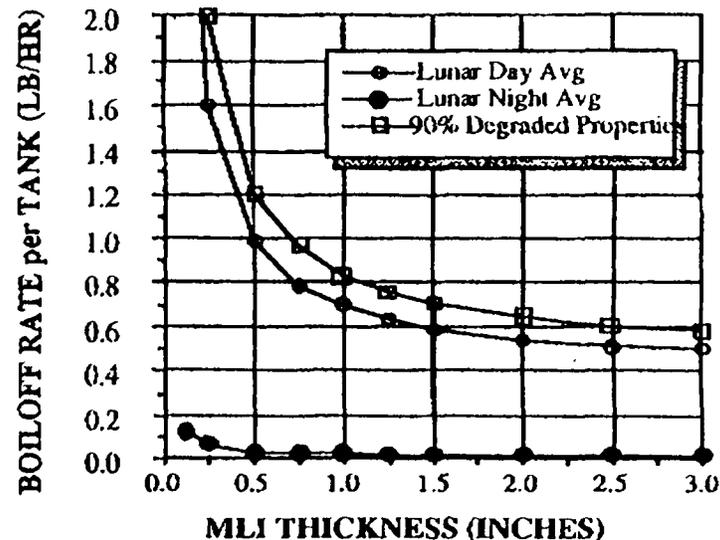
TOTAL PERCENT BOILOFF
vs. STAY TIME
(Degraded Absorptivity)
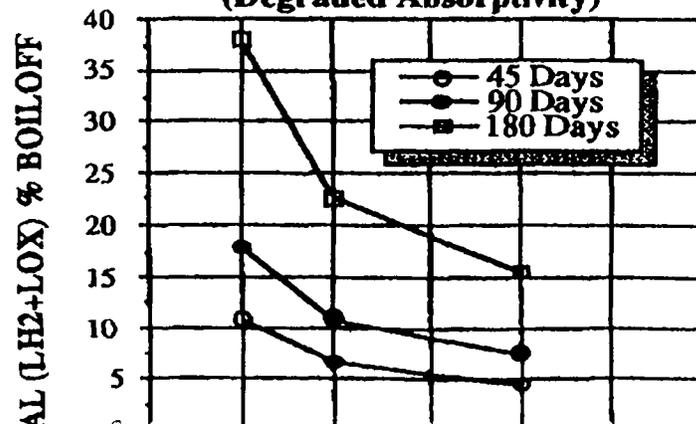
Earth Days	Lunar Days/Nights
45	2/1
90	3/3
180	7/6

Cryogenic vs. Storable Propellant Ascent Stage for Varying Stay Times on Lunar Surface
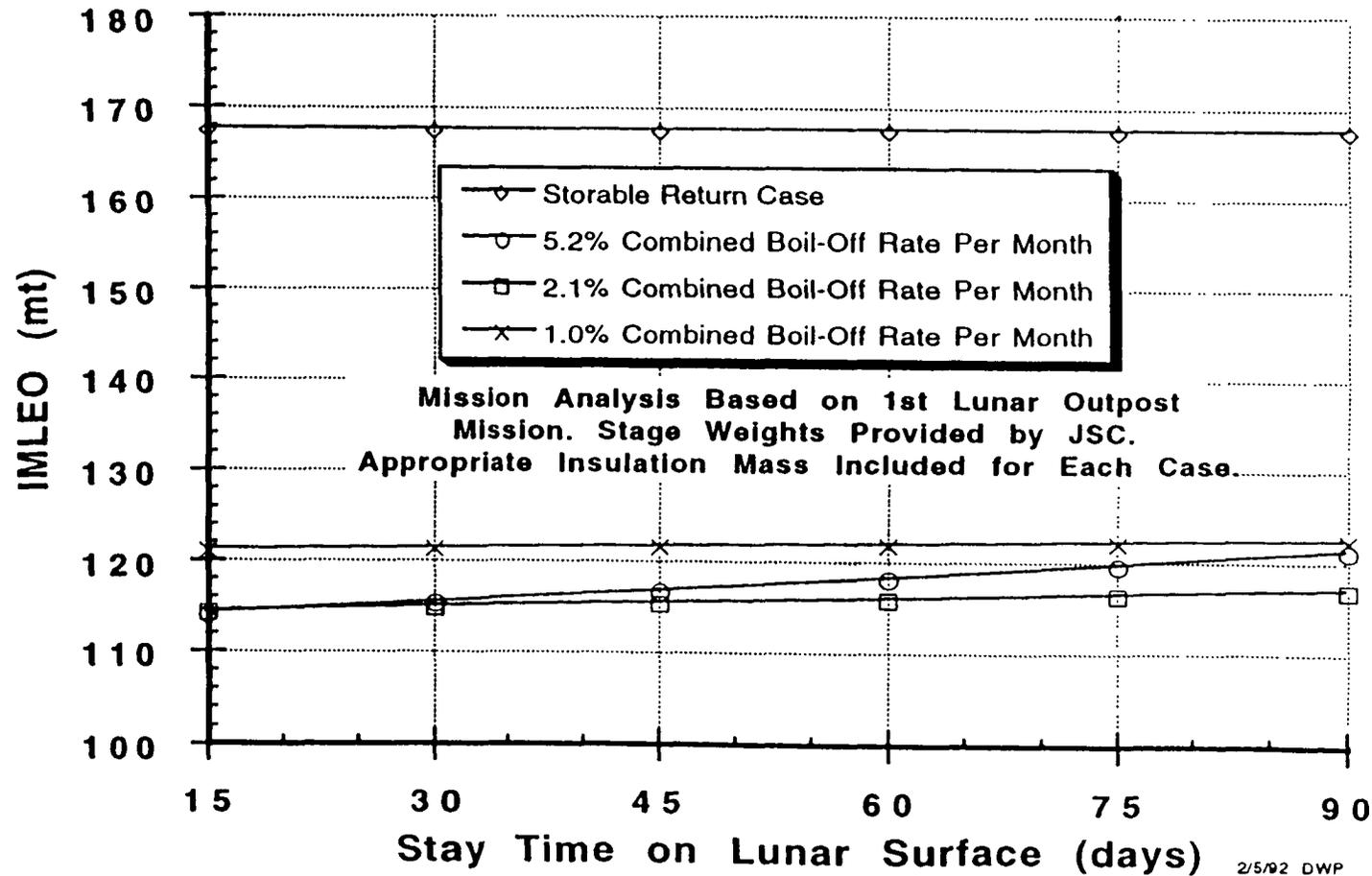
SHUTTLE/CENTAUR THERMODYNAMIC VENT SYSTEM

- Compact, active type TVS concept well-suited for upper stages and space transfer vehicles

- Developed to enable Centaur hydrogen tank venting from within Orbiter cargo bay

- Eliminated the need for settled venting

- Fully ground tested and demonstrated to be insensitive to gravity

- Capable of venting up to 25 lb/hr

- Had nearly completed flight qualification when Shuttle/Centaur program was canceled in 1986

ZERO-G THERMODYNAMIC VENT SYSTEM

Requirement: Vent to Control Tank Pressure Rises Caused
by Propellant Tank Heating

VAPOR

MIXER &
MOTOR

HEAT
EXCHANGER

VENT
INLET

PRESSURE
REGULATOR

OVERBOARD
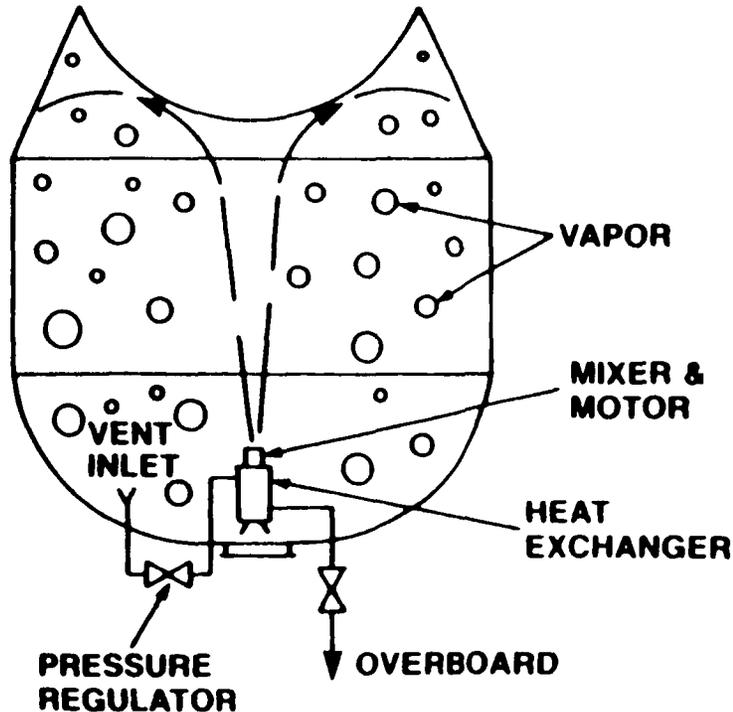
Component Design Requirements

Mixer

- To Provide: (a) Thermal Equilibrium Mixing of Bulk Propellants, and (b) Heat Exchange Mechanism Between Tank Fluid and Vent Fluid

Heat Exchanger

- Lowers Bulk Energy Level
- Assures Pure Vapor Venting Regardless of Fluid Quality at System Inlet

Pressure Regulator

- Controls Vent Side Fluid Pressure

SHUTTLE/CENTAUR THERMODYNAMIC VENT SYSTEM ASSEMBLY

CVE911476
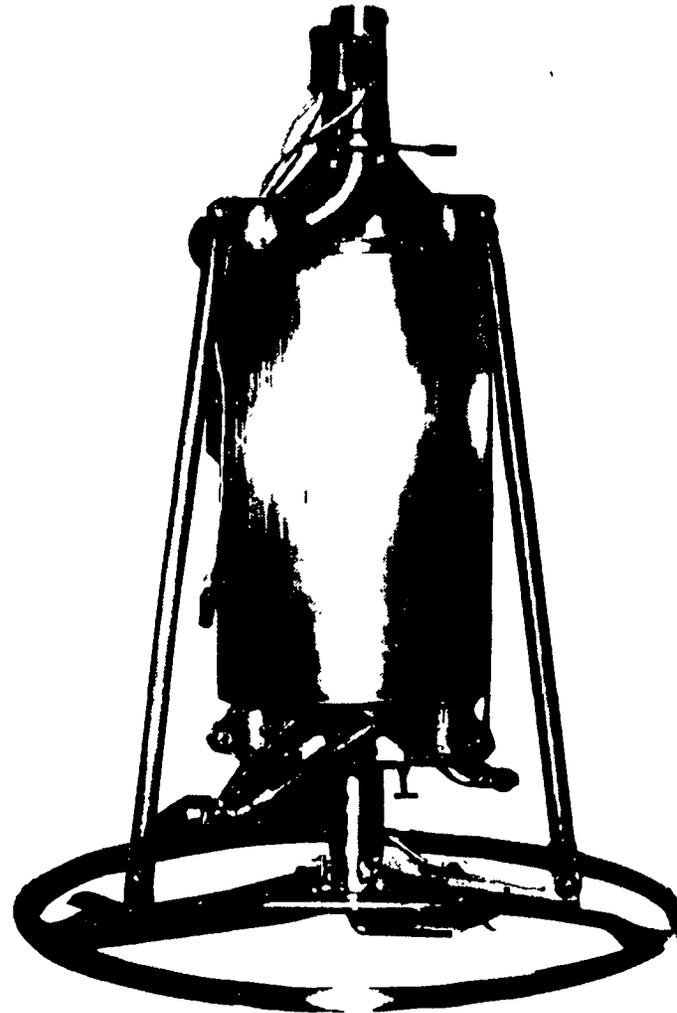
HEAT EXCHANGER / MIXER PUMP MODULE

SHUTTLE/CENTAUR
LH₂ TANK
FORWARD DOOR
OPENING

VENT SIDE
FLOW CONTROL
ELEMENT

MIXER PUMP
ELEMENT
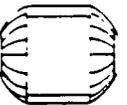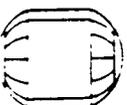
HEAT EXCHANGER
ELEMENT

MOUNTING
STRUTS

37"

26.24"

5.75"

14" DIA.
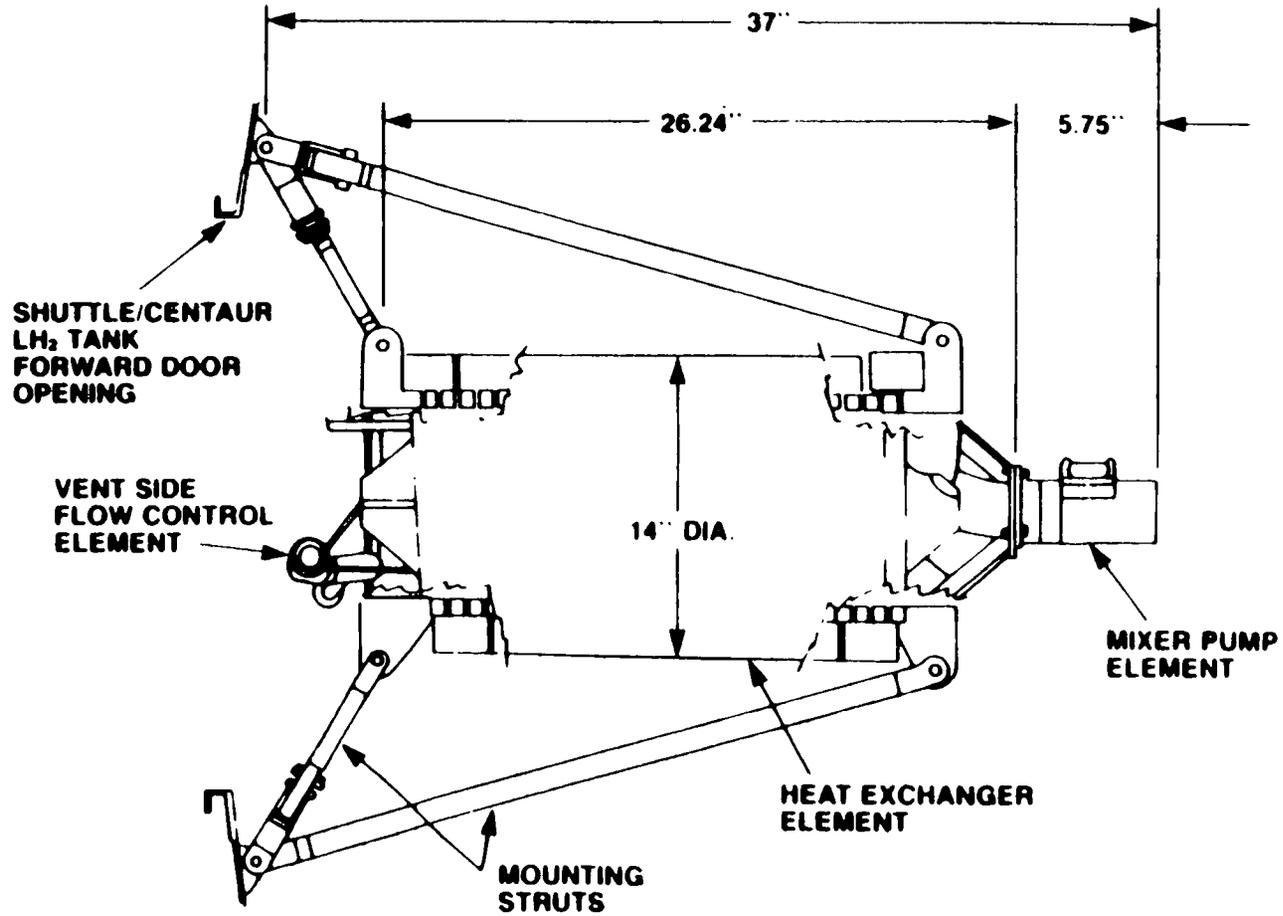
THICK MULTILAYER INSULATION DEVELOPMENT PROGRAM

- Completed by General Dynamics in 1969 under contract to MSFC

- Four-inch thick Superfloc MLI

- Design of system for full-scale tank

- Fabrication and testing of system on 1/4-scale tank

    - Utilized same attachment methods

    - Scaled to achieve same blanket stresses

    - Combined vibration, acceleration and rapid pumpdown testing to simulate launch

    - No degradation in thermal performance

- Technology ready for full-scale application

BLANKET GEOMETRY CONCEPT EVALUATIONS
(4 BLANKET LAYERS, EACH 1.0 IN. THICK)

Configuration	αgore ∢	βconic ∢	Wrinkling Coefficient	Seam Length (in.)	No. of Fasteners	No. of Blankets	Blanket Shapes
1	20 deg (18 gores)	17 deg (2 caps)	1.05	12,390	2,052	80	8
2	24 deg (15 gores)	20 deg (2 caps)	1.52	10,305	1,775	68	8
3	30 deg (12 gores)	25 deg (2 caps)	2.35	8,256	1,349	56	8
4	(3 cyl. segments)	24 deg (2 caps) (10 domes)	2.25	7,806	1,296	60	16
5	(3 cyl. segments)	18 deg (2 caps) (16 domes)	1.26	9,610	1,600	84	20
6	30 deg (12 gores)	24 deg (2 caps) (4 domes)	2.35	8,524	1,418	72	12
7	20 deg (18 gores) (3 cyl. segments)	17 deg (2 caps)	1.05	12,410	2,060	164	12
8	24 deg (15 gores) (3 cyl. segments)	20 deg (2 caps)	1.52	10,857	1,810	140	12
9	30 deg (12 gores) (3 cyl. segments)	25 deg (2 caps)	2.35	9,348	1,560	116	12
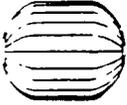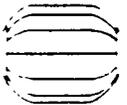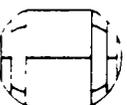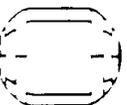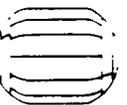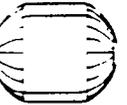
Rating: 1 = Configuration No. 4
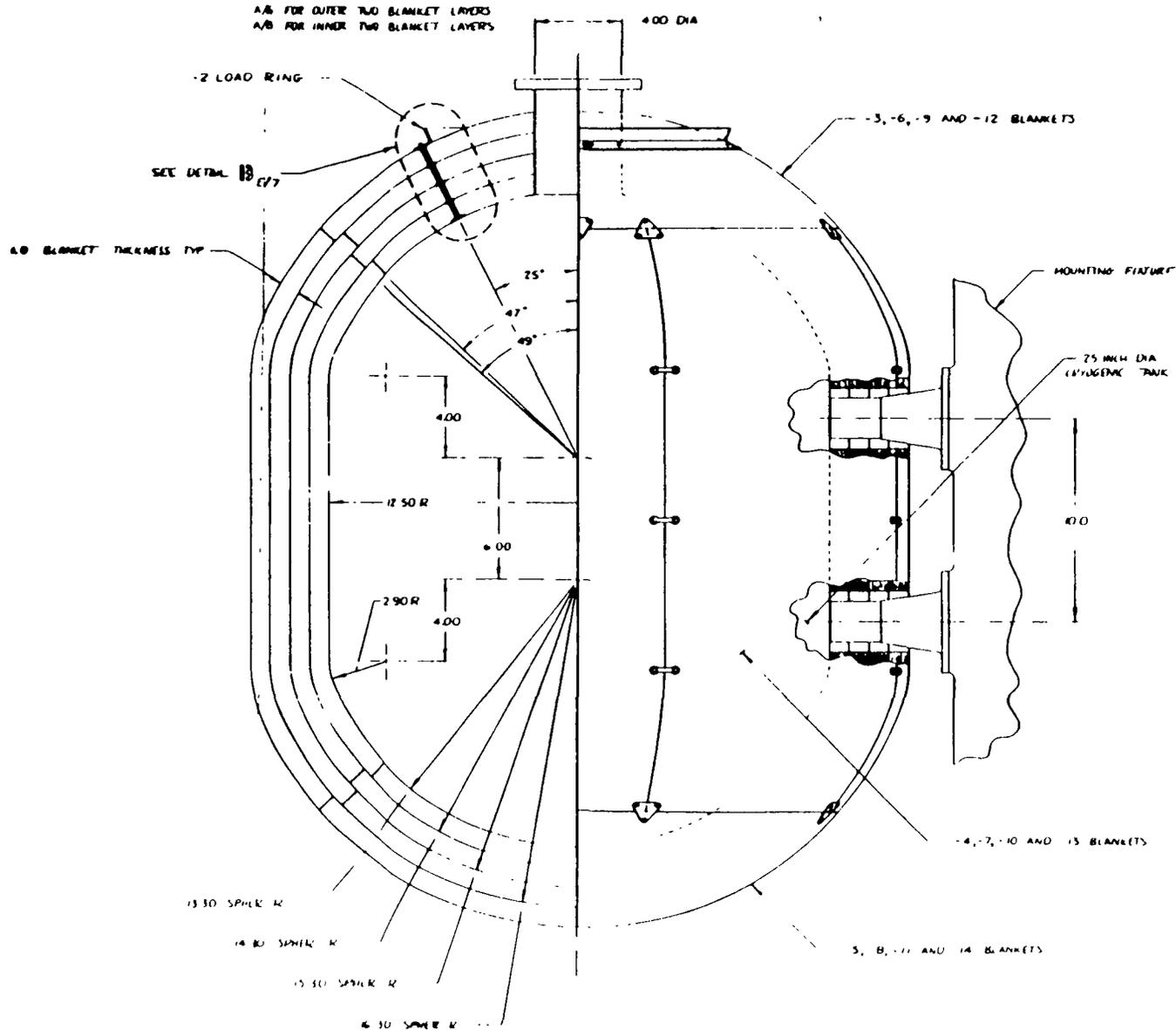   2 = Configuration No. 3
         3 = Configuration No. 6

FOUR-INCH THICK MLI SYSTEM
ON 1/4-SCALE TEST TANK

FOUR-INCH THICK MLI SYSTEM
ON 1/4-SCALE TEST TANK

COMBINED ENVIRONMEMTS OF VIBRATION, ACCELERATION, AND TEMPERATURE TESTING (CEVAT)
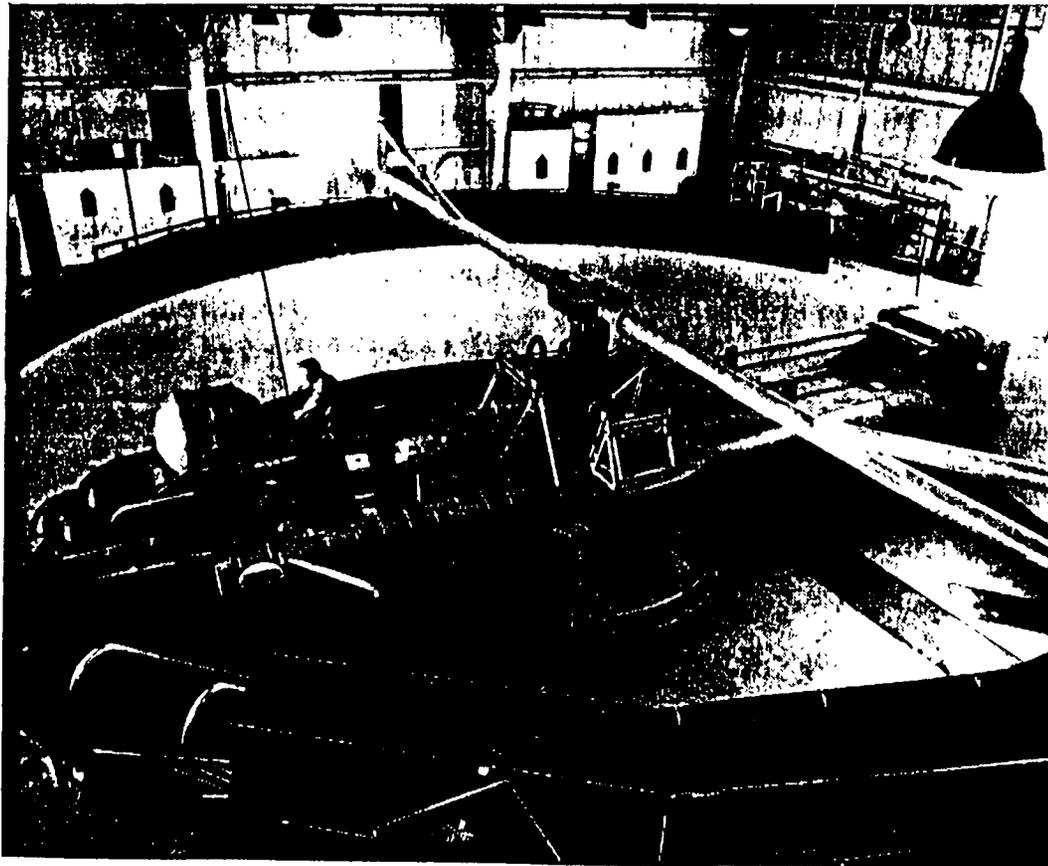
25-INCH TANK
Predicted & Actual Thermal Performance

ITEM	THERMAL ANALYSIS	SPACE ENVIRONMENT 1st THERMAL TEST	2nd THERMAL TEST	GROUND HOLD THERMAL TEST
Cryogen	LH_2	LH_2	LH_2	LH_2
Chamber Pressure	1×10^{-6} torr	2.7×10^{-6} torr	2.8×10^{-6} torr	14.7 psia
Tank Pressure	14.7 psia	14.7 psia	15.3 psia	14.7 psia
Tank Percent Ullage	10	10	10	40
Sink Temperature	40°R	37°R	37°R	37°R
Source Temperature	535°R	523°R	520°R	390°R
Heat Flux, Total	0.19 B/hr. ft.^{2}	0.19 B/hr. ft.^{2}	0.18 B/hr. ft.^{2}	50 B/hr. ft.^{2}
Insulation	17%	—	—	—
Seams	24%	—	—	—
Pins	59%	—	—	—
Penetration	0%	—	—	—
Heat Flow Rate	4.4 B/hr.	4.6 B/hr.	4.3 B/hr.	1160 B/hr.
Mass Flow Rate	0.023 lb./hr.	0.024 lb./hr.	0.023 lb./hr.	6.2 lb./hr.

CENTAUR FIXED FOAM INSULATION

- Insulates tank against atmospheric heating on pad and during ascent

- Controllable and well-characterized

- Tailorable, and well-suited as a substrate for multilayer insulation

- Operational on Atlas II vehicles

SYSTEM REQUIREMENTS

- Stainless steel tank

- Semi-flexible foam and adhesive

- Maintain structural integrity

    - Aerodynamic heating during atmospheric ascent

    - Prevention of spallation due to cryopumping

    - Adhesive bond for multiple tankings

- Available safe/nontoxic materials

- Low cost

- Low weight

332

GSV01 2737

INSULATION SYSTEM DESIGN
Material Selection

Polyvinyl chloride (PVC) foam

Stainless steel tank skin

Modified epoxy
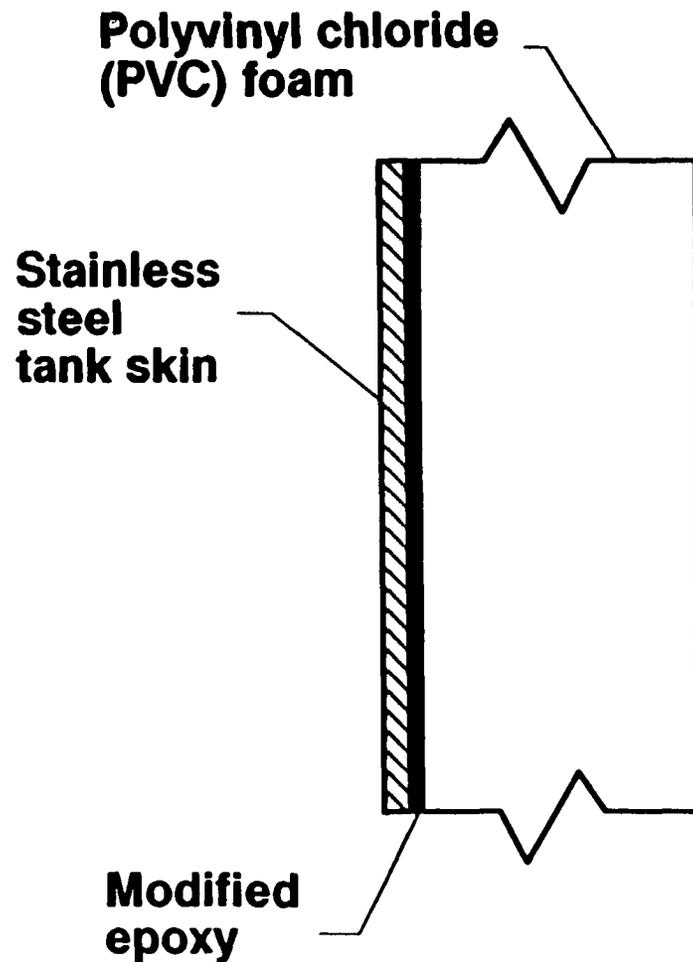
PVC foam characteristics

- Heat formed to tank contour
- Smooth surface finish
- Closed-cell/no sealant
- Nontoxic/noncorrosive

Adhesive characteristics

- Mixed with solvent/sprayable
- Good working pot life
- Room temperature cure

FIXED FOAM ON OPERATIONAL ATLAS II
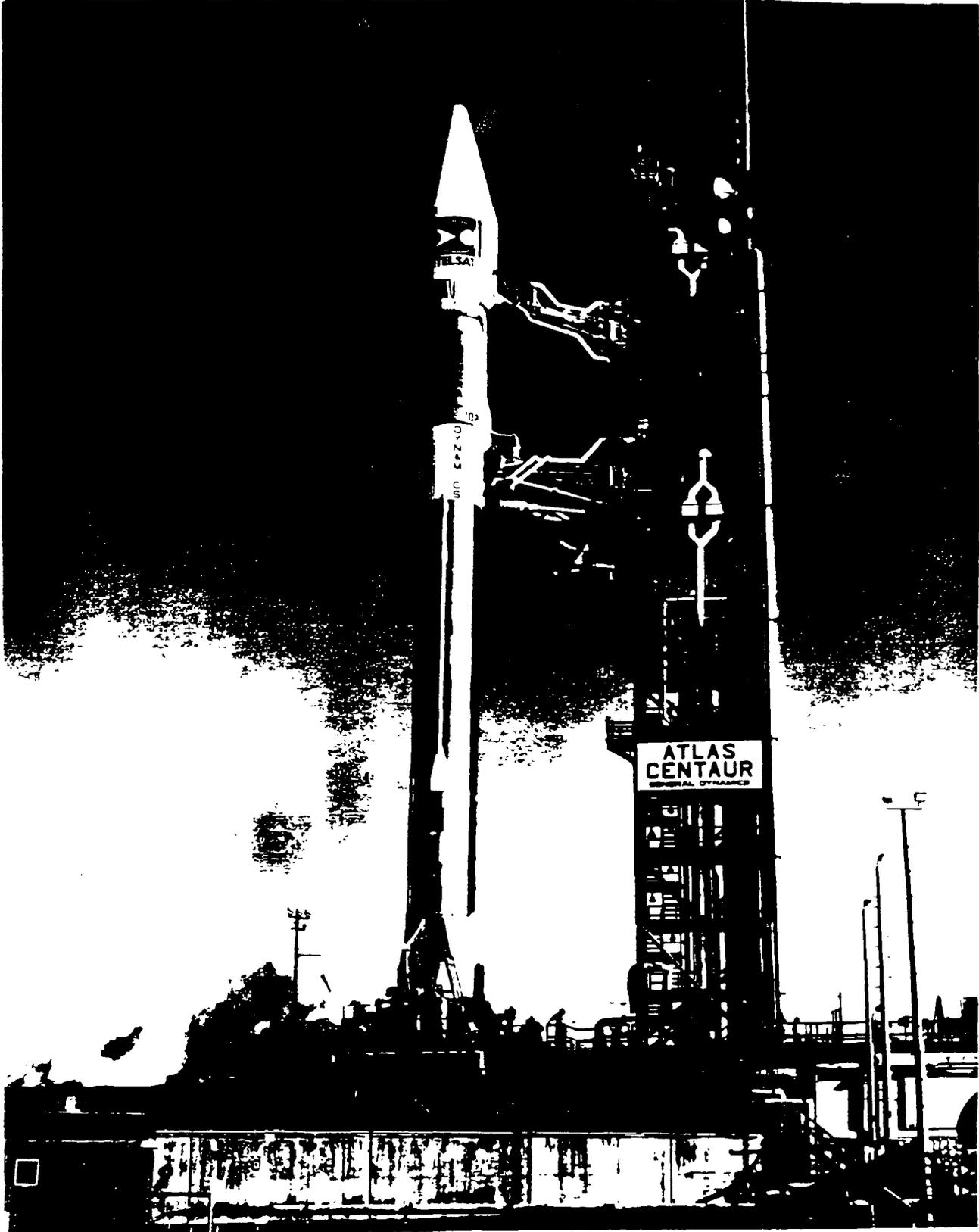
CLOSING COMMENTS

- Cryo propulsion for First Lunar Outpost ascent and return shows a 20% to 30% reduction in IMLEO over storable propulsion

- Cryo storage technology requirements are modest, compared to a Mars mission or an orbital depot

  - Combined Earth/space insulation providing low boiloff

  - Zero-gravity venting for tank pressure control

- Cryo technology to perform the lunar mission is available, has been adequately demonstrated, and is ready for full-scale application

  - Shuttle/Centaur compact thermodynamic vent system

  - Atlas II fixed foam insulation

  - Fully-tested thick MLI

AIAA 92-1031
Inflatable Structures for a Lunar Base Habitat

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Center for Engineering Infrastructure and Sciences
in Space
Colorado State University
Fort Collins, CO

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INFLATABLE STRUCTURES FOR A LUNAR BASE HABITAT

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ABSTRACT

Design and construction of a structure on the Moon requires addressing a host of issues not encountered on Earth. A modular quilted inflatable structure consisting of thin membranes of composite material integrated with supporting columns and arches is proposed. An initial linear analysis of the proposed structure is briefly reviewed. The actual response of an inflatable membrane is nonlinear and, hence, a nonlinear numerical analysis for the stresses and displacements was undertaken. Initial results clearly indicate that an inflatable structure is a feasible concept and is ideally suited for a lunar structure.

INTRODUCTION

A human-tended outpost on the Moon that will evolve into a functional base is a crucial stepping stone in the expansion of humanity into space. A lunar base is one of the prime missions of the Space Program and has been proposed in numerous studies in recent years. The establishment of human-tended base on the Moon wa recommended by the Nationa Commission on Space report[1] as national mission, by the Ride repor as one of the four initiatives neede to ensure U.S. leadership in space the 21st century, and in the studie of the NASA Office of Exploration[3] the case study for human exploratic of space. On July 20, 1989, tl 20th anniversary of the first huma landing on the Moon (Apollo 11) President Bush called for a permaner human return to the Moon before 20. to be followed by a manned mission Mars. The long-term plan for huma expansion into space and tl establishment of a lunar base wa studied by the NASA 90-Day Report and further recommended by tl Augustine Report.[5] The Staffor Report[6] recommends four spac architectures, all of which incluc different levels of lunar ba: development.

The establishment, constructic and existence of a lunar base ai contingent upon the development of structure capable of accommodating lunar habitat, i.e., a Luna Engineered Closed/Controlled Ec System (L-ECCES).[7] An L-ECCE consists of human, plant and anima modules and associated scientific manufacturing and mining modules.

The design of a lunar structur requires addressing a host of issue

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that are not encountered on Earth. A structure on the Moon must meet both the harsh and benign lunar environmental conditions, and as a result, unconventional structural concepts must be explored. The primary lunar environmental conditions to be considered for a structure include: (1) an atmosphere that is essentially a vacuum; (2) no weather conditions; (3) gravity of 1/6 g; (4) surface temperature variations that range from a minimum of -173 deg C (-279 deg F) during the lunar night to a maximum of 127 deg C (261 deg F) during lunar day[8] which results in a 300 deg C (540 deg F) temperature variation; (5) harmful solar flare protons and galactic cosmic radiation such as HZE particles (high charge-Z and high energy-E particles); (6) meteorite and micrometeorite impact; (7) minimal seismic activity; and (8) specific regolith (lunar soil) properties at the selected structure site.

The loads applied on the inflatable structure membrane consists of: (1) the internal pressure, selected to be 69 kPa (10 psi) which corresponds to an elevation of about 3050 m (10000 ft) on Earth; (2) the dead load (gravity load) induced by the structure's material weight; (3) the gravity load of a layer of regolith of about 3.3 m (10 ft) for radiation and micrometeorites shielding[9] which amounts to a load of 7.9 kPa (1.15 psi); and (4) the thermal stresses caused by the lunar temperature variations.

Light transportation weight, expandability, small stowage volume, flexibility, low volume to usable floor area ratio, modularity, durability, safety, reliability, and short construction time with minimal extravehicular activity are issues to be considered in the design and construction of a lunar structure. Constraints on the cost of transportation, materials, construction time, construction equipment, and architectural requirements must be incorporated in the design and construction processes.

Inflatable structures are ideally suited for a L-ECCES in view of their features. The primary loading on the membranes of an inflatable structure in the lunar environment is the internal pressure which, if designed correctly, will induce only tensile stresses. These stresses make the most efficient use of material strength since there are no stability problems associated with tension. Membrane materials typically are of low density and flexible, and therefore, transportation costs, construction time and the amount of equipment required are reduced compared with traditional Earth-bound construction materials (e.g., concrete, metals and wood). In addition, the structure can be tested for constructability and pressure containment on Earth prior to the actual construction on the Moon.

LINEAR ANALYSIS

A concept for inflatable structures in a lunar habitat was initially proposed by Vanderbilt, Criswell and Sadeh[10] and was further refined by Nowak, Criswell and Sadeh.[11] This concept is based on a structure comprised of identical inflatable modules. Each module consists of the following structural components: (1) four external wall membranes; (2) a roof and a floor membrane; (3) four inflatable columns

with footings; and (4) four rigid arches. A sketch of this inflatable structure is shown in Fig. 1, and a photograph of a model, built to a scale of 1:80, is given in Fig. 2. A 3.3 m (10.0 ft) thick layer cover of regolith primarily for radiation shielding, and also for meteorite and thermal protection,[9] is shown in both figures.

The size of the basic module was determined based on the size of a typical office and\or living room in terrestrial structures. A spacing of 6.1 x 6.1 x 3.0 m (20 x 20 x 10 ft; l x w x h), was selected. A radius of curvature of the roof membrane of 6.1 m (20.0 ft) was chosen based on a compromise between reducing wasted internal volume (a low radius) and lowering the induced stresses (a higher radius). Attributes of this modular approach include modularity, a minimal number of structural components to facilitate manufacturing, expandability through any of the exterior wall membranes, the ability to isolate a pressure loss with interior pressure resistant partitions, and a low volume to usable floor space ratio.

Rigid arches are integrated with the roof membranes to stabilize the structure, limit the roof deformations, and support the gravity loads when there is a loss of internal pressure. This is necessary since the internal pressure supports the inflated structure. One concept for rigidizing the arches is to fill them with a structural foam that remains flexible until the foam is vented to a vacuum. Once the foam is subjected to the vacuum it becomes rigid. This concept was developed by the Goodyear Aerospace Corporation in the 1970's with promising results.[12]

The membrane material in the columns holds the roof down and acts in tension when the structure is pressurized. If there is a pressure loss, the columns would support the rigid arches and act in compression. The compressive capacity of the columns is provided by pressurizing them.

Kevlar 49, a material widely used in space applications, which can be woven into a membrane with a tensile strength of 690 MPa (100 ksi), was chosen as the membrane material for the case study. Calculations for the stresses in the roof and column membranes were conducted based upon the linear elastic response of a pressurized sphere and cylinder, respectively.[11] Thicknesses of 0.3 mm (0.012 in) for the roof membrane and 1.94 mm (0.076 in) for the column membrane, based on a 61 cm (24 in) diameter, were found to be structurally adequate. Closed form solutions for the stresses in the wall membranes and rigid arches from a linear analysis have not been developed yet.

Based upon the results of the linear analysis, inflatable structures offer an efficient, practical and economic solution for a lunar structure. The next step is to conduct a nonlinear analysis of the entire module in order to check, verify and refine the linear results and modify the structure if necessary.

NONLINEAR ANALYSIS

Membranes are made from thin sheets of materials that are formed into the desired geometry of the structure. They can only transmit loads through the plane of the material. Any strength through

transverse bending and shear is negligible. The structural behavior of membranes is governed by a set of homogeneous coupled fourth order nonlinear partial differential equations with constant coefficients.[13] In order to perform a structural analysis for the actual stresses and deformations induced by the loads, these equations must be solved. Very few closed form solutions exist for the response of membrane structures due to their nonlinear behavior. Numerical solution techniques are employed to overcome this difficulty. It is important to note that the solutions for the deformations and stresses are nonlinear even if the material remains linear and elastic under the applied loads.

To perform a numerical nonlinear analysis of membrane structures, two approaches that involve specifying the geometry of the structure are possible. The first approach is to describe the initial or unstressed geometry and then to proceed with the analysis. A second approach is to define the final or stressed geometry before conducting the analysis. The first approach requires incremental or iterative nonlinear solution techniques to solve for the deformations and stresses, and is desirable from a fabrication view point. In the second approach, the initial shape for fabrication is found from the analysis, but may be impractical to fabricate. Consequently, the first approach is applied since the fabrication of the module is simplified.

SOLUTION TECHNIQUE

The Finite Element Method (FEM) was chosen as the numerical technique to perform the nonlinear structural

analysis. This method was selected since it is widely utilized in most of the recent advances in nonlinear structural analysis. This method requires the geometry of the discretized structure to be input. A finite element software package that has the capability of analyzing nonlinear membrane structures was used.

Results from a structural analysis are only as accurate as the description of the initial membrane shape since errors accumulate during the incremental solution techniques. In order to obtain accurate results, a computer program was written to generate the geometries of the structure. This program (GEOMM) computes the locations of specified points on the surface of the structure that are required in the FEM. The code was written so that a wide variety of structural shapes and finite element meshes can be generated for immediate input into the software package.

An analysis was performed on a module that is located at an external corner of the structure. This selection was made since there would be no "balancing" of the stresses and deformations by an adjacent module, as occurs in an internal module. One fourth of a module was analyzed due to its symmetry. The rigid arch was modeled with the structural properties of balsa wood since this material is lightweight yet strong, the properties required of the actual structural foam. The finite element mesh of the module is shown as dashed lines in Fig. 3.

RESULTS

Input for the nonlinear structural analysis consisted of the same loadings, geometries, material properties and strengths used in the linear analysis. The same material found to be adequate for the roof membrane from the linear analysis was used as a starting point for the roof, floor and wall membranes. Data for all of the component geometries were generated using the GEOMM code.

The initial nonlinear analysis resulted in maximum tensile stresses in the roof membrane within 20% of those in the linear analysis. Variations of stresses throughout the entire roof membrane were obtained. These stresses can not be found from a linear analysis. Areas of compressive stresses occurred at two locations within the roof membrane. Since membranes cannot resist compressive stresses, "wrinkles" developed at these sites. To eliminate this problem, the initial geometry of the roof membrane was varied so that the material is stiffened and more tensile stresses are developed at these locations. The results of the nonlinear analysis revealed that the thickness of the roof membranes was adequate.

Contour plots of the von Mises stresses in the roof membrane are shown in Fig. 4. Stresses in the roof membrane reached a maximum value of 580 MPa (84 ksi; 84% of the yield stress) at the center. At the corners of the roof membrane the material is isolated by the intersecting arches and the stresses are reduced. This situation is advantageous since the connection detailing there is complicated if high stresses exist.

Other problems were revealed with the original structural concept. One was at the location of the intersection of the floor and external membranes. Results indicated that there are large stress concentration there. To alleviate this problem, another rigid arch was placed between the columns at the floor level. This change aids in the connection detailing at the intersection of different membranes.

The rigid arches were modeled with a diameter of 46 cm (18 in) with solid finite elements. Results indicates no stress concentrations and a maximum tensile stress of 10.4 MPa (1.5 ksi). This stress is less than the strength of balsa wood of 21 MPa (4.0 ksi). Compressive stresses were 9.5 MPa (1.4 ksi) also within the strength limitations.

Results of the analysis for the columns, which were modeled as pressurized tubes, revealed localized stresses and deformations. The concentrations are caused by the membrane forces induced by the two external wall membranes acting on the column material. This problem was anticipated but can not be found from the linear analysis. Based on these results, the columns were modeled with the same properties as the arches since the solid arches reveal no localized stresses or deformations. The nonlinear analysis of the columns revealed that a diameter of 46 cm (18 in) was adequate and yielded a maximum tensile stress of 1.1 MPa (0.2 ksi) and a maximum compressive stress of 4.4 MPa (0.7 ksi). These stresses were caused by bending in the columns from the wall membrane loadings. It is important to note that these compressive stresses are higher than those that would occur in an interior

column and that stability is not an issue.

The results for the stresses in the wall membrane revealed that the material was overstressed by 70%. This resulted because the stresses are not evenly distributed throughout the wall membrane since it is not a segment of a sphere. To eliminate this overstressing, the material thickness of the wall membrane was increased to 0.6 mm (0.022 in). Stress contours for the wall membrane are shown in Fig. 5. These results show in a maximum von Mises stress of 621 MPa (90 ksi; 90% of the yield stress).

Displacement contours of the entire module under the loadings of the internal pressure and regolith are shown as solid lines in Fig. 3. These displacements are magnified by a factor of 4 for clarity. The maximum displacement occurs at the center of the external wall membrane of 12.7 cm (5.0 in) outward. A displacement of 4.3 cm (1.7 in) upward occurred at the center of the roof arch. The displacement at the center of the roof arch is 1.8 cm (0.7 in) downward. This was caused by the external wall membrane pulling it down.

The results from the nonlinear analysis reveal a different pattern for the stresses and displacements in the roof membrane than the linear analysis. The magnitudes of the maximum stresses are within 20%. The resulting stresses and deformations are within tolerable limits for serviceability and strength requirements. Results from this nonlinear structural analysis reveal that the proposed inflatable structure is very feasible for a lunar base.

CONCLUSIONS

The use of an inflatable membrane structure for a lunar base is addressed. Initial calculations from a linear analysis demonstrated that the structure is feasible for a lunar base. The actual behavior of membrane structures is inherently nonlinear. Results from a nonlinear analysis reveal that the proposed structure is feasible with some modifications. Revisions from the nonlinear analysis include: a thicker wall membrane (0.6 mm; 0.022 in); the use of a solid column with a reduced diameter of 46 cm (18 in); and, a 46 cm (18 in) diameter arch. The induced stresses and deformations from the internal pressure load of 69 kPa (10 psi) and the gravity load of a 3.3 m (10 ft) layer of regolith shielding are within acceptable limits. Results from the nonlinear analysis reinforce the initial results of the linear analysis. Thus, inflatable structures are ideally suited for use in a lunar base habitat.

ACKNOWLEDGEMENT

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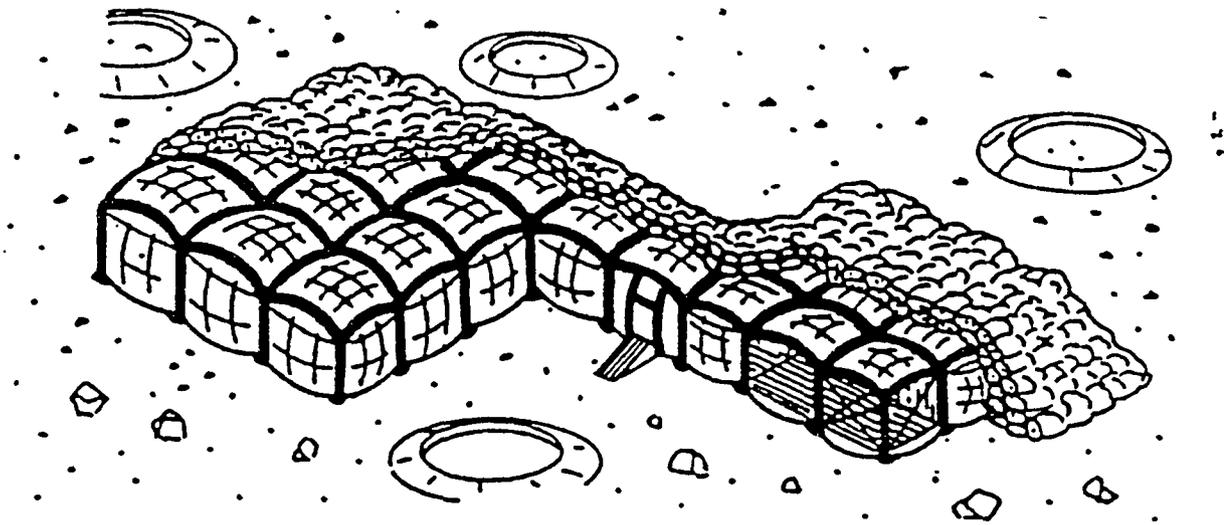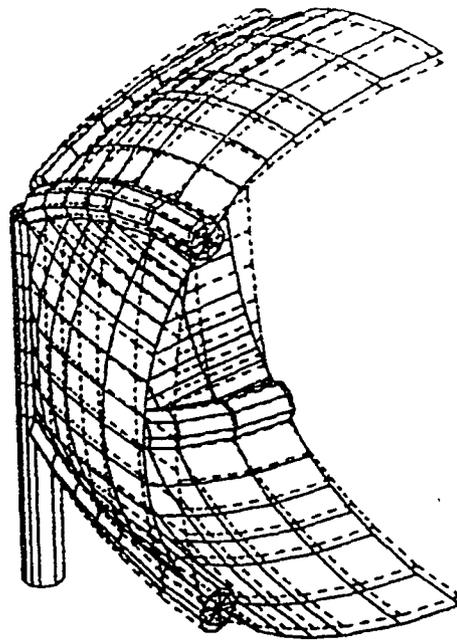[13]Leonard, J.W., Tension Structures, Behavior and Analysis, McGraw-Hill, 1988.

Fig. 1 Overall view of the inflatable structure including a cutaway.
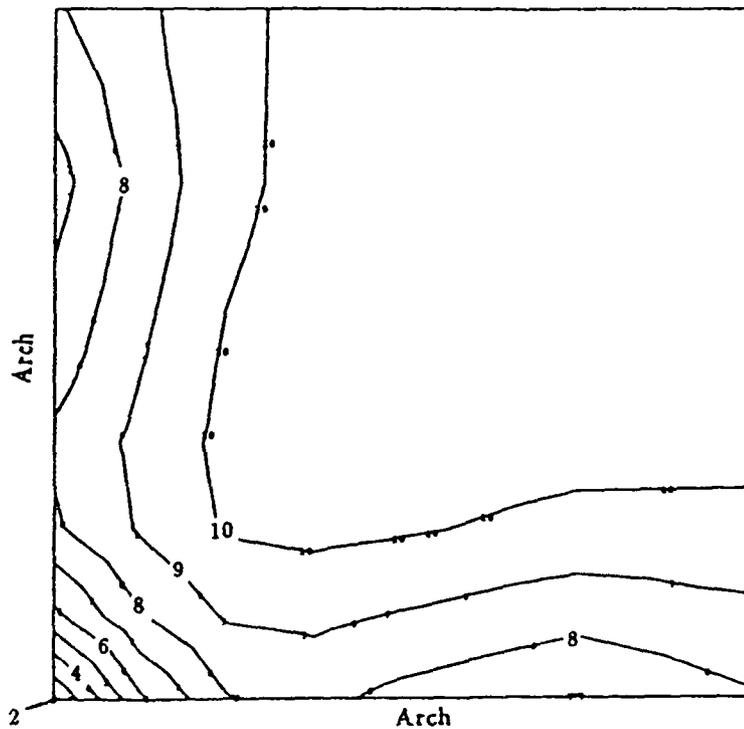

Fig. 2 A physical model of the inflatable structure including a cutaway (scale 1:80).
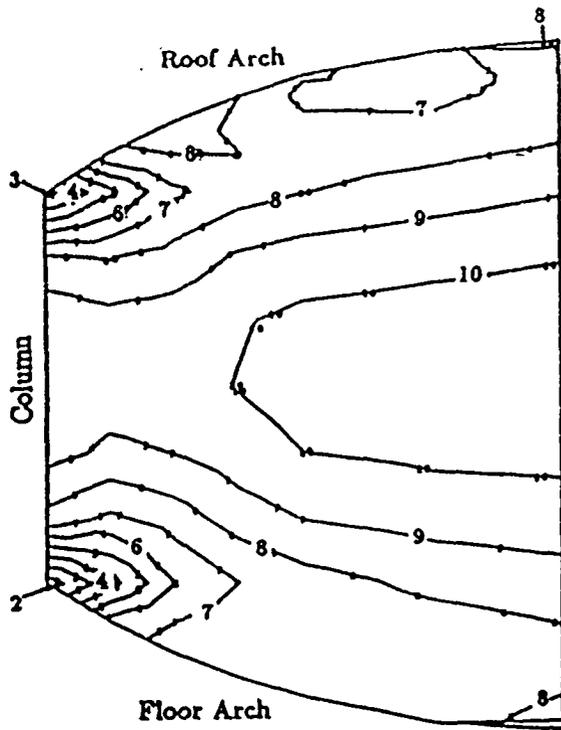
Solid line — displaced shape
Dashed line — original shape
Magnification factor = 4

Fig. 3 Displaced shape of one–fourth of the module.

Contour	von Mises str‹ (% of yield)
1	0.20
2	0.27
3	0.34
4	0.41
5	0.48
6	0.55
7	0.62
8	0.69
9	0.76
10	0.83
11	0.90

Fig. 4 Stress contours of the roof membrane.

Contour	von Mises stress (% of yield)
1	0.10
2	0.18
3	0.26
4	0.34
5	0.42
6	0.50
7	0.58
8	0.66
9	0.74
10	0.82
11	0.90

Fig. 5 Stress contours of external wall membrane.

PRESENTATION #7
SEI TECHNICAL INTERCHANGE: 5 MAY 1992
UNIVERSITY OF HOUSTON AT CLEAR LAKE

PRESENTATION #7

RELATON OF THE LUNAR POWER SYSTEM TO THE SEI PROGRAM AND TO LANDERS

presentation by

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presentation based on the following paper

IAA-91-699:   INTERNATIONAL LUNAR BASE AND LUNAR-BASED POWER SYSTEM TO SUPPLY EARTH WITH ELECTRIC POWER

by

Dr. David R. Criswell

and

Dr. Robert D. Waldron

15339 Regalado St.
Hacienda Heights, CA 91745
818-333-6991

presented in
5th Symposium on International
Space Plans and Policies
Session on
Lunar Base Study Follow-On
7 October 1991

A-7

TERNATIONAL LUNAR BASE AND LUNAR-BASED POWER SYSTEM TO SUPPLY EARTH WITH ELECTRIC POWER

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ABSTRACT

The people of Earth will need more than 20,000 billion watts (GWe) of electric power by 2050 for high level of prosperity. Power needs in the 22nd Century could exceed 100,000 GWe. By 2100 the al quantity of thermal energy used could fully deplete the known inventory (10^7 GWt-Y) of all non-newable sources on Earth except for deuterium and hydrogen for use in proposed fusion reactors. e Table summarizes the labor, capital, and mass of power plants required to produce 1 GWe-Y of ergy from present-day power plants. Fossil and nuclear plants respectively consume 80 to 190 M$ d 12 to 48 M$ of fuel per GWe-Y.

wer ants	Labor Work-Y	Capital $10^6$$	Plant Mass tons	Net Energy
ssil	260	200	10,000	3 to 4
ssion	800	250	41,000	3.3
SP	1,500	470	314,000	11.5
SP	3,100	760	434,000	1.4
S	<20(Earth) <1 (moon)	20	5,200	90 rectn 200 moon

ble: Generation of 1 GWe-Y of Energy

The Lunar Power System (LPS) uses solar power bases on the moon to beam electric power to Earth (Criswell and Waldron 1990, 1991a, b). The LPS in the figure supplies load-following power to rectennas on Earth. Additional solar power conversion units are located across the lunar limb from their respective Earthward transmitting stations. LPS can be augmented by mirrors in polar orbit about the moon. The construction of rectennas on Earth determines the base cost (0.001s $/kWe-H) of LPS power. Stafford (1991) recommends study of LPS.

A manned International Lunar Base (ILB) can celerate the development of LPS by providing the itial transportation and habitation facilities and base erations. ILB can greatly reduce up front costs and ·ks by emplacing a moderate scale LPS (1-100 GWe).

LPS can accelerate the development of the ILB providing greater funding than is reasonable to pect for purely scientific research. An ternational ILB/LPS program can foster world trust d prosperity.

Figure: LPS Schematic

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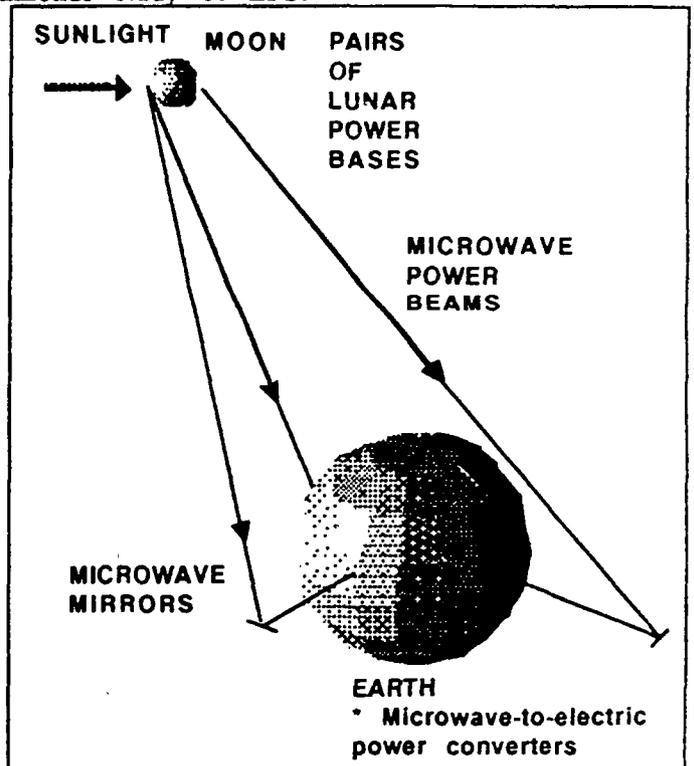
INTERNATIONAL LUNAR BASE AND LUNAR-BASED POWER SYSTEM TO SUPPLY EARTH WITH ELECTRIC POWER

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Abstract

The people of Earth will need more than 20,000 billion watts (GWe) of electric power by 2050 for a high level of prosperity. Power needs in the 22nd Century could exceed 100,000 GWe. The Lunar Power System (LPS) can provide solar electric power to Earth at less cost than conventional terrestrial systems and with far less environmental impact.

A manned International Lunar Base (ILB) can accelerate development of LPS by:
• providing the initial transportation and habitation facilities that will greatly reduce up front costs and risks;
• demonstrating the emplacement over a 5 to 10 year period of a moderate scale LPS (1-100 GWe);
• enabling early exploration of alternative LPS designs, emplacement methods, maintenance, and in-situ manufacturing of implementation equipment.

LPS can support the establishment of an ILB by:
• substantially increasing the net wealth of the world and enabling general prosperity;
• providing wider support and greater funding of operations beyond Earth than for purely scientific research;
• accelerating the development of resources in cis-lunar space and on the moon.

An international LPS program can foster world trust that lunar resources are being developed for the greatest good of mankind. The costs of SPS and LPS are compared. The organization of an international program for LPS is outlined.

Need for Solar Electric Power From Space

Figure 1 displays the two extreme power options for the world. The top curve depicts our world as it is presently dependent on thermal sources of power derived from the resources of Earth (Edmonds and Reilly 1985, DoE 1991, Holdren 1990). Notice that the world has really just started to make intensive use of its non-renewable resources for thermal energy.
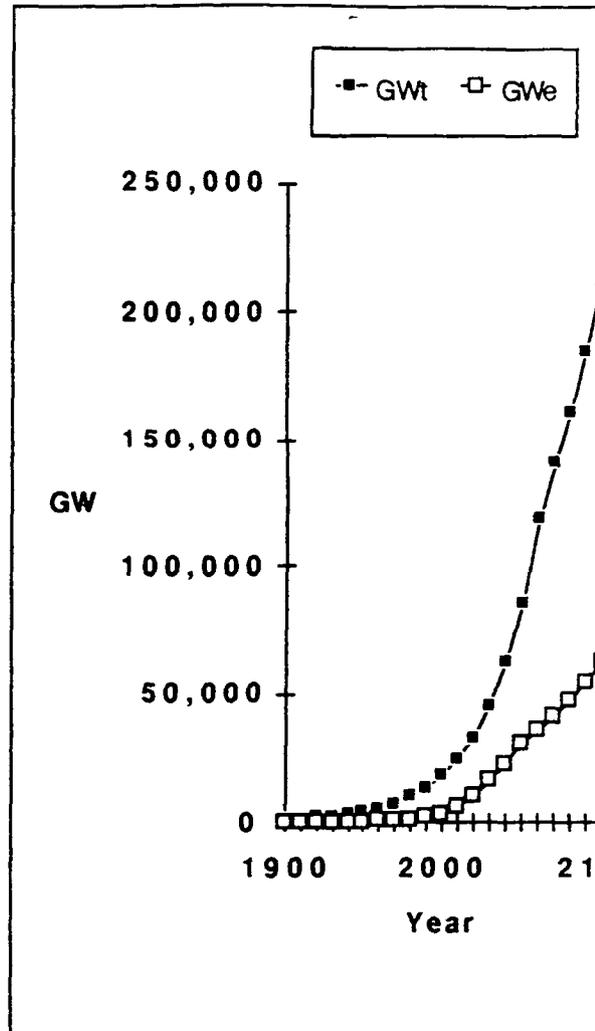

Figure 1. Growth of Global Power System

Since the start of the industrial revolutio total world use of industrial energy, primari thermal, has grown at approximately 3.6 %/ In Figure 1 this rate of growth is assumed to continue until the world per capita productic power equals 10 kWt/person at the middle of 21st century. The 4 billion people of the developing countries now use less than 0.7 kWt/person. Increasing per capita use of e is the driving function. Population growth i secondary factor. The world population was assumed to grow at 0.9%/Yr from 1900 to 195

.4%/Yr afterwards.

By 2100 the total quantity of thermal energy in this model will fully deplete the known ~tory (10^7 GWt-Y) of all non-renewable ~es on Earth except for deuterium and ~gen for use in proposed fusion reactors.

~able 1 summarizes the labor, capital, and mass ~ower plants required to produce 1 GWe-Y of ~gy from present-day power plants (DeLaquill ~. 1988; DoE 1980a; DoE 1980b; DoE 1988; Martin ~etta 1984). The terrestrial thermal solar ~er (TTSP) and terrestrial photovoltaic solar ~er (TPSP) systems are scaled up by a factor of ~ This simulates their use as providers of base ~ power rather than for power in only the ~noon and early evening.

In addition, to produce 1 GWe-Y of energy a ~il fuel plant must burn approximately ~0,000 tons of coal. This costs 80 to 190 M$. The ~on plant must consume approximately 200 ~ of yellow-cake at a cost of 12 to 48 M$. ~tment of wastes from fossil and nuclear ~ms adds significantly to the fuel and capital ~nses.

~ver ~nts	Labor Work-Y	Capital $10^6$$	Plant Mass tons	Net Energy
~sil	260	200	10,000	3 to 4
~ion	800	250	41,000	3.3
~P	1,500	470	314,000	11.5
~P	3,100	760	434,000	1.4
~	<20(Earth) <1 (moon)	20	5,200	90(rect) 200(moon)

Table 1. Generation of 1 GWe-Y of Energy

It is unlikely that a terrestrial solar power ~em (TPS) can be designed to be the major ~lier of power to Earth. On average a ~ldwide TPS incorporating advanced ~nology will provide to end users less than 20 ~ per m^2 of collector area. In addition, ~ensive secondary facilities require storage of ~terminately immense quantities of energy ~ - 1,000s GWe-Y) and the worldwide ~stribution of that power. Table 1 does not ~ude the costs of these major elements of a ~etary power system based on TTSP or TPSP ~swell 1991).

~The Net Energy column refers to the lifetime ~ for the respective power plants. This is the ~gral, over the life of the plant, of the annual ~ energy output divided by the sum of the ~ual external energy inputs. The energy of ~fossil or nuclear fuels is not included. The

inputs include externally provided operating energy such as the oil to power a coal train or the energy to refine uranium ore into yellow-cake. It includes the energy tapped from the primary fuel to operate the plant and the energy inputs to build the plant and its fuel supply systems. TTSP, TPSP, and LPS bring net new quality energy to Earth. The larger the net energy ratio the more energy one gets out of the system for the energy necessary to build and maintain it. Fossil and nuclear fission plants decrease the non-renewable energy stores of Earth.

The bottom curve in Figure 1 provides the functionally equivalent level of electric power to the thermal energy of the top curve. If the assumed population and energy utilization scenarios continue as expected a transition from terrestrial to space solar power must occur between 2000 and 2050.

Lunar Power System

In 1989 a NASA sponsored task force concluded that the moon has a vital role to play in supplying electric power to Earth in the 21st century. A commission of the Office of the President of the United States has recommended study of the use of lunar resources to provide power to Earth (Stafford 1991). One of the options presented in both reports is the establishment of solar power bases on the moon to beam electric power to Earth. Criswell and Waldron (1990, 1991a) originated the LPS concept. These recent studies indicate that LPS can supply all the electric power needs of Earth by the year 2050 (>20,000 GWe) and grow to meet greater demands.

After a demonstration-LPS is built, all the costs of expanding LPS can be borne by profits from the sale of power from the moon. The LPS row in Table 1 indicates that the mature system will have low capital and labor costs. LPS can provide an internal rate of return that exceeds 30% per year. This can occur within 10 years of the start of construction on the moon. Net profits the order of 15,000 B$/Yr are reasonable to expect if 20,000 GWe is sold at 0.1 $/kWe-H. Preliminary economic models indicate that LPS will have a positive impact on the world economy. LPS can provide a stable growth of power and stablize the cost of energy (Thompson and Criswell 1991).

Several options for LPS architecture minimize deep space operations and orbital components.

The basic LPS includes pairs of solar power stations that beam power directly to rectennas on Earth during the time those rectennas can view the moon. Power storage on the Earth or on the moon can provide continuous power output on Earth when the moon is not in view (<16 hours/day) or when the moon is in eclipse (<3 hours).

Figure 2 illustrates a more advanced system that includes microwave mirrors in orbit about Earth. This system would continuously supply load-following power to rectennas on Earth except during the three day period around new moon. The microwave reflectors, at a given intensity of the microwave beams, would allow a factor of three reduction in the size of rectennas required to power a region on Earth. However, approximately three days of power storage would be required on Earth or on the moon during the period of new moon when bases on both limbs are in lunar night.
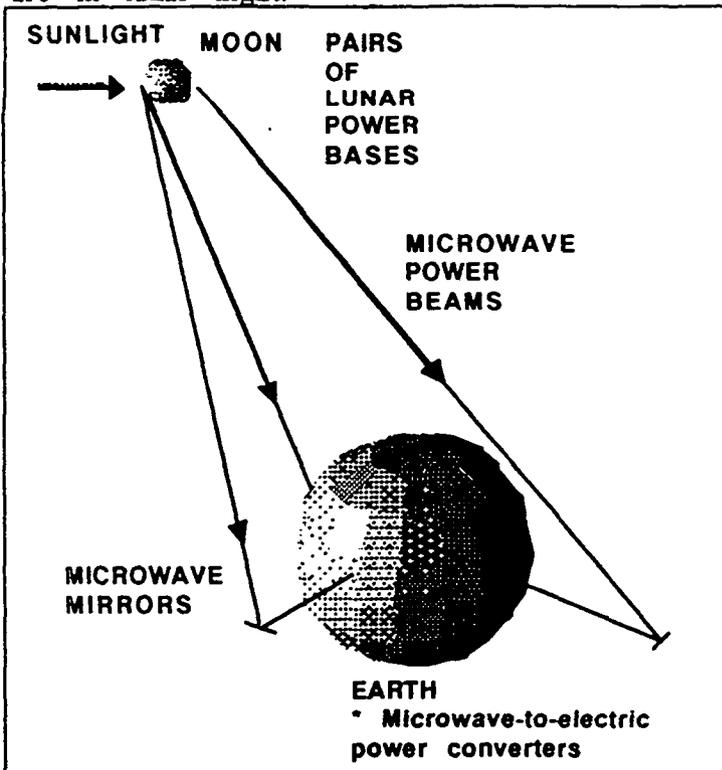
Figure 2. Lunar Power System Schematic

It is preferable to minimize the use of costly power storage. Microwave mirrors in orbit about Earth can minimize power storage. The duration of power storage is also reduced by increasing the fraction of the lunar month that each power station is sunlit. Additional solar power conversion units could be constructed across the lunar limb from their respective Earthward transmitting stations. Each set of cross-limb

arrays provides electric power during new and for three-quarters of the lunar month (Waldron and Criswell 1991).

LPS can be augmented by placing solar reflecting mirrors (i.e., solar sails) in polar about the moon. These mirrors illuminate t lunar bases during new moon, during an e and when a base is deep in its night cycle. sails would also augment the solar flux to th power stations during surface daytime. The sails operate as "light-buckets" that simply all of their sunlight into a section of the clo lunar power base. They do not have to imag sun or be continuously boresighted. The m LPS would likely include all the above elem

Figure 3 is a schematic representation of of the LPS limb bases indicated in Figure 2. #1 is the 10 to 100 km diameter aperture as from the Earth. That aperture is composed o many stand -alone power plots. The power occupy an elliptic area on the moon that is located Earthward of the terminator as seen Earth. View 2 shows a string of the power p This string extends from just Earthward of t lunar limb (top) along a line directed towar Earth. This string includes the "black" plot view #1.

View #3 shows the primary components typical power plot. Sunlight collected by so converters (a) is changed to electricity. The electric power is collected by subsurface wi and provided to many solid state microwave integrated circuit converters (MICCs). Each (b) sends an individually controlled signal t microwave reflector grid (c) at the opposite of the power plot. That signal is reflected t Earth as the sub-beam (d) contributed by th power plot. A set of MICCs, one MICC per po plot, in the thousands of power plots in view acts to form a beam. The 100s to 1,000s of MIC positioned before each microwave reflector can form 100s to 1,000s of individual beams. the beams radiate out from the same segmen antenna shown in view #1, but each of the can be directed to a different rectenna on E

Each LPS beam from a 40 to 100 km diam base is fully controlled in intensity, to a scale few 100 meters, across its cross-sectional are Earth. Control of the phase and amplitude of each of the subtransmitters that contributes energy to a given beam produces the desire amplitude distribution at Earth.

Figure 4 depicts the operations needed to construct a lunar power plot (De Generes an Criswell 1983). Several tractors smooth the surface, extract fine-grained iron, and bury
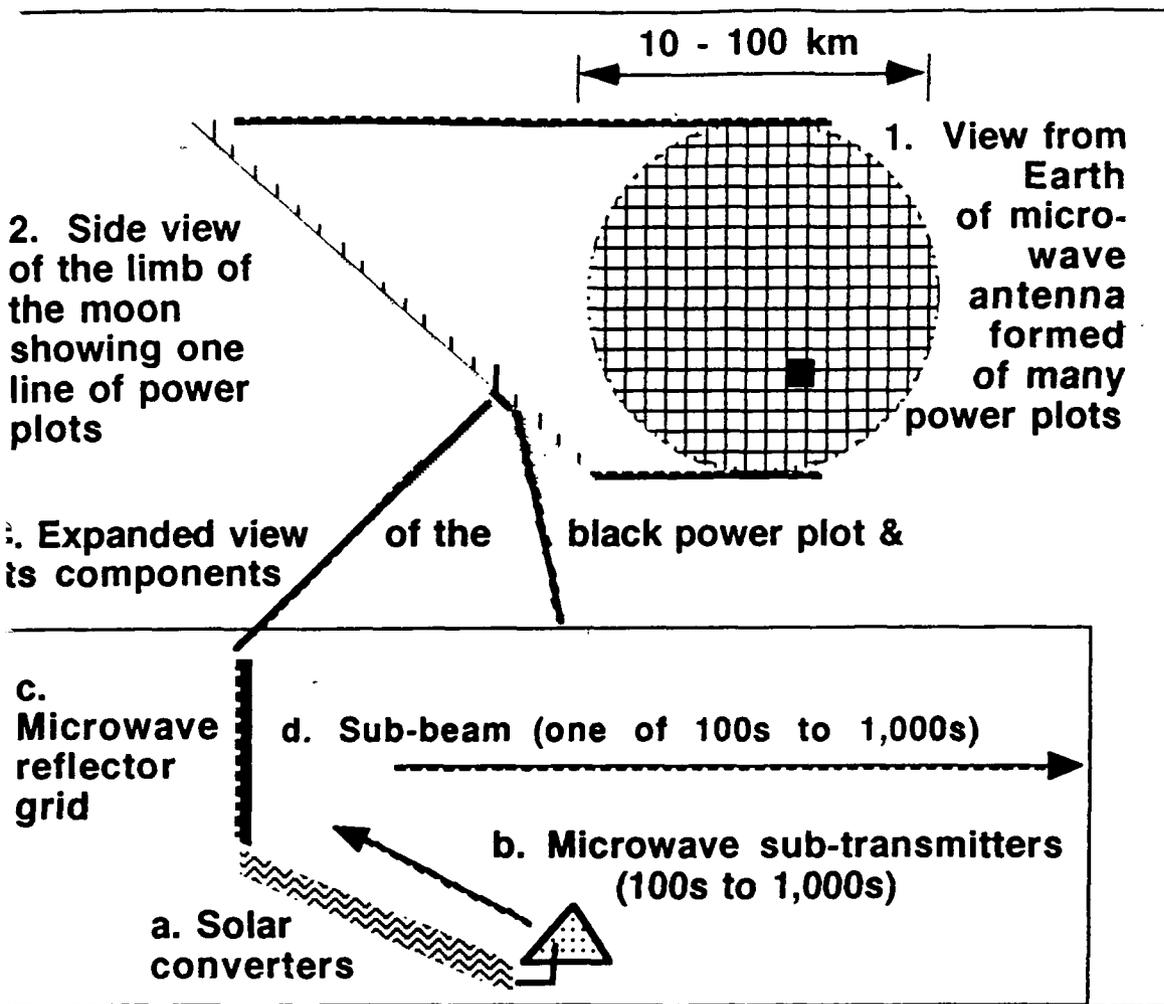
10 - 100 km

1. View from Earth of micro-wave antenna formed of many power plots

2. Side view of the limb of the moon showing one line of power plots

?. Expanded view ts components of the black power plot &

c. Microwave reflector grid

d. Sub-beam (one of 100s to 1,000s)

b. Microwave sub-transmitters (100s to 1,000s)

a. Solar converters

Figure 3. LPS Antenna, Power Plots & Components

NASA (1980, 1979) SPS studies. The extensive advances in the technology of power conversion and power beaming will be taken into account (Battelle 1991). One to two years of these studies and the laboratory studies that are done in parallel can provide adequate options for field demonstrations.

Early Steps in LPS Development

At this stage LPS faces three primary challenges. The first is to demonstrate on Earth the engineering and economic feasibility of the critical components and systems of production. The second is to reduce the up-front costs. The third is to show that LPS is acceptable to billions of potential users on Earth. The International Lunar Base is relevant to all three challenges.

Systems Studies

There is an immediate need for more extensive conceptual design studies of LPS and alternatives to the LPS-reference system described in this paper. LPS is different from all other major aerospace and power systems. The primary systems integration that forms the beams occurs in free space between the moon and the Earth. The electromagnetic fields from the thousands of power plots of a given power base sum up to produce the various synthetic beams. Each of the contributing microwave sources must be accurately phased and controlled in amplitude. This is primarily a time-base and ephemeris problem and is well within the capabilities of modern electronics.

power collection. They also lay down glass ets under which are layered thin films of ar converters. Thin films of moderate iversion efficiency, 5 - 10%, are adequate. In foreground is a mobile glass processor that lts lunar soil to produce foamed glass supports, erglass, and glass sheets. The supports and erglass are used to make the microwave lectors. One reflector is being erected. Solar ctric power is provided to sets of microwave -transmitters that are buried under the mound the Earthward end of each power plot. Note t the Earth remains in the same general sition in the sky at a given base. The fleet of atively small and independent machines move m one construction area to another. The rate installation of new power is proportional to the mber of machines and their productivity. The conceptual design studies for LPS should done as part of an engineering systems aluation and accompanied by life-cycle sts/benefits analyses. These studies should ke maximum use of the DoE (1980c, 1981) and

Figure 4 Artist's Concept of the Construction of a Demonstration Lunar Power Ba

he moon provides the platform that
rates the physical systems. The minimum
for integration of large-scale physical
ms has profound implications for the
leering and economics of LPS. The power
and the machines that build them do not
to be extensively matched, as do for example,
tiles of the space shuttle. Many different
ns can be explored, many different sets of
components tried, and many different
ods of production employed.

ratory & Field Manufacturing

olumn 5 of Table 3, which will be discussed
provides R&D priorities for the lunar
ms. First, extensive laboratory work is
ed on thin-film solar cells that can be readily
ced on the moon using local resources.
ext, the microwave sub-reflectors (Figure 3)
to be formed and the wire to collect power
ced. The design of these systems is coupled
the design and demonstration of prototype
rials handling and the manufacture of
ment. The objective is to use equipment that
a relatively low mass per unit of output [Tons-
ment/ (Tons-output/Hour)]. The equipment
d require little make-up mass or components
Earth, be highly automated, and be
rable on the moon.

to the moon with equipment that can immediately
begin to emplace a demonstration level LPS. In
the early to mid-1960s, extensive testing of lunar
equipment was done in simulation facilities on
Earth. This was prior to landing on the moon.
Since then a vast accumulation of knowledge of
lunar materials and lunar conditions has been
accumulated.

Adequate simulations of a lunar construction
site can be done on Earth. The most promising
laboratory and bench studies will be incorporated
into sets of autonomous, mobile production
machinery for demonstrations on Earth. The
production equipment can be rover units of the
general nature depicted in Figures 4 and 5.

Figure 5 illustrates three emplacer units
traveling past several power plots to another
emplacement area (Mortenson and Saul 1991). A
set of emplacer units should be be transportable
by a class of aircraft with the cargo capacity of
the U.S. STS or the Soviet Shuttle (< 30 tons
payload). A C-130 is a good analog.

The set of prototype production equipment is
air-lifted to a high desert area. The plane lands
and the prototype units are driven out under
automatic or remote control. The production
units then go to a succession of sites to build
power plots. Each site represents a different
lunar terrain and soil type.

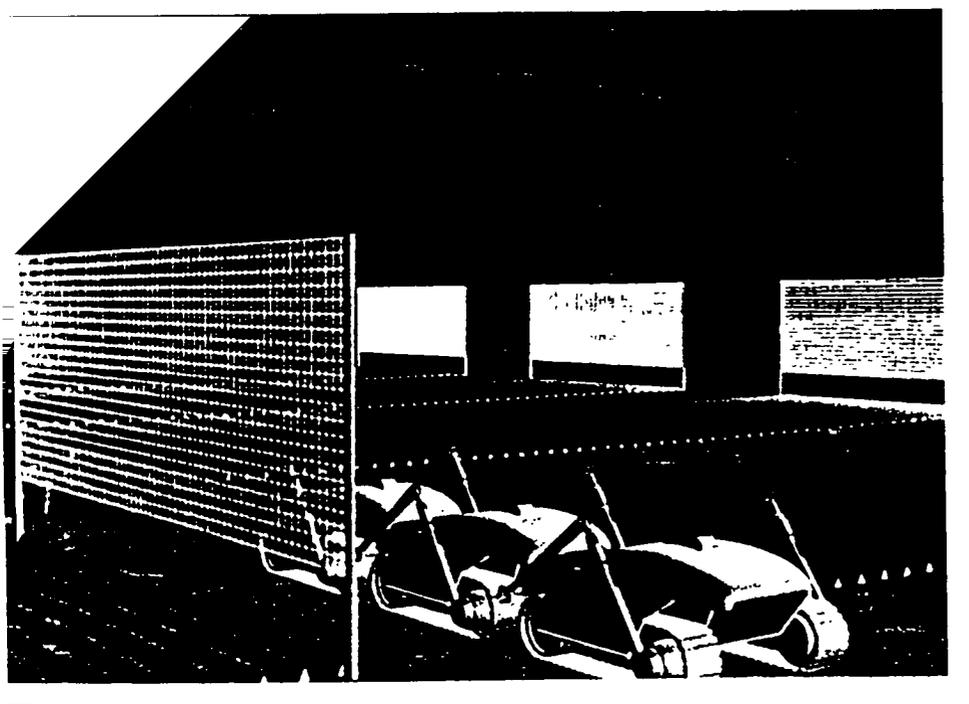The sites are created in a set of five inflatable
buildings established in the high-desert area. Four of the buildings are located along the perimeter of an elliptical area 10 to 100 km in diameter, and the fifth is located near the center. The roof of each building is transparent to sunlight and 10 cm microwaves. The floor of each building is covered to a depth of one to two meters with simulated lunar soil and rocks. Highland (aluminum-rich) and mare (iron-rich) areas are simulated. The buildings are pressurized with an inert gas, and entry ways are provided for the robotic construction equipment.

The production equipment is designed for autonomous operation in routine production. However, remote control is provided for exception-handling and non-routine

Figure 5. Emplacement Rovers in Demo Power Plot

ggressive design efforts and engineering of
able production systems will allow a return

operations. Machine and human repair of unusual maintenance is allowed.

Each building houses a fully operational power plot. During the day time the power from one plot is beamed to nearby ground and airborne receivers. The plots are phased together to demonstrate beaming of very low-level power to satellites, to distant receivers on Earth via orbital reflectors, or to a set of lunar landers (next section).

This exercise requires compromises; for example, solar cell production may take place in a mobile vacuum chamber temporarily placed on the demonstration plot. Designs must accommodate the operation of equipment in an inert atmosphere and air, when traveling between plots, versus the vacuum of the moon.

Power Beaming

There are no basic technical mysteries about the beaming of power by way of microwaves. The basic theory is understood and the practice is well within the state of the art of electronics. Consider for example, that all radar sets use power beaming, either fixed by their physical optics or controlled through phasing of their individual sub-radiators. Routine, long-term phasing of very large microwave power systems has been demonstrated at the thee kilometer long Stanford Linear Accelerator since 1968 (NRC 1981, p. 21). Microwave technology is well developed. However, the application of the technology to beam at realistic power levels for reasonable periods of time must be demonstrated to the satisfaction of both the general population and the scientific community.

Demonstrations of beam control and beam power can be done separately. The demonstration of control of moon-to-Earth beaming can be done at very low power levels. High power level beaming can be done from the Earth to orbit and from Earth to the moon. These demonstrations serve several purposes. The moon will be confirmed as an adequate platform for a large synthetic array, and several different methods of phasing the lunar array will be exercised. The effects of the atmosphere and ionosphere on high power density beams can be examined.

The lunar demonstrations can be done by soft landing a set of unmanned vehicles on the moon. The lander array will operate for several years. The landers would be simple and could easily be on the moon within five years. Three to four of the landers are evenly placed along the perimeter of the site of a potential lunar power

base. The last lander is placed near the c Each lander carries a microwave transmis system, solar arrays, and battery storage th adequate for overnight operation of the transmitter. The microwave transmitters phased to send very low power but very collimated test signals to Earth. The signal normally be received and continuously m at deep space stations. However, short-du higher-power signals could _be directed to expensive receivers at any point on Earth. example, the beam could be scanned over the campus of large universities to demon beam control and localization.

The landers can also contribute to othe engineering and scientific studies. Lander be equipped with diggers to bury the trar under 10 to 30 centimeters of lunar soil. T would simulate the placement of transmitt the very constant subsurface environmen Engineering packages such as solar cell te articles and metal-coated glass fibers can a attached. Perhaps self-contained robotic (few kilograms) can be included to conduc surveys for several hundred meters about lander.

The set of landers can also receive ver power test signals from Earth. The atmos and ionosphere of Earth create the greates disturbances of microwave beams. Beams primarily absorbed (few percent or less). secondary effect is to sporadically self-foc fraction of the beam energy into a new d The deflection effects will be much larger signals sent from the Earth to the moon th the reverse situation. Large radar system: as those associated with the early warning and ICBM tracking can be adapted to this function. These radars can provide beams wide range of power and frequency to es the full response of the atmosphere and ionosphere.

LPS provides continuous, load-followir power to a rectenna on Earth by reflectin, power beam from a succession of microwa mirrors in orbit about Earth (Figure 2). T mirrors of >100 meter diameter can be pla LEO from the Space Shuttle or unmanned The Earth-based radar systems can direct power test beams to these microwave mir The beams can then be reflected and redir local test receivers or to receivers thousan miles away. The mirrors are low-mass bu area devices that can be readily derived f existing NASA work on large space struct high-gain antennas. Given adequate pric

i meter reflectors could be in orbit within
e to five years. Ground-based transmitters of
) and NSF (e.g., Arecibo) and the microwave
rors can completely test microwave beaming
full power levels. Such tests could be rapidly
/eloped and initiated.

Space Station Freedom can support R&D
/elopment of the larger orbital reflectors.
portant tasks include verification of surface
erances, demonstration of assembly and
intenance procedures, and accelerated aging
key components.

)duction and Resources

The percentage distribution of costs in column
of Table 3 suggests additional activities at an
3 to reduce the costs of LPS. ILB can provide
iustrial laboratories for the development and
monstration of better means of production.
ose laboratories can also make all or large
rtions of future production systems out of lunar
terials. Both advances will significantly scale
wn the transportation system (HLLV, other)
d space construction. These will decrease the
:tor of 0.0025 for this part of the model. On the
ier hand there may be an increase in the
mber of research personnel and habitats.

Lunar resources can be developed for direct
e in space transportation ("other"), logistics,
d habitat construction. The factor of 0.0025
ould be sharply reduced.

SSF can provide the manned components for
'S logistics facilities in LEO and LLO and
celerate the testing of lunar base facilities for
bitation and repair activities.

Demonstration LPS & the ILB

An ILB can greatly advance the development
LPS by emplacing the demonstration system
AA 1990). An ILB program can significantly
duce the up-front cost of the demonstration LPS
d the full-scale program by providing most of
e initial transportation, habitation, and
frastructure.

Table 2 provides estimates of the total costs
$(90)) of lunar bases scaled to permanent
pulations of 30 (Case 1: 60 B$), 85 (Case 2: 91 B$),
d 300 (case 3: 243 B$) people. These estimates
sume the base is operated for 10 years. The
timates include R&D for facilities and
uipment (line 1.c) and transportation (line 1.d).
tablishment and operation of the flight systems
ne 2) are also included.

The purpose of these bases is to demonstrate

the emplacement of 1, 10, or 100 GWe of power at
the end of the ten-year period. The incremental
cost of creating the production machinery for
emplacing the LPS components is given in lines
la and b. The additional R&D cost for the LPS
production machinery increases from 12 B$ (Case
1: 1 GWe) to 22 B$ (Case 3: 100 GWe).

Several costs, such as transportation, do not
scale linearly to lower rates of power
emplacement. Case #3 is closest to the SPS
production modeled by General Dynamics. The
estimates are extrapolated from a study of using
lunar materials to emplace one 10 GWe SPS every
year (Bock 1979). Thus, the cost estimates of the
smaller bases should be treated as preliminary.
Consider these values primarily as
encouragement to deeper analysis. NASA (1991) is
studying this size of vehicle but for considerably
lower launch rates. Those studies could be
immediately extended to the launch rates implied
by Table 2 and to the use of lunar materials to
provide propellants.

Cases	# 1	# 2	# 3
GWe installed over 10 Years	1	10	100
GWe-Yrs of energy	5	50	500
Gross Revenue (B$) (@0.1$/kWe-H)	4.4	44	438
Net Revenue (B$)	-56	-47	195
Total Costs (B$) (sum 1+2+3)	60	91	243
1. R&D (B$) (sum a+b+c+d)	42	51	86
a. LPS Hrdw	11	11	11
b. CNSRT. SYST	1	3	11
c. FACILITIES & EQ	5	10	30
d. TRANSPORT	26	27	35
2. Space & Ops (B$)	17	34	103
3. Rectenna (B$)	0.6	6	55
$/kWe-H	1.4	0.2	0.06
Moon (tons)	2,300	6,200	22,000
Space (tons)	970	2,700	9,700
People (moon, LLO, & LEO)	30	85	300

Table 2   Parameters of Smaller Bases

Costs of Space Equipment and Operations
increase sharply between 10 and 100 GWe of final
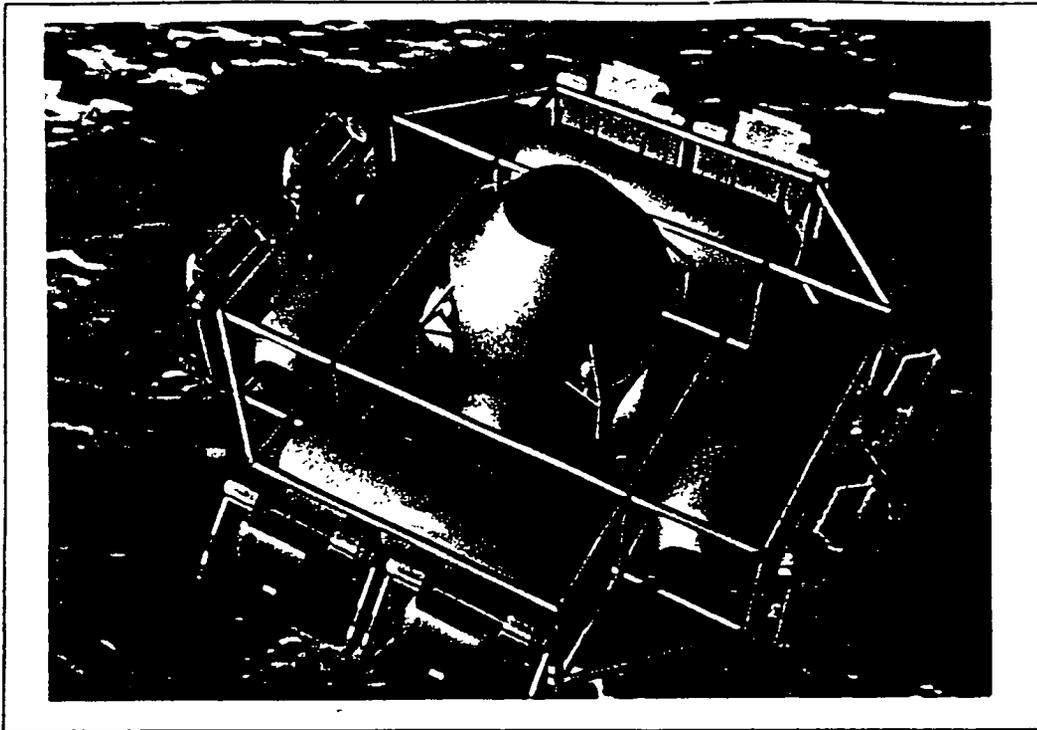installed capacity. The cost of power drops

Figure 6.   Lunar Lander with Eight Emplacement Rovers

sharply as the installed power increases.

Power from the demonstration base is sold to Earth. The sale price is assumed to be 0.1 $/kWe-H. At some time between the installation of 10 and 100 GWe, the integral of the net cash flow from the sale of power goes positive. In this model the net expenditures for all aspects of the lunar base and the LPS will be less than 100 B$ by the time positive cash flow begins. Power could sell for much greater prices to customers off Earth.

Figure 6 illustrates a lunar vehicle that is scaled to land 30 tons of equipment on the moon (Mortensen and Saul 1991). The vehicle carries eight emplacement vehicles. Assumming the base described in Case 1 is emplaced over a period of three years then approximately 10 landing operations would be required per year. This tonnage and flight rate are consistent with with studies of larger-scale lunar bases considered by NASA (1989b) in the 90-day report.

Costs and Payoff

The cost of power from the mature LPS can be very low, the order of 0.001s $(1990)/kWe-H, assuming a beam intensity at the rectenna of 23 milliwatts per cm^2. Rectennas on Earth are the major cost elements (Tables 2 and 3). The cost of power will decrease as the cost of rectenna construction decreases or beam intensity increases. LPS enables the profitable operation of

small rectennas of on few hundred meters diameter with 10s MW output. Rectenna enlargement can be financed from profits finished portions. LP$ economically robust major increases in construction and maintenance costs. L appears to be competi costs and environmer considerations agains conventional power s

NASA, DoE, NRC, and L Models

The early studies of were scaled to buildin 10 GWe or two 5 GWe satellites and compan rectennas per year ov thirty year period (G 1977, NRC 1981). The fleet of 60 satellites, v peak capacity of 300 GWe, would feed approximately 9,000 GWe-Y of energy to Ea over 60 years. The order of 2,100,000 tons o satellites and supplies would be transported space over the period of construction and operation. This assumes each satellite has a of 35,000 tons and that 1% of its mass is rep over the period of operation.

The costs of major components and ope per GWe-Y are in the top part of column #1 Table 3. "Solar array" refers primarily to t solar cells. SPS structure, microwave gene and rotary joints constitute the "Other port Crew habitats, construction facilities in LEO GEO, and maintenance equipment are inclue "Other (habs, etc.)." The heavy-lift launch vehicles are the HLLVs. "Other" transporta elements include E-LEO and LEO-GEO persor vehicles and ion-drive engines to transport components from LEO to GEO. The nominal the electricity to emplace the fleet is predic be 88.1 M$/GWe-Y or 0.01 $/kWe-H. The distribution of these costs as percentage of Capital Total cost is in column #1a.

These engineering costs do not count the value of money required to establish the fu system. The bottom section of column 1 sho that the cost of financing the SPS dominates cost of power. NASA and the National Rese Council (1981, p. 37) adapted the "compound

an annuity" to evaluate cost recovery of the
ney required to finance SPS (Copeland and
:sson 1979). The "Capital Recovery Factor"
RF) in Eq. 1 is modified for the GWe-Y basis of
:ting used in Table 3.

. 1.   CRF = Years*R/[1-(1/(1+R))^{Years}]

Equation 1, R is the Rate of Return (= 15% in
. SPS example) and Years is the operating life
an SPS-rectenna set.
Multiplying CRF = 4.57 times the Capital Total (=
.1 M$/GWe-Y) yields the Capital Recovery = 447
:/GWe-Y. The time value of money to build the
S fleet dominates the cost of power from the SPS
et.   Note that the product of "Capital
tal*Years" in Eq. 1 is the "Present Value of an
iuity."   Capital Recovery in Table 3 is the
ount of the periodic annual payment of the
iuity.   The Capital Total in Table 3 is slightly
:her than the NASA and NRC estimate because
includes the RDT&E.
To obtain the full cost of SPS power, NASA
led an estimated cost of 5.2 mills per kWe-H or
M$(77)/GWe-Y for maintaining the SPS fleet
l rectennas.   The sum of Capital Recovery and
intenance yields 493 M$(77)/GWe-Y or
)roximately 0.056 $(77)/kWe-H.   Current cost of
S energy, 0.102 $(90)/kWe-H, is obtained by
:ltiplying by 1.7.   This cost is approximately the
ce of wall-plug electricity in the United States.
The National Research Council (1981)
.intained that almost all the SPS costs would be
:her than the NASA estimates in column 1.
lumn 2 contains the multipliers.   They project
it high efficiency, single crystal silicon solar
.ls will be 50 times more expensive and
:refore will be the dominant cost factor (82.5%
column 3a).   Transportation from Earth to orbit
a factor of three higher.   However, rectenna
)duction and operation are unchanged.   Notice
it rectennas dropped to 2.1% of the costs,
lumn 3a, from 16.7% in the NASA estimate,
lumn 1a.   The Capital Total in column 3
:reased by a factor of 7.8 to 687 M$(77)/GWe-Y.
The NRC made no changes in the financial
sumptions of NASA but did indicate that the
certainties were larger than indicated by
\SA.   The Capital Recovery Factor of 4.57 was
.ained.   The Total Energy Cost thus rose to 3,550
$(77)/GWe-Y.   This is the same as 0.4 $(77)/kWe-
or 0.7 $(90)/kWe-H.   This cost is approximately 7
ies the price of wall plug-electric power in the
.ited States.

Qualitative Costing of LPS

The NASA and NRC estimates of SPS cost can be
used to provide a better understanding of the
fundamental differences between deploying SPS
from the Earth and sending equipment to the
moon to make the components of the LPS system
from local materials.   These differences include
maximum potential power, manufacturing versus
deploying, efficiency of rectenna illumination,
and financing the growth of SPS versus LPS.
First, consider RDT&E and the energy yield.
The reference-SPS is scaled to provide 300 GWe.
Each satellite would operate for 30 years.   Thus,
the reference-SPS fleet would yield 9,000 GWe-Y
of energy.   LPS has been modeled for growth to
20,000 GWe.   Averaging over 40 years of build-up
and 30 years of full operation, the LPS would yield
1,000,000 GWe-Y (= 20,000 GWe * 50 Y).   To a first
approximation, the cost of RDT&E per unit of
energy can be scaled to the respective total
energy output of each system.   This LPS/SPS ratio
is 0.009 (=9,000/1,000,000).   This ratio, in column
4, is multiplied against the NRC costs for RDT&E
per unit of energy output in column 3.
Multiplication yields the RTD&E cost of 100,000
$/GWe-Y for LPS that is shown in column 5.
Rescaling SPS to a greater energy output would
similarly reduce the RDT&E for SPS.   However, it is
doubtful that even 300 GWe of SPS could be
deployed from Earth because of environmental
restrictions on launch operations.
Next, consider SPS deployment from Earth
versus manufacturing LPS on the moon.   The
figure of merit is tonnage shipped from Earth per
GWe-Y of energy returned to Earth.   The 5 GWe
SPS is estimated to weight 35,000 tons.   Sixty would
be deployed.   We arbitrarily estimate that 1% of
the mass of the SPS fleet is added for components,
fluids for station keeping and orientation, and
transportation propellant.   The projected SPS
mass-to-energy ratio is 230 tons/GWe-Y (=
2,100,000 tons/9,000 GWe-Y).
Detailed estimates are available for the 592,000
tons of equipment, facilities, and components that
must be taken from Earth to emplace a 20,000 GWe
LPS (Criswell and Waldron 1990).   Thus, the LPS
mass-to-energy ratio is 0.59.   The combined
LPS/SPS ratio is 0.0025.
Note the seven items in column 4 that are
scaled by the LPS/SPS mass to the energy ratio.
This ratio will not change greatly with total
energy.   However, it will vary with the level of
technology used to implement SPS and to do
manufacturing on the moon.   It seems likely that
SPS and LPS components, as opposed to the LPS

production system, can converge to similar mass-to-energy ratios. Thus, LPS will always have a relative advantage in terms of increasing efficiency of machines that make the LPS components on the moon.

The dominant effect of solar cell cost is reduced from 82.5% in column 3a, to 18.7% column 5a. The relative cost of solar cells in column 5a may be estimated too high. LPS does not need the high-efficiency solar cells that the NRC assumed could only be obtained from relatively thick, single-crystal silicon cells. Rather, LPS is compatible with thin-film cells, 5% - 10% conversion efficiency, that use very small quantities of photoconverter. Thin-film cells based on amorphous silicon, polycrystalline silicon, GaAlAs, and several other active lay can achieve this level of efficiency. If nec the active layers can be brought from Earth little effect on costs.

LPS costs for emplacing a unit of power continuously drop. This is because of the accumulation of industrial learning in the construction of LPS components, the increa use of production machinery made from lur materials, and the use of lunar materials and power in logistics. None of these positive fa are considered in Table 3. In large-scale production, the LPS should have a unit cost c power that is primarily dominated by the cos

TYPE OF SYSTEM (across) CAPITAL ITEMS (below)	(1) NASA M$(77) per GWe-Y	(1a) NASA % of Total Costs	(2) NRC Cost Multi-liers	(3) NRC M$(77) per GWe-Y	(3a) NRC % of Total Costs	(4) Ratios LPS/SPS	(5) LPS M$(77) per GWe-Yper (3)*(4)	(5a) LPS % of Total Costs
RDT&E(& 1st 5 GWe set)	11.39	12.9%	1.4	15.9	2.3%	0.009	0.1	1.9
SPS portions								
Solar array	11.3	12.9%	50	566.7	82.5%	0.0025	1.4	18.7
Other portions	15.3	17.4%	1.4	21.5	3.1%	0.0025	0.1	0.7
Other (habs, etc)	2.0	2.3%	1.4	2.8	0.4%	0.0025	0.01	0.1
Space Construction	6.7	7.6%	1.4	9.3	1.4%	0.0025	0.02	0.3
Space Transportation								
HLLV (Earth to orbit)	13.0	14.8%	3	39.0	5.7%	0.0025	0.1	1.3
Other (LEO-out)	5.7	6.4%	1	5.7	0.8%	0.0025	0.01	0.2
Management & Integration	8.0	9.1%	1.4	11.2	1.6%	0.0025	0.03	0.4
Rectenna (Earth)	14.7	16.7%	1	14.7	2.1%	0.4	5.9	76.4
Capital Total (M$/GWe-Y)	88.1	100.0%		686.7	100.0%		7.7	100.0

Cost of electricity: (Financing Impact)	Rate return= 15.0% Plant Life(Yrs)= 30.0		Cpt Fac.= 90.0%			LPS Rate Return= 15.0% LPS Plant Life(Yr)= 30.0		LPS Cr Fac.= 90.0
Capital Recovery Factor*Yrs	4.57			4.57			4.57	
Capital Recovery (M$(77)/GWe-Y)	447			3,486			39	
Maintenance (M$(77)/GWe-Y)	46		1.4	64		0.31	19.55	
Total Energy Cost (M$(77)/GWeY)	493			3,550		(Wt. avg)	59	
$(77)/kWe-H	0.0562			0.4050			0.0067	

Table 3. Summary of Cost Studies: NASA and DoE, NRC, and LPS

ructing and operating the rectennas on
h.
he NRC (1981) considered rectennas to be
understood technology.   No adjustments were
. in column 2 to the projected costs of
nna construction and operation.
.PS has a considerable advantage over the
ence SPS, because rectennas are the
nant engineering cost for LPS (5.9
'7)/GWe-Y and 76.4%).   LPS uses an oversized
mission aperture at each power base on the
1.   This allows rectennas larger than several
red meters in diameter to be illuminated
. evenly than in the reference SPS and,
:fore, increases the power production per
area of rectenna.   A rectenna receiving an
power beam outputs approximately 2 to 2.5
; more power than when receiving a power
i of the same peak intensity from a reference

This greater output level produces the
:tion factor of 0.4 in column 4 and the lower
;y costs in column 5.
he factor of 0.4 assumes those microwave
ctors in orbit about Earth are used to provide
following power to the rectenna.   It also
nes that the life of the rectenna field is
rned by environmental factors such as wind,
and corrosion rather than being
ortional to the total energy received.
he Capital Recovery Factor (CRF) increases
projected cost of LPS power to 0.0067
)/kWe-H or 0.011 $(90)/kWe-H.   This cost is
below present-day electric costs.   Note in
nn 4 that the Maintenance Factor = 0.31 [=
76.4% + 0.0025*(100%-76.4%-1.9%)].   This
ely adjusts for the fractions of the
itenance going into the space and terrestrial
ems.
he Capital Total of 7.7 M$(1977)/GWe-Y
'ed from the NRC estimates in column 4
ies a total LPS-program cost of 7,700 B$(1077)
3,000 B$(1990).   The cost adjustments in
mn 4 are derived from far more detailed
els of LPS construction and operations
swell and Waldron 1990).   The detailed model,
ming a 1990s level of technology, projects the
cost of a 20,000 GWe LPS to be approximately
00 B$(1990) with 16,000 B$(1990) for rectenna
truction on Earth.   Advances in the
nologies of components and production
hinery, use of lunar resources in logistics,
building portions of the systems of
uction from lunar resources can
ificantly reduce the mass that must be sent
Earth   (tons/GWe-Y).

Questions Concerning Financial Analyses

Future work on power from space must
challenge the NRC (1981) cost model.   The 15%/Y
Rate of Return requirement, while representing
the 1970s experience with high interest rates, is
not typical of major public-related programs.   A
much sounder approach is to use the real rate of
return (RRR).   RRR is the difference between the
yield on long term, high value securities and the
rate of inflation.   Over the years, RRR = 2 to 3%/Y
is typical (R. Thompson personal
communication).   Table 4 applies a 3%/Y rate to
engineering costs in   Table 3.   Notice that the
Total Energy Cost of the NASA estimate in column
1 falls to a reasonable value, <0.04 $(1990)/kWe-H.
The NRC estimate is higher, > 0.2 $(1990)/ kWe-H,
than most electricity today.   The Capacity Factor is
increased from the 90% in Table 3 to 95%.

Direct application of the annuity formula to
the LPS, as shown in Table 4, is inappropriate.
Most of the long-term investment is in the system
of production and transportation, in space and on
the moon, that emplace power units on the moon.
The lunar and space investments constitute less
than 17% of the total.   Thus, the Capital Total in
column 5 decreases to 1.81 M$(77)/GWe-Y for the
lunar operations.   For a 3%/Y rate and a 30
investment horizon, CRF = 1.53 and the Capital
Recovery =   2.8 M$(77)/GWe-Y.   Maintenance is
already included in the lunar operations.   With
these adjustments, the cost of the lunar portion is
approximately 0.0003 $(77)/kWe-H.

A rectenna serviced by LPS has a much
shorter payback period than when serviced by
the reference SPS.   In the latter case a complete
SPS and 10 km by 20 km field of rectennas must be
installed before power is produced.   In contrast,
the oversize transmitting apertures on the moon
can send power efficiently to a rectenna only a
few hundred meters across.   The small rectenna is
built in a fraction of one year and will
immediately return a positive cash flow.   Further
expansion of the rectenna comes from current
revenue.   Using Equation 1 it is reasonable, to a
first approximation, to take the investment period
as one year, the Capital Total for the rectenna as
5.87 M$(77)/GWe-Y, and RRR = 3%/Y.   Thus, CRF =
1.03 and the Capital Recovery is 6 M$(77)/GWe-Y
or 0.0007 ·$(77)/kWe-H.

Once a field of sub-reflectors is constructed on
the moon, the installation of new capacity can
proceed incrementally.   This expansion of power
on the moon is paid for by the sale of power from
existing   rectennas.

We have assumed in Table 4 that the Capacity Factor of LPS is approximately 99%. LPS is a fully distributed, highly redundant system that more closely resembles a telephone network than a conventional central power station. In addition, LPS can average its power feed over the entire globe and is also decoupled from terrestrial feedbacks and the electromagnetic effects of solar storms.

Construction of conventional power stations (Table 1) will cost 10 to 30 times more than LPS. Terrestrial power plants will have additional, and increasing, costs for labor, fuel, compliance with environmental standards, and power storage and distribution. These costs can equal or exceed the costs of building and maintaining the power plants.

Organizing and Developing LPS

To develop LPS, three types of investors are anticipated: governments, consortia, and local organizations. Between now and 2001, government programs will likely pay for the development and initiation of the transportation elements and the lunar base. In that period, expenditures can be comparable to present United States government expenditures in aerospace products for the U. S. Department of Defense and NASA. LPS can provide a peaceful focus for the present defense- and technology-related organizations of the space-faring nations.

A national or international consortium formed to develop, procure, and implemen elements for LPS production and do the RD rectennas. After the year 2001, this cons can conduct all off-Earth operations. Betv 2001 and 2005, the consortium would begin receiving a net positive revenue from the power on Earth. More than one consortiun be formed. Many lunar bases are needed.

Rectenna R&D, both for rectennas and means of production, can involve all the of Earth. Rectennas can, as appropriate, b constructed, operated, and paid for by priv groups, cooperatives, and countries. Virtu the costs of rectenna production will be c by current cash flow. The major challeng startup costs and public confidence in LPS.

Conclusion

A vigorous Apollo-like program coul the construction of the ILB and the demo LPS on the moon within ten years. LPS wo firmly establish a permanent dual-planet economy, growing commerce between the and the moon, and world wide prosperity.

TYPE OF SYSTEM (across) CAPITAL ITEMS (below)	(1) NASA M$(77) per GWe-Y	(1a) NASA % of Total Costs	(2) NRC Cost Multi-liers	(3) NRC M$(77) per GWe-Y	(3a) NRC % of Total Costs	(4) Ratios LPS/SPS	(5) LPS M$(77) per GWe-Yper (3)*(4)	(5a) LPS % of Tota Cost
Capital Total (M$/GWe-Y)	88.1	100.0%			686.7	100.0%			100
Cost of electricity:	Real Rate Return= 3.0%		Cpt Fac.= 95.0%				LPS Real Rate 3.0% Return=	LPS Fac.-
(Financing Impact)	Plant Life(Yrs)= 30.0						Payback Time(Y)= 30.0	99
Capital Recovery Factor*Yrs	1.53			1.53			2.55	
Capital Recovery (M$(77)/GWe-Y)	142			1,106			20	
Maintenance (M$(77)/GWe-Y)	46		1.4	64		0.31	19.55	
Total Energy Cost (M$(77)/GWeY)	187			1,170		(Wt. avg.)	39	
$(77)/kWe-H	0.0214			0.1335			0.0045	

Table 4.   Alternative Financial Assumptions: NASA and DoE, NRC, and LPS

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DYNAMIC ISOTOPE POWER SYSTEMS
FOR FLO

M. E. HUNT
ROCKWELL INTERNATIONAL
ROCKETDYNE DIVISION

MAY 5 & 6 1992

CANDIDATE SYSTEMS

CONTINUOUS MOBILE/REMOTE POWER

SYSTEM	< 1 kWe	> 1 kWe
RADIOISOTOPE THERMOELECTRIC GENERATORS	X	
ISOTOPE - BRAYTON	X	X
ISOTOPE - STIRLING	X	X
PHOTOVOLTAIC/REGENERATIVE FUEL CELLS	X	X
ISOTOPE - ALKALI METAL THERMOELECTRIC CONVERTER (AMTEC)	X	X

FREE SPACE MISSIONS WITHIN APPROXIMATELY 1.5 AU WILL BE PHOTOVOLTAIC/BATTERY OR SOLAR DYNAMIC AND ARE NOT CONSIDERED WITHIN THIS STUDY. ISOTOPE SYSTEMS ARE NOT CONSIDERED CANDIDATE SYSTEMS FOR THESE APPLICATIONS.

COMPLEMENTARY DIPS AND RFC UNITS
- Roles Dependent on Time Between Recharge Opportunities-
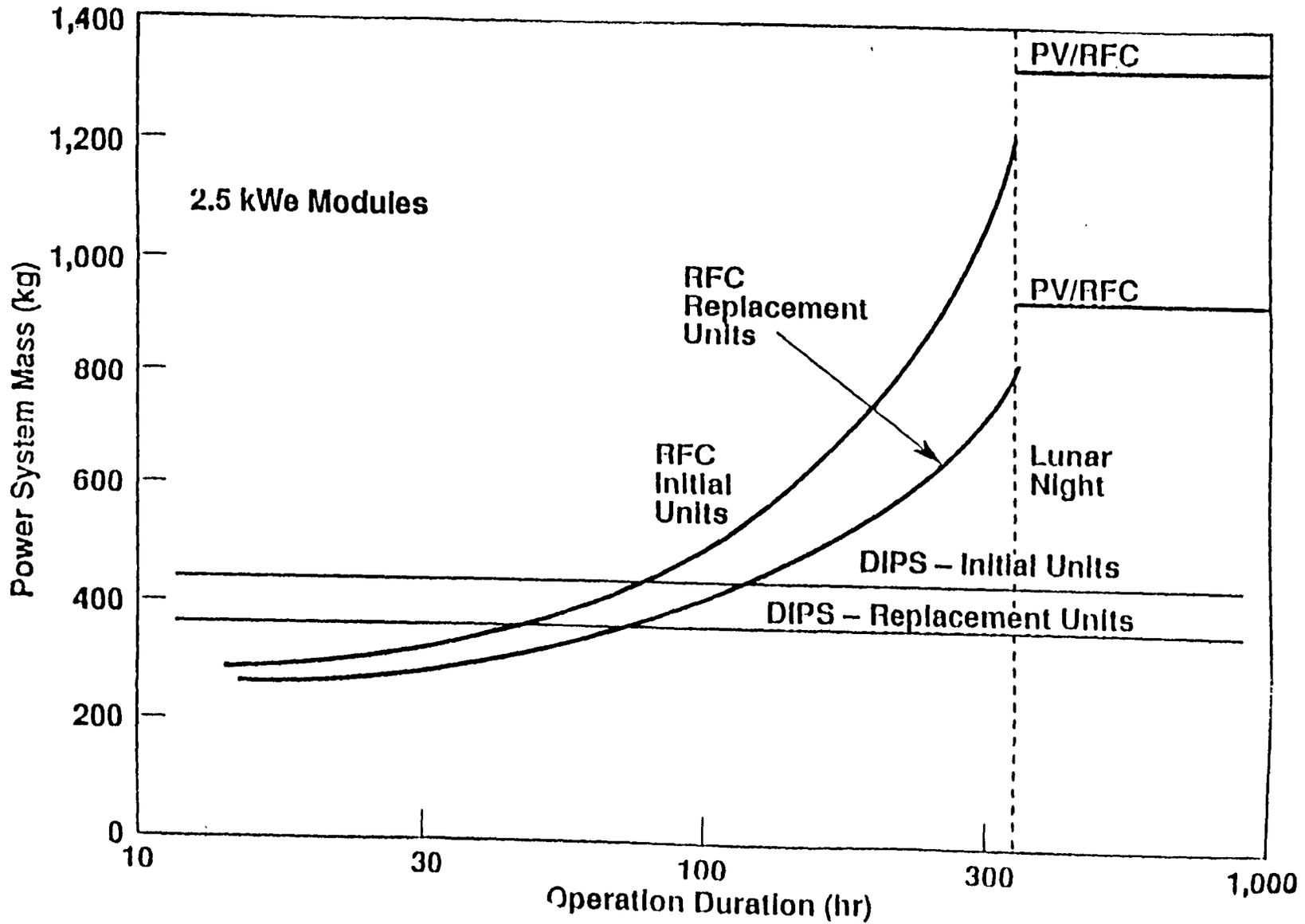
# SUMMARY OF FIRST LUNAR OUTPOST POWER SYSTEM OPTIONS
NASA-LeRC

ELEMENT	POWER (kW) (Day/Night)	BASELINE	ISOTOPE OPTION	NON-NUCLEAR. OPTION
HABITAT	10/9	Array + RFC	DIPS + Array	PV + RFC
ISRU	2/.2	Habitat	Habitat	Habitat
ROVER				
Tele	1.0	Batt or RFC	DIPS	RFC
Manned	2.2	RFC	DIPS	RFC
LANDER	.5/2	PFC	DIPS + PFC	PFC
SCIENCE				
Geophysical Solar Physics	.25	RTG	Sm DIPS	RFC
Telescope Suite	.65	RTG	Sm DIPS	RFC
Geologic Toolset + Traverse Package	.60	Rover	Rover	Rover
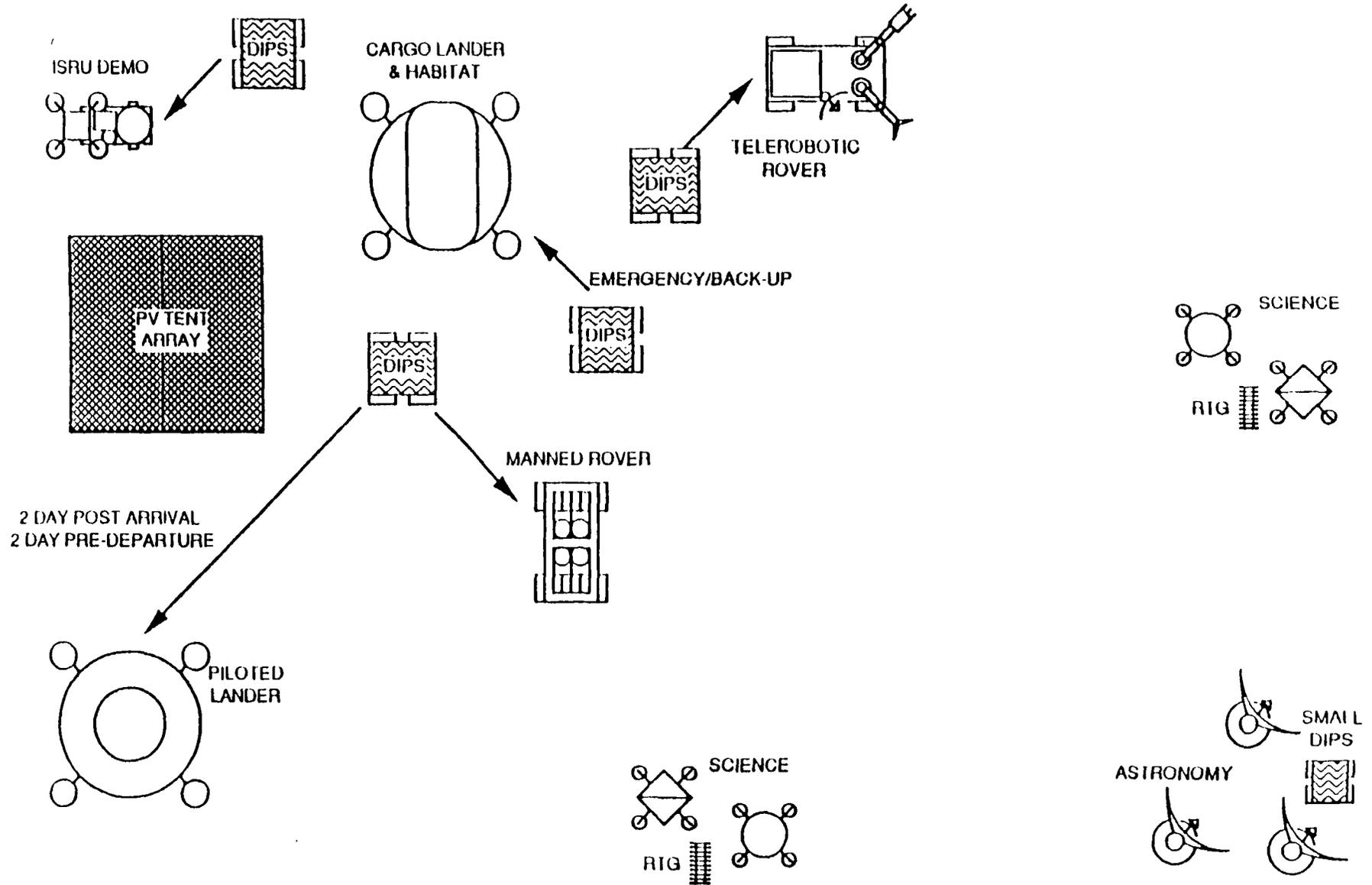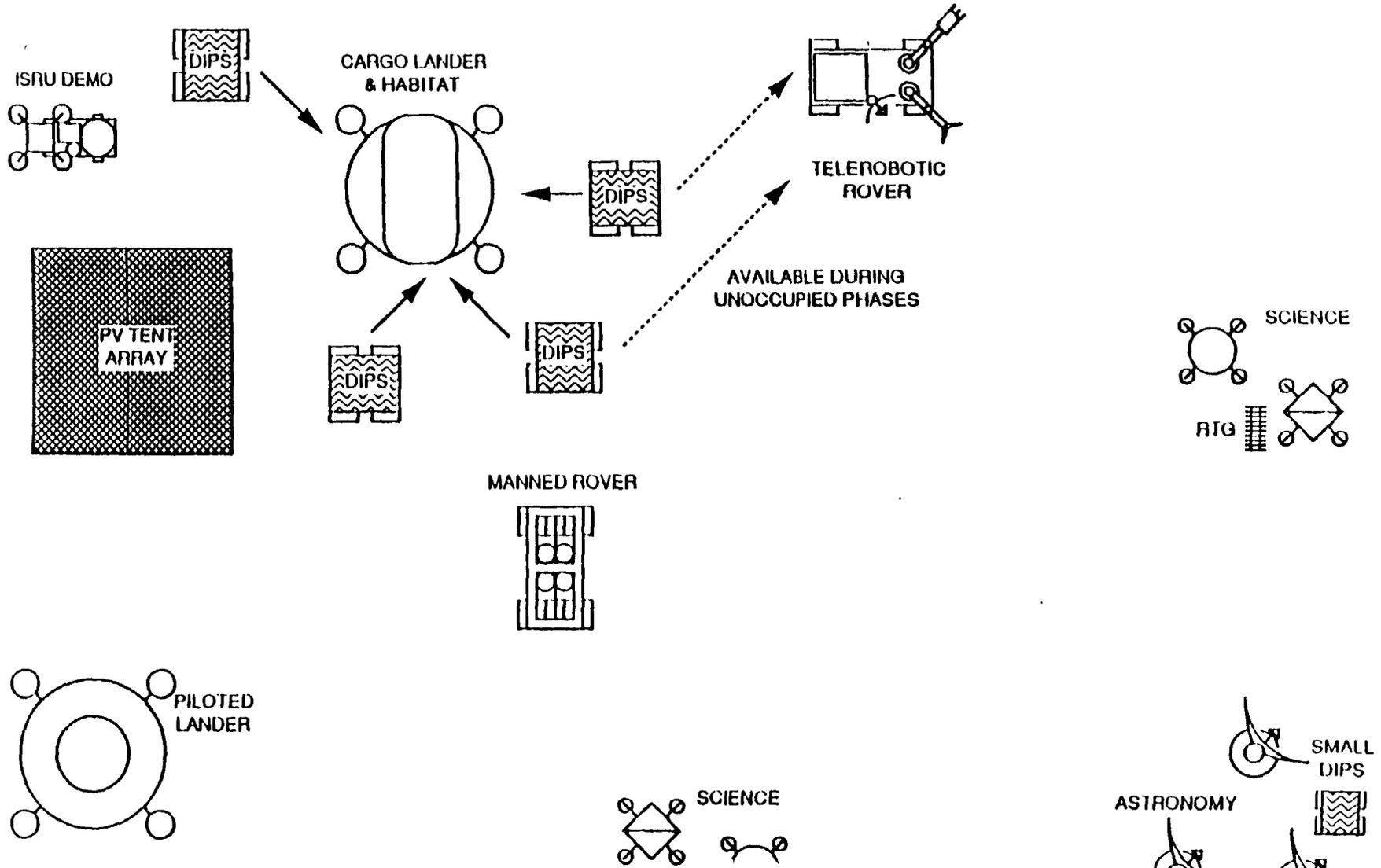
DIPS vs RTG ESTIMATED COSTS

$3000/g x 450 g Pu/ BRICK = $1.3 M/ BRICK FOR Pu

$1.3 M/ BRICK (Pu) + $700 K/ BRICK (FABRICATION) = $2.0 M /BRICK

1 kWe DIPS = 17 BRICKS x $2 M/ BRICK = $34 M

1 kWe RTG = 60 BRICKS x $2 M/ BRICK = $120 M

TOTAL COST DIPS = $34 M FUEL + $4 M UNIT = $38 M/kWe
TOTAL COST RTGS = $120 M FUEL + $30 M CONVERTERS = $150 M/kWe

2.5 kWe DIPS POWER CART

6.42 FT

12 FT
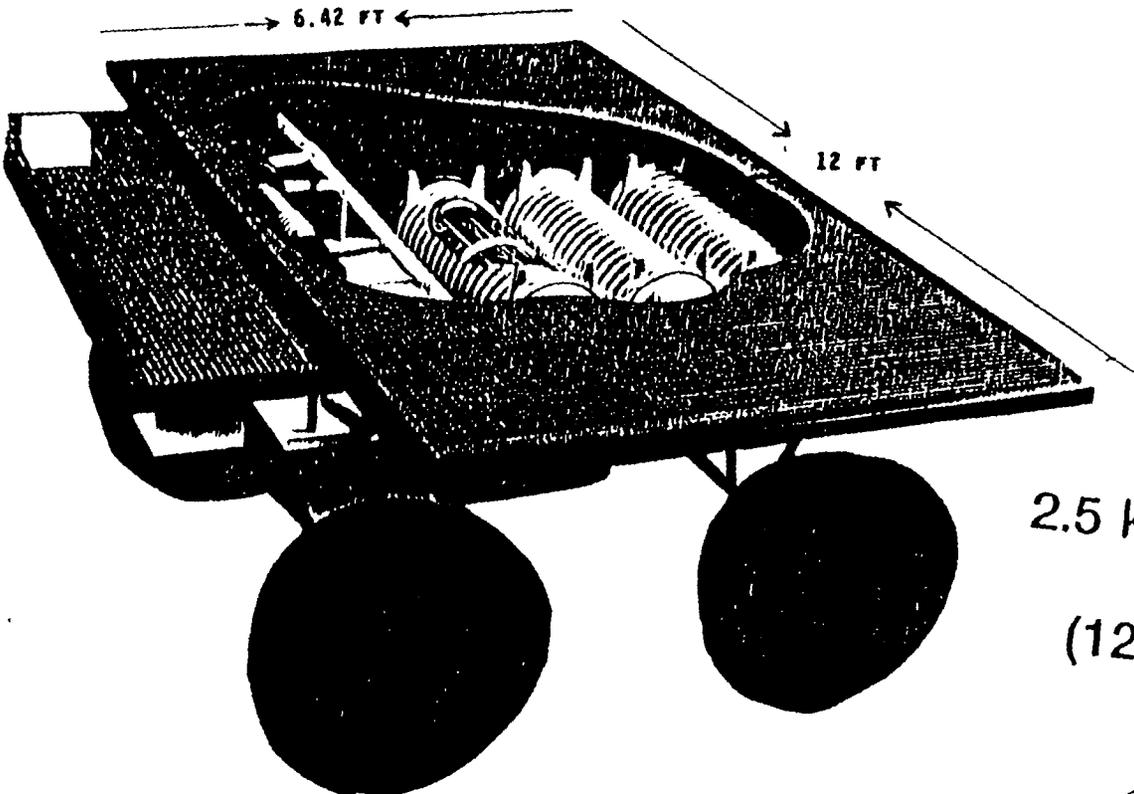
2.5 kWe RFC POWER CART

(12 HOUR DURATION)

12.99 FT

6.42 FT
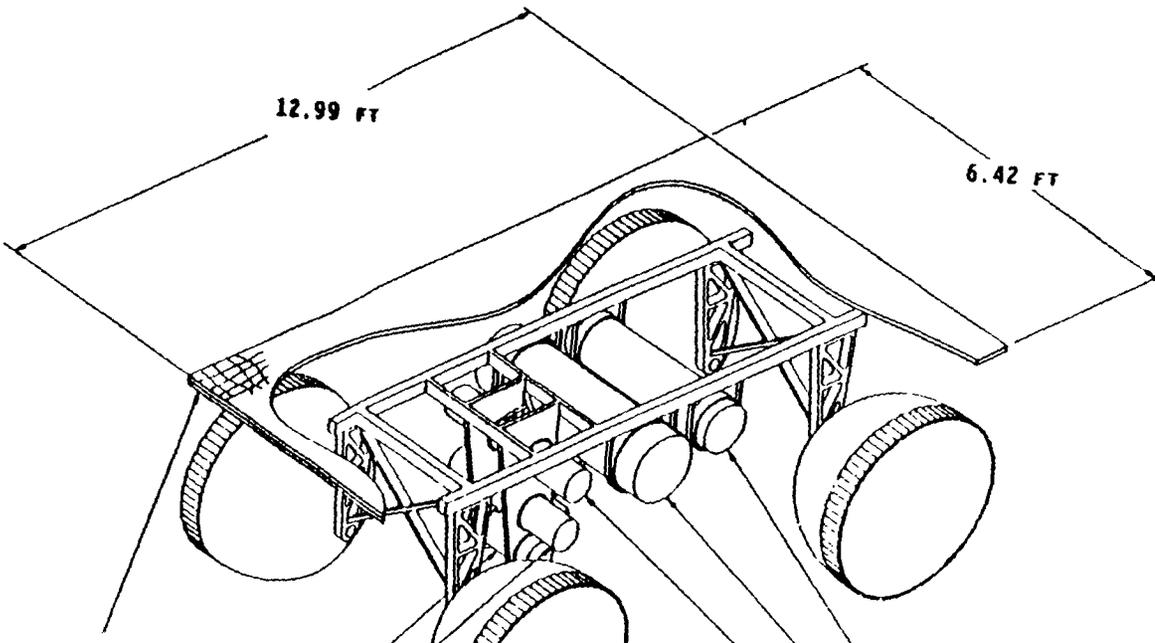
TWO-2.5 kWe DIPS MODULES (5 kWe TOTAL)

FISCAL YEAR	92	93	94	95	96	97	98	99

DESIGN POINT SELECTION

SYSEM/COMPONENT DESIGN

COMPONENT DEVELOPMENT

  HEAT SOURCE

  POWER CONVERSION

  HEAT REJECTION

  POWER COND & CONTROL

INTEGRATED SYSTEM TEST — FAB — ASSY — TEST — LIFE TEST

FLIGHT COMPONENT FAB

COMPONENT QUALIFICATION

SYSTEM QUALIFICATION — ASSY — TEST — LIFE TEST

FLIGHT UNITS — ASSY — TEST — Package — Ship — Integration — Launch

SAFETY PROGRAM — PSAR — USAR — FSAR

LAUNCH SUPPORT

Near-Term
Thermoelectric Nuclear Power Options
For SEI Missions

Jerry R. Peterson
General Electric Company

Presented to
The 3rd SEI Technical Exchange Meeting
University of Houston
May 5 - 6, 1992

- **General Purpose Heat Source RTG**

  - Qualified and Flown for Galileo/Ulysses

  - Fabrication Underway for Cassini

- **Modular RTG**

  - Life Verification Underway

  - Available for 1996 MESUR Launch

- **SP-100 Early Flight Options**

  - Current Technology 6 kW_e System for 1996 Launch

  - Baseline Technology 15 kW_e System for 1999 Launch
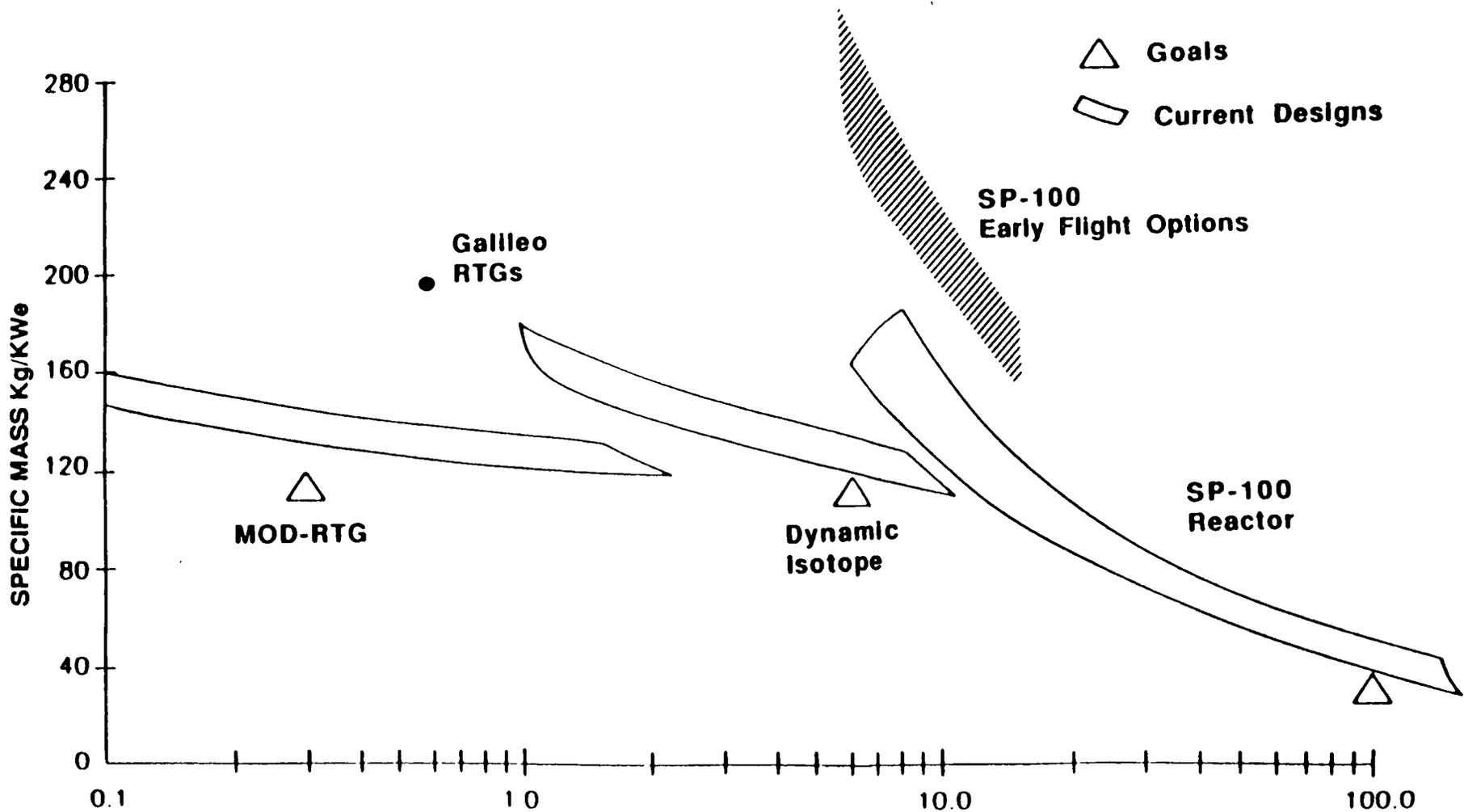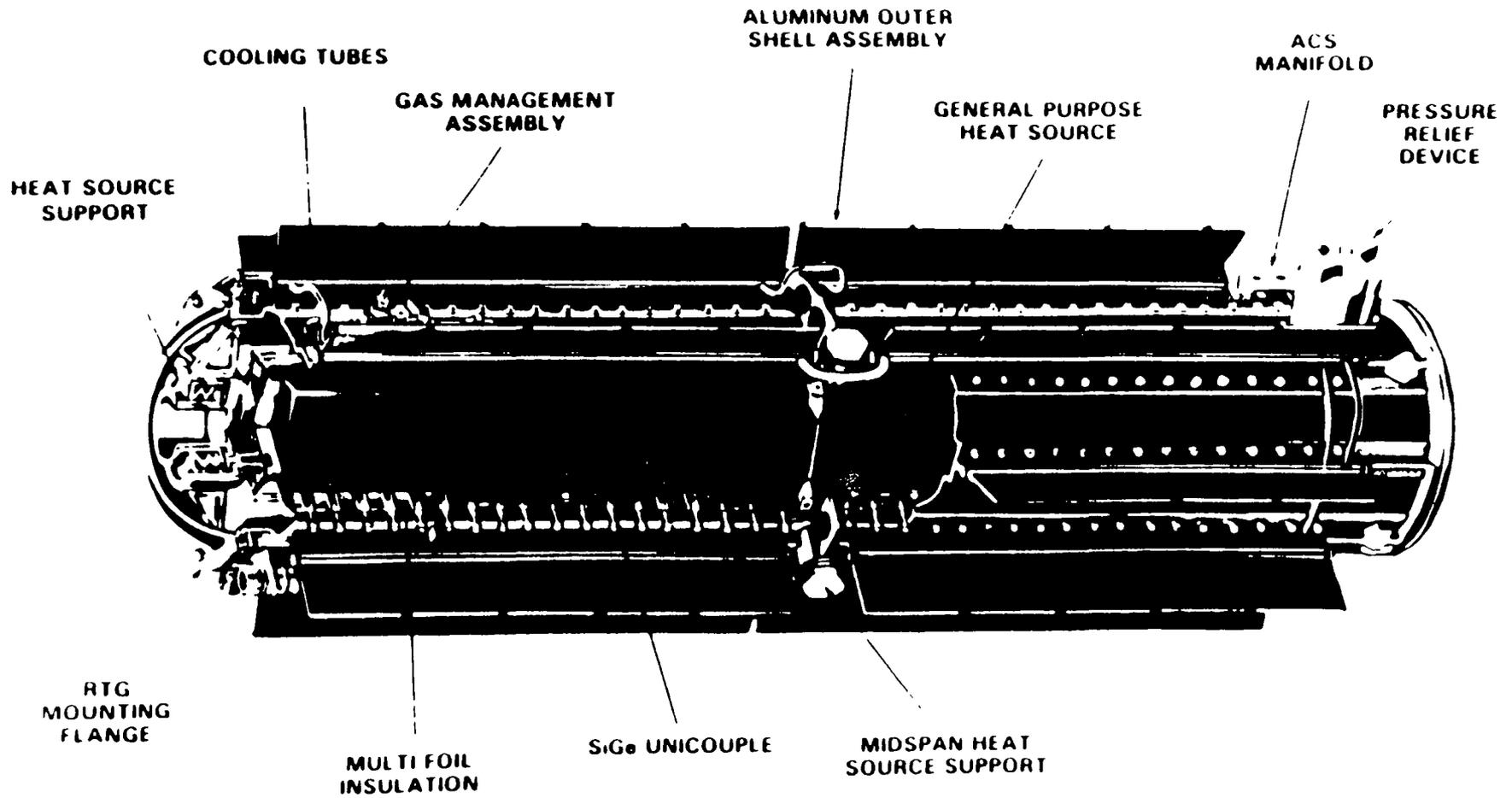
RTG Cumulative Megawatt-Hrs In Space
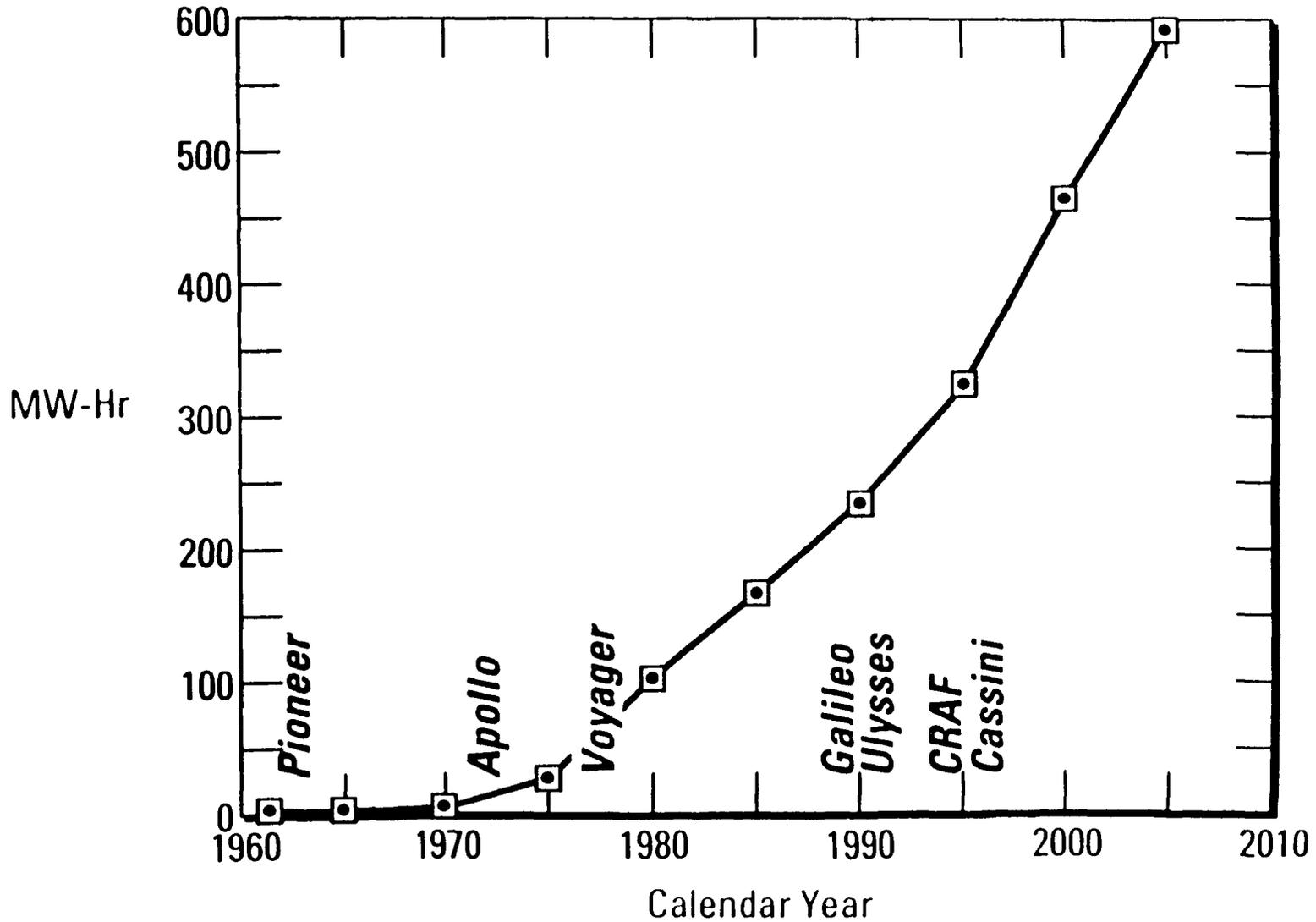
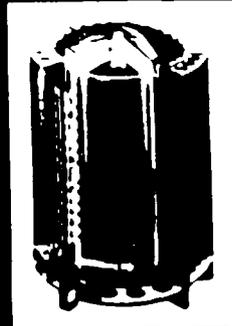Projected Specific Mass
For
Nuclear Space Power Systems

GE Aerospace

COOLING TUBES

GAS MANAGEMENT
ASSEMBLY

ALUMINUM OUTER
SHELL ASSEMBLY

GENERAL PURPOSE
HEAT SOURCE

ACS
MANIFOLD

PRESSURE
RELIEF
DEVICE

HEAT SOURCE
SUPPORT

RTG
MOUNTING
FLANGE

MULTI FOIL
INSULATION

SiGe UNICOUPLE

MIDSPAN HEAT
SOURCE SUPPORT

92 091 14

Evolution of RTG Technology

	1960s	1970s	1980s	1990s +
	Early RTGs (SNAP 3, 9A, 19, 27)	MHW-RTG	GPHS-RTG	MOD-RTG
Missions	- TRANSIT 4, 5 - TRIAD - VIKING - PIONEER - NIMBUS - APOLLO	- LES 8/9 - VOYAGER 1/2	- GALILEO - ULYSSES - CRAF - CASSINI	- MARS-ROVER
Specific Power (Watts/KG)	≤ 2.2	4.0	5.3	7.7
Thermoelectric Material	Lead Tellurides	Silicon Germanium	Silicon Germanium	Improved Silicon Germanium
Thermoelectric Component Configuration	Single Couples	Single Couples (Unicouple)	Single Couples (Unicouple)	Multiple Couples (Multicouple)
Heat Source	Single Container	Single Container	Multiple Containers	Multiple Containers (General Purpose)

GENERATOR DESIGN

MODULAR SEGMENT
19 WATTS, 30.8 VOLTS

MULTI-FOIL INSULATION

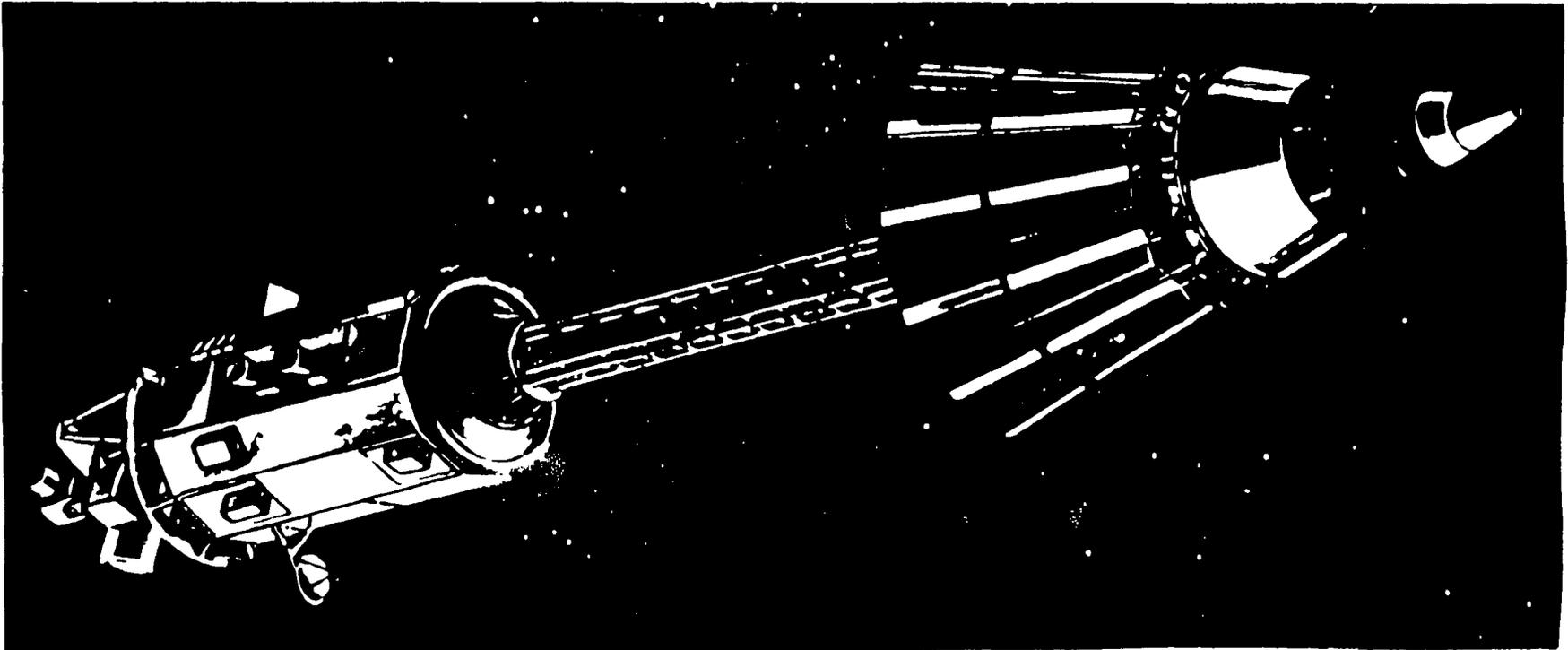GENERATOR HOUSING

HEAT SOURCE MODULE

RADIATOR FIN

THERMOELECTRIC MULTICOUPLE

42.5"

13.1

Design

Design Refinements Based On Extensions Of Present Technology Results In Significant Improvements

Survivability

Concepts Defined To Achieve Hardness To Current Spec And Super Threat Levels

Flexibility/Scalability

Technology Scales Over Large Power Range And

Development

Significant Accomplishments Made In

- **Based On Using Proven RTG Unicouple Thermoelectric Converters That Provide a Logical Evolution to Higher Performance GFS Thermoelectric Converters**

- **Will Demonstrate The Safety Features and Facilitate Safety Approval for Subsequent Missions Through the INSRP Process**

- **Flight Demonstration Mission Will Provide the Catalyst Needed to Enable Subsequent Operational Missions Based on Technology That is Adaptable to a Variety of Orbital, Interplanetary and Lunar/Mars Surface Power Applications**

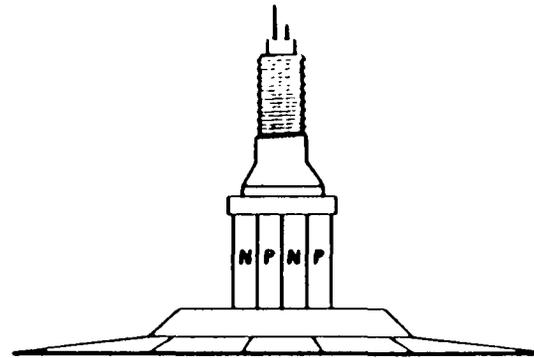- **SP-100 Early Flight Design Concepts Have Been Developed That are Compatible with Atlas and Delta Launch Vehicles**
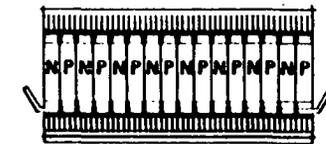
92 056 06
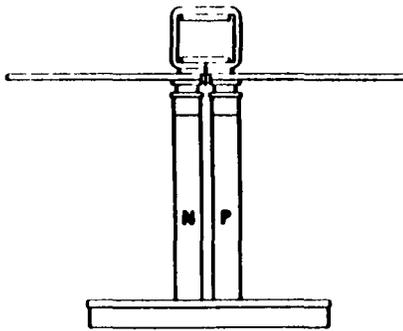
**LES 8/9, Voyager &
Galileo Unicouple**

SiGe
Single TE Couple
Radiative Coupling
Ths – 1000°C
Tcs – 300°C
0.5 Watt

**MOD-RTG
Multicouple**

SiGe + GaP
Multiple TE Couples
Radiative Coupling
Ths – 1000°C
Tcs – 300°C
2.5 Watts

SP-100 TE Cell

SiGe + GaP
Multiple TE Couples
Conductive Coupling
Ths – 1032° C
Tcs – 590°C
13 Watts

CONVERTER MODULE
INTERCONNECTS
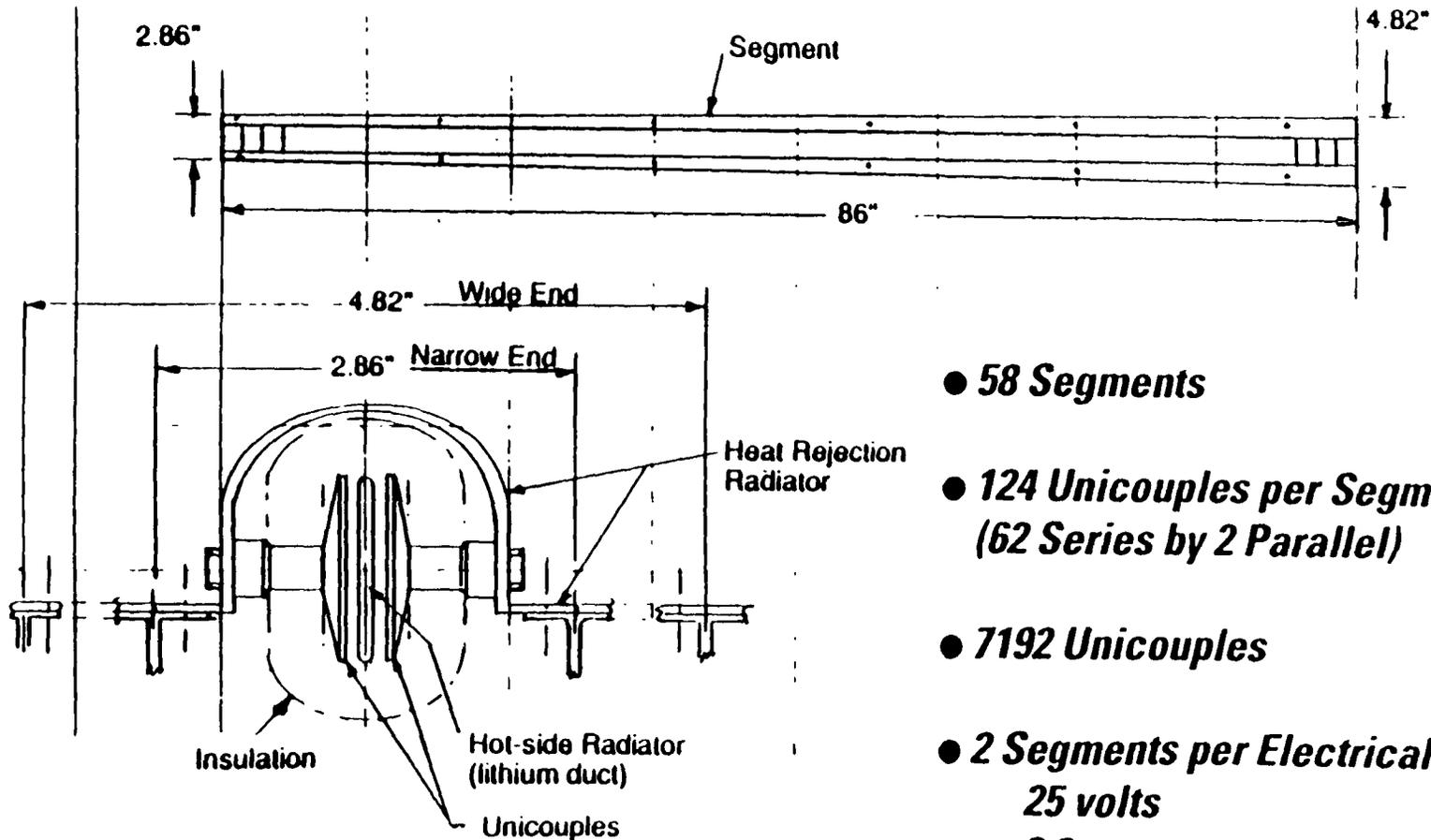HEAT EXCHANGER
STRUCTURE
TEM PUMP
HEAT EXCHANGER
CONTROL MOTORS
UNICOUPLE
SHIELD
CONTROL REFLECTORS
SAFETY RODS
REACTOR
PUMP RADIATOR
GAS SEPARATOR ACCUMULATOR
HEAT TRANSPORT PIPING
RE-ENTRY SHROUD

Segment

2.86"

4.82"

86"

4.82" **Wide End**

2.86" **Narrow End**

Heat Rejection Radiator

Insulation

Hot-side Radiator (lithium duct)
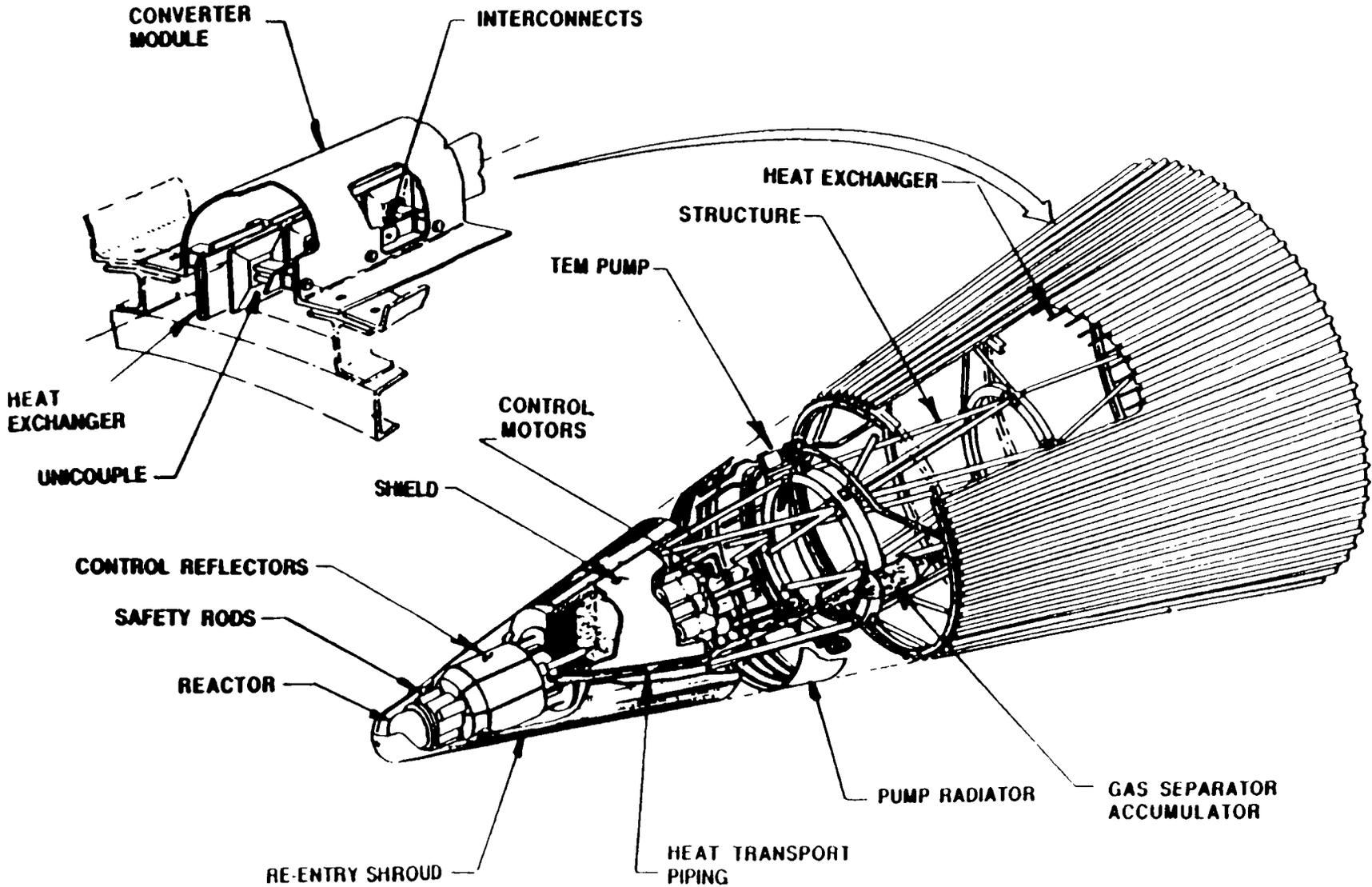
Unicouples

- **58 Segments**

- **124 Unicouples per Segment (62 Series by 2 Parallel)**

- **7192 Unicouples**

- **2 Segments per Electrical Module:**
  25 volts
  8.8 amps
  220 watts

Option	Power	Life	Specific Mass	Conversion Type	Launch Vehicle	Earliest Launch Date	Status
GPHS-RTG	285 W_e	>15 Yrs.	190 Kg/Kw_e	Unicouple	Titon - IV	Flying	Qualified for Flight In Production
Modular RTG	15 - 340 W_e	>2 Yrs.	140 Kg/Kw_e	Radiant Multicouple	Thor/Delta	1996	Life Verification in Progress MESUR Launch Date
SP-100 Baseline Technology	15 Kw_e	2 Yrs.	175 Kg/Kw_e	Conduction Multicouple	Atlas IIAS	1999	Technology Under Development. Two Year Fuel Load.
SP-100 Current Technology	6 Kw_e	2 Yrs.	250 Kg/Kw_e	Unicouple	Delta II	1996	Existing Technology. Two Year Fuel Load.

- All Near-Term Options Under Existing Contract
- Current SP-100 Contract Has Flight Option
- Mod RTG Flight Would Require New Contract

RF FEL For Power Beaming

Rocketdyne Briefing
Space Exploration Initiative Technical Interchange

Lyndon B. Johnson Space Center
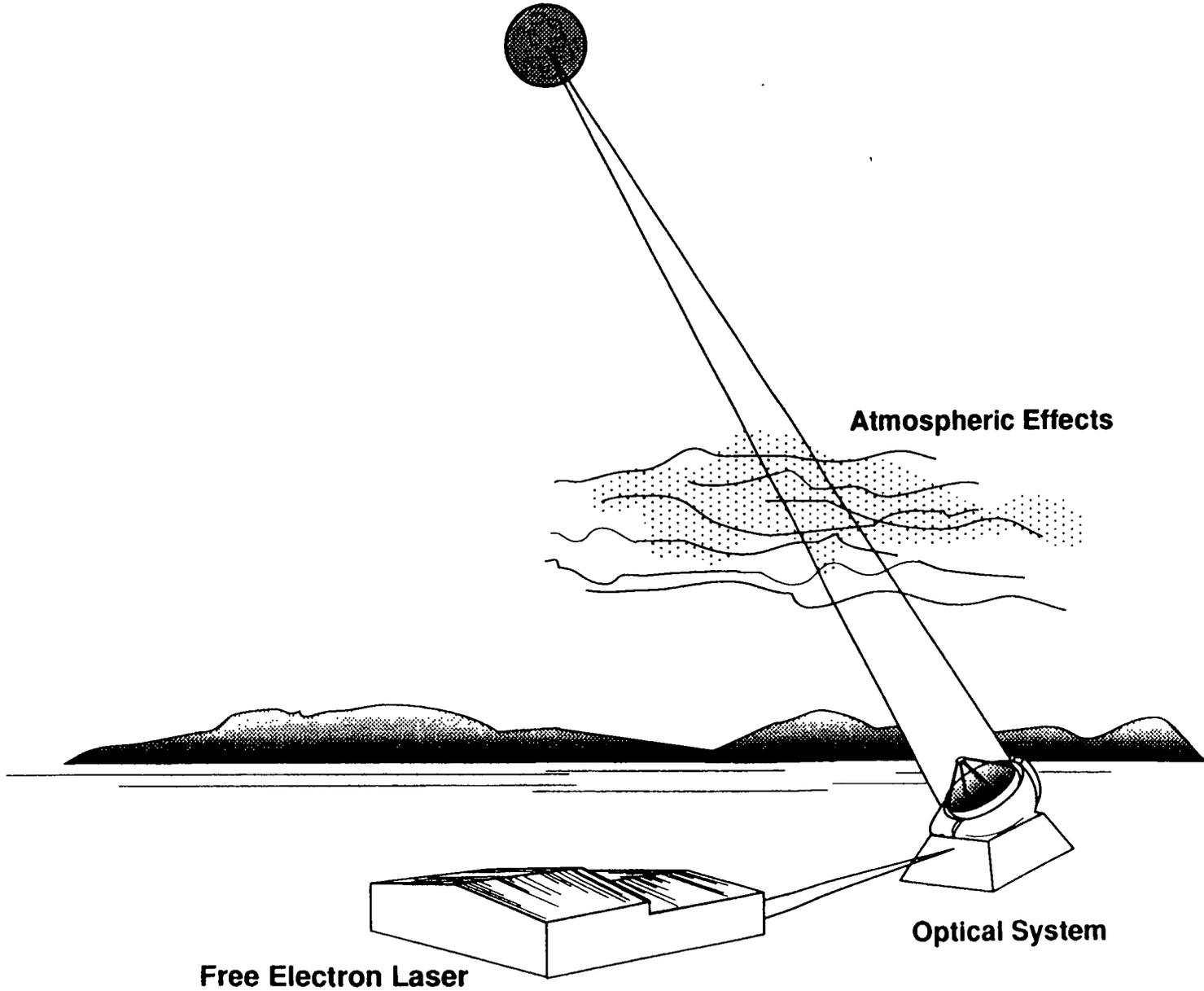May 5 & 6, 1992

Rockwell International
Rocketdyne Division
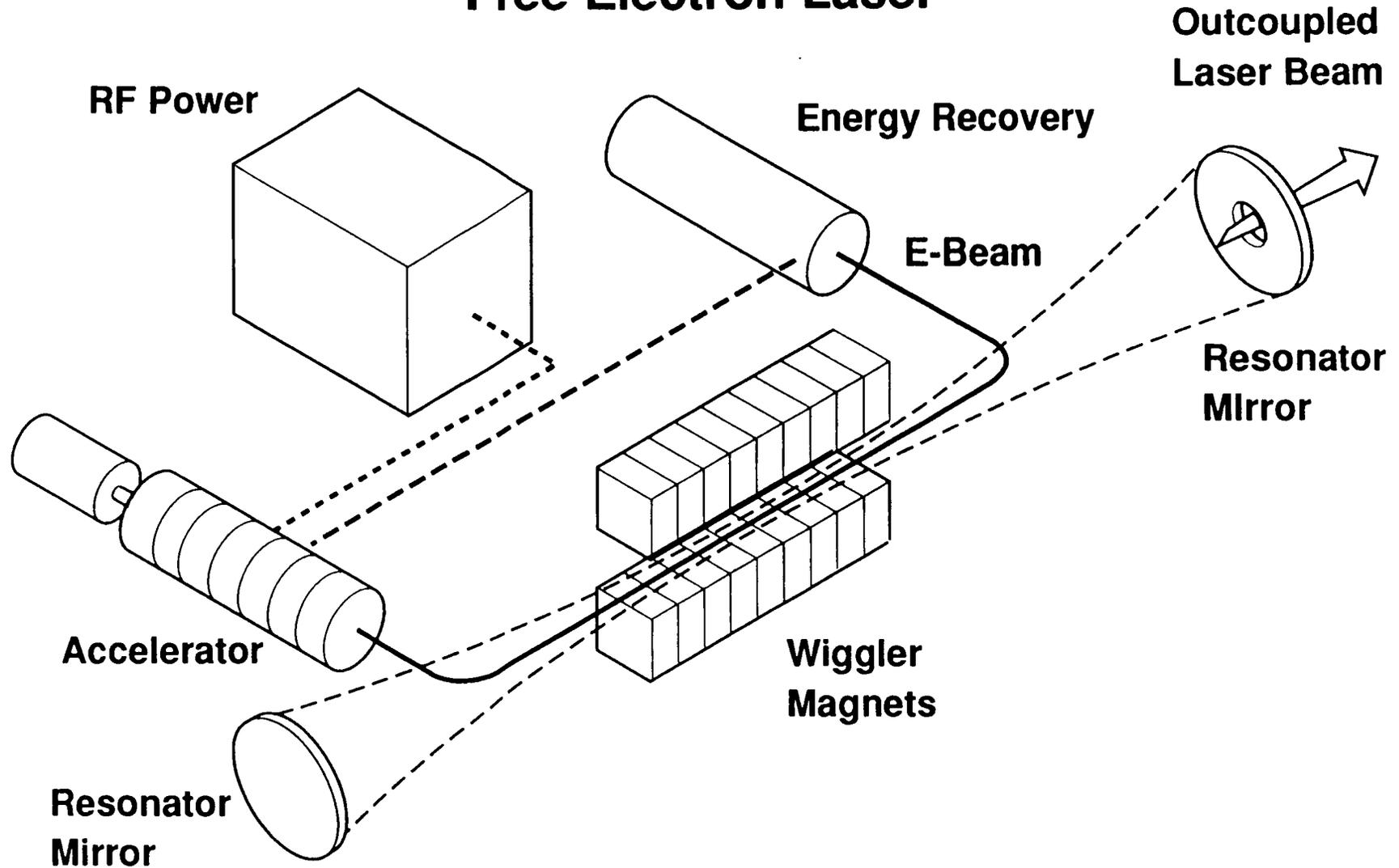
Rocketdyne RF FEL For Power Beaming

Rocketdyne Division of Rockwell International has designed and tested all of the laser device components associated with operating an RF FEL for beaming power from Earth. Analysis of the power beaming system requirements reveals that the FEL, identified by NASA as the laser of choice, is the major subsystem requiring demonstration before proceeding further in proving the efficacy of laser power beaming. Rocketdyne has identified a series of low cost, low risk demonstrations which proceed sequentially from a 1kW proof-of-principle demonstration through a 150kW demonstration of beaming power to a satellite to a MW class demonstration of Earth to lunar surface power transmission. This sequence of events can be completed in 5.5 years at a cost of $188M, with key milestones each year. In coordination with the High Energy Laser System Test Facility (HELSTF) directorate at WSMR, Rocketdyne has identified available HELSTF facilities which support implementation of this tightly scheduled, economical program. POC: Dr Robert Burke, (818) 700-3917

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Rocketdyne Division

387

Laser Power Beaming

Atmospheric Effects

Optical System

Free Electron Laser

Free Electron Laser

RF Power

Energy Recovery

Outcoupled Laser Beam

E-Beam

Resonator Mirror

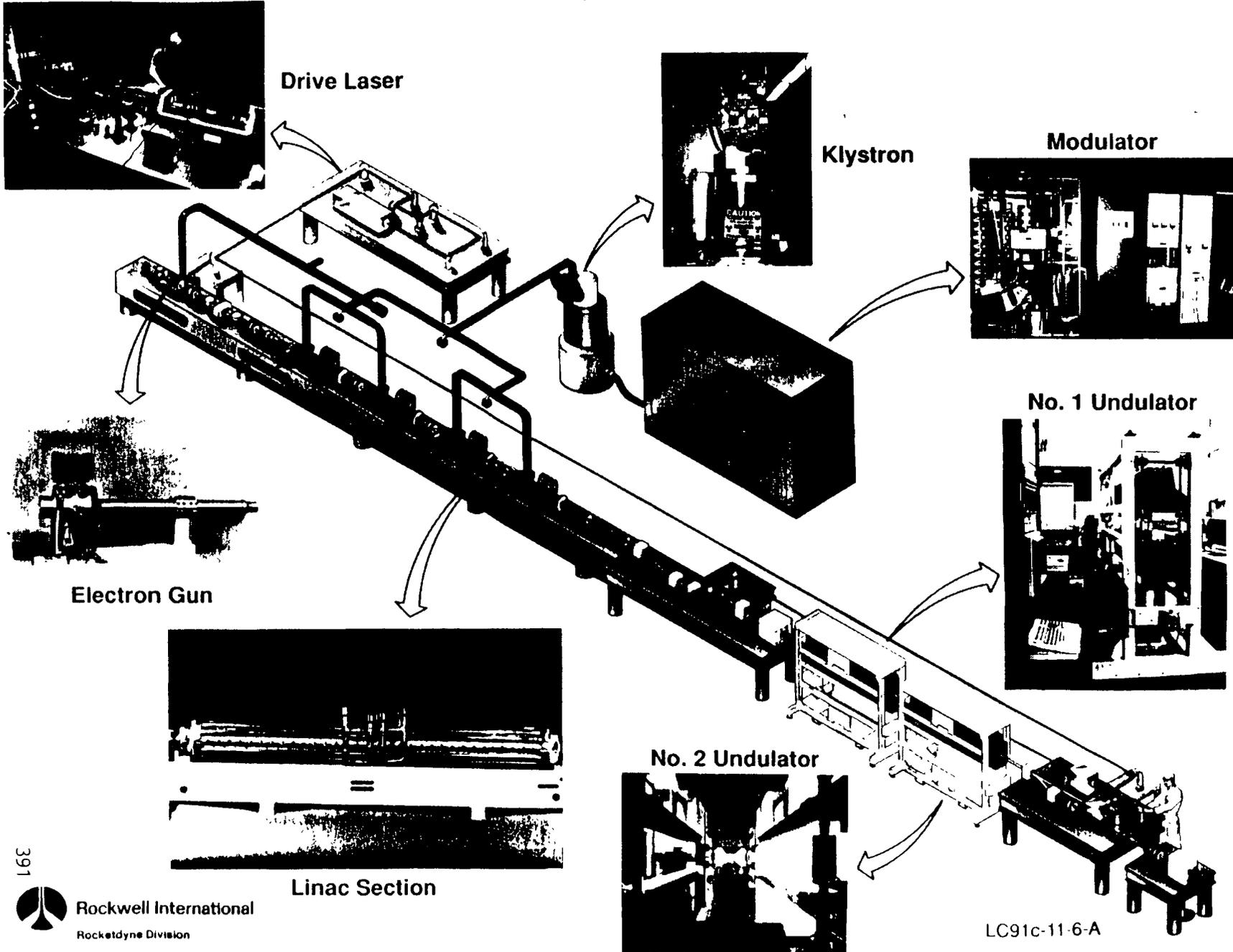Accelerator

Wiggler Magnets

Resonator Mirror

1581-5

389

Rocketdyne Approach to Power Beaming

- **RF FEL is the right choice**
  - **Chosen by SDIO for Ground Based FEL Experiment**
  - **Demonstrated high photo cell conversion efficiency in JPL, MSFC, LeRC illumination demo at Rocketdyne**
  - **Easily tunable to maximum response point of whatever P-V cell material is selected**
- **Rocketdyne RF FEL components are all proven, highly capable designs**

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Rocketdyne 1 kW FEL

Drive Laser

Klystron

Modulator

No. 1 Undulator

Electron Gun

No. 2 Undulator

Linac Section

Rockwell International

Rocketdyne Division

LC91c-11-6-A

Rocketdyne is Ready to Start
RF FEL Power Beaming Demo Program

- **Currently working with USASDC to install components at HELSTF Test Cell 4**
  - Coordinated requirements for 1 kW demonstration with HELSTF director and staff
  - Developed layout for 1 kW FEL
  - Coordinating with Navy for use of HELSTF SKYLITE beam director for power beaming demo
  - Growth to MW power levels on 5 year program
  - Can provide low cost demonstration of GEO satellite illumination within 4 years of funding
- **Milestones and cost of sequential scaling of RF FEL to higher power demonstrations have been developed**
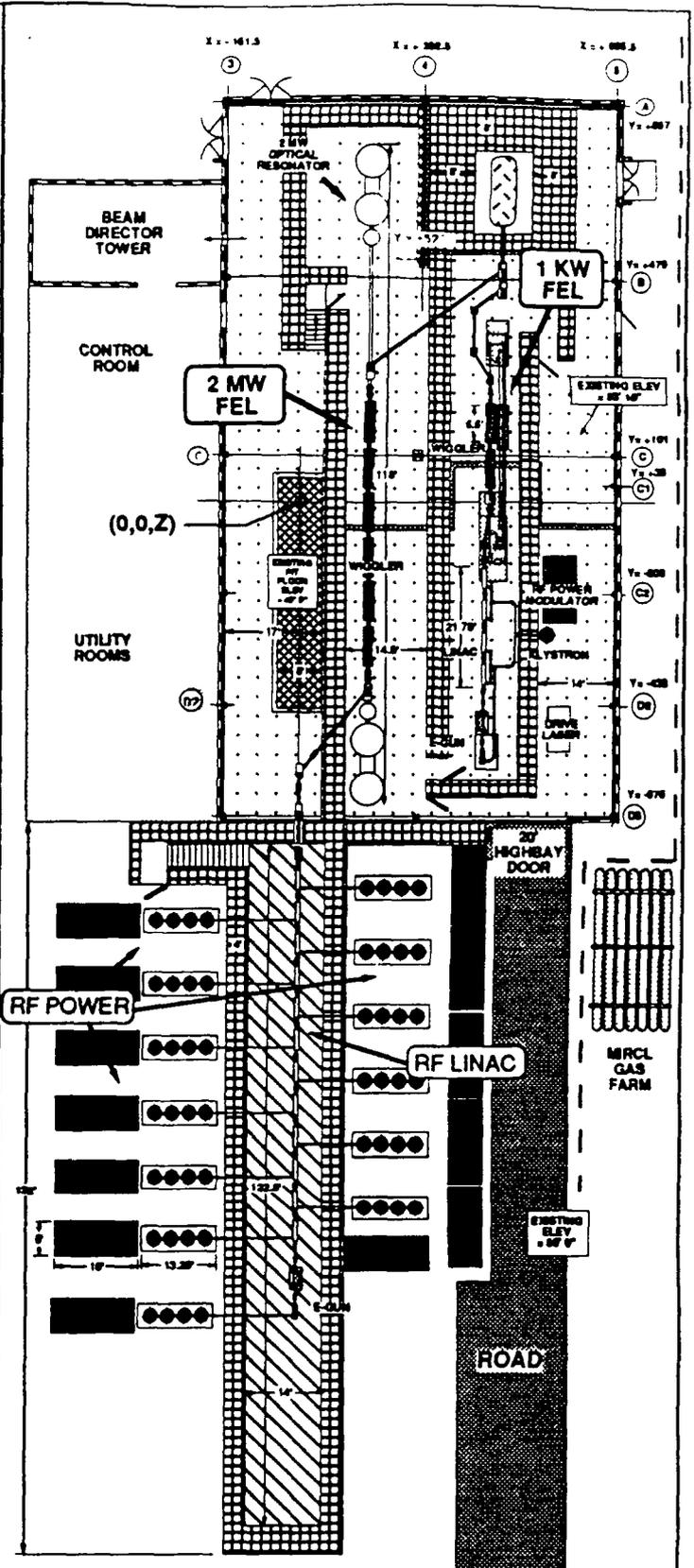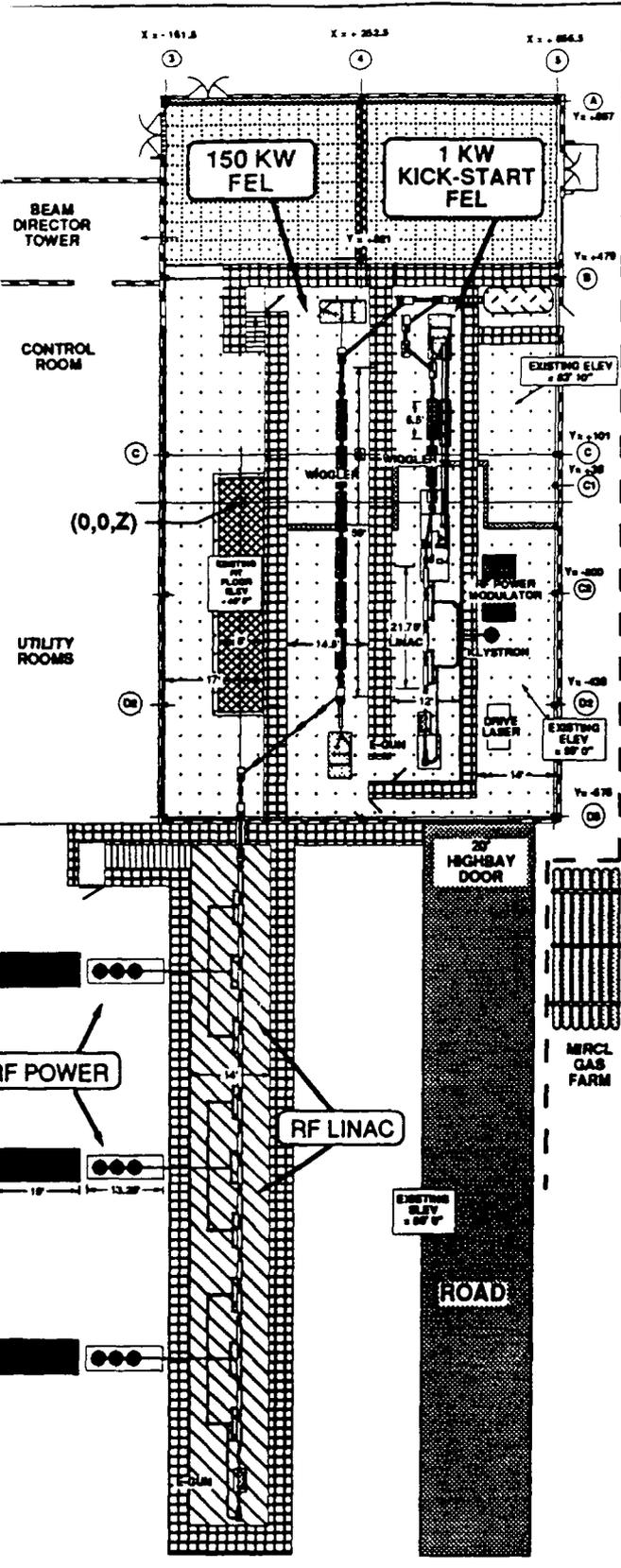
150 KW FEL INSTALLATION

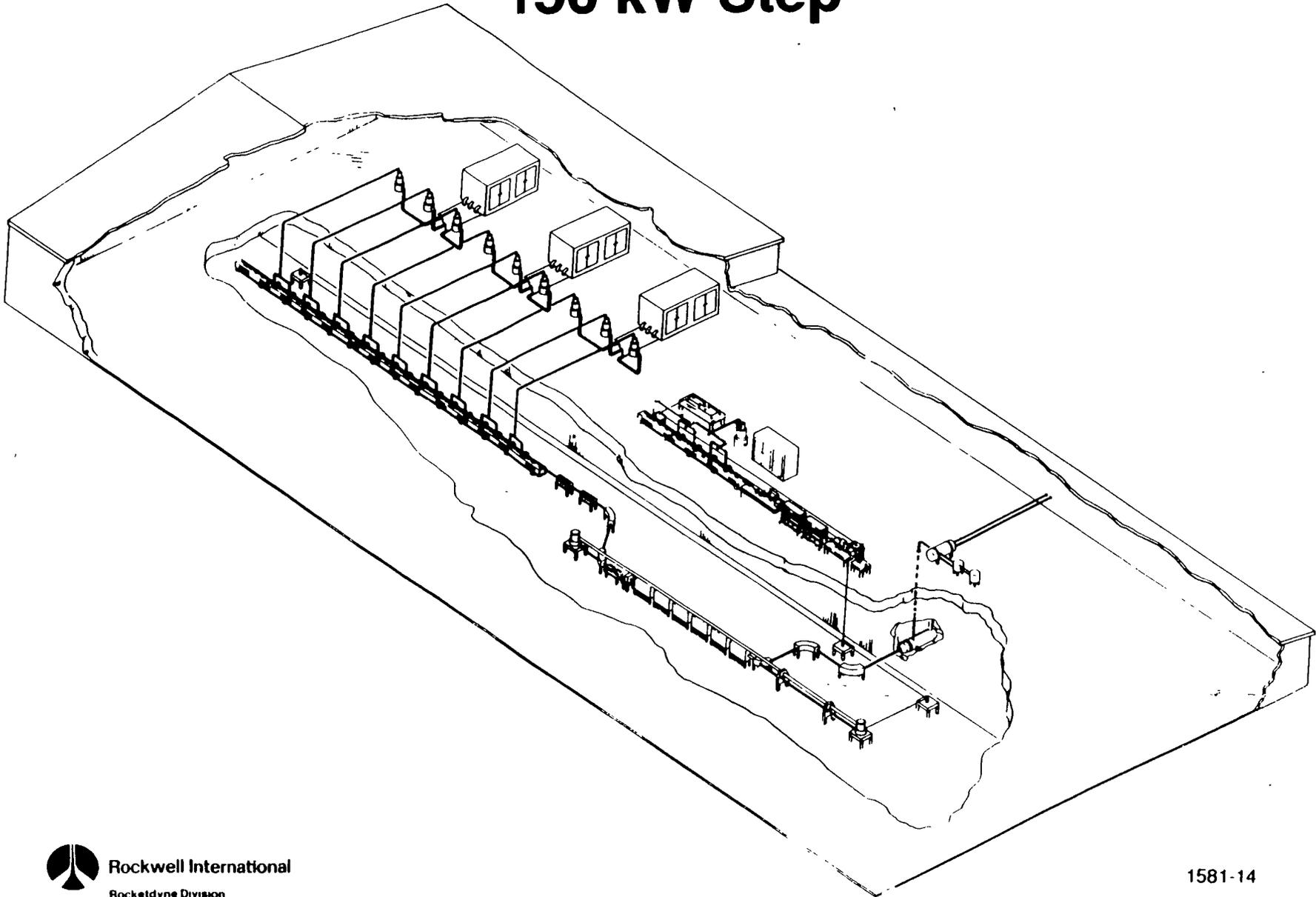**2 MW FEL INSTALLATION**

HELSTF CELL 4, WSMR

393
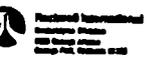
HELSTF Cell 4
White Sands Missile Range

150 kW Step

1581-14

Multi-MW FEL
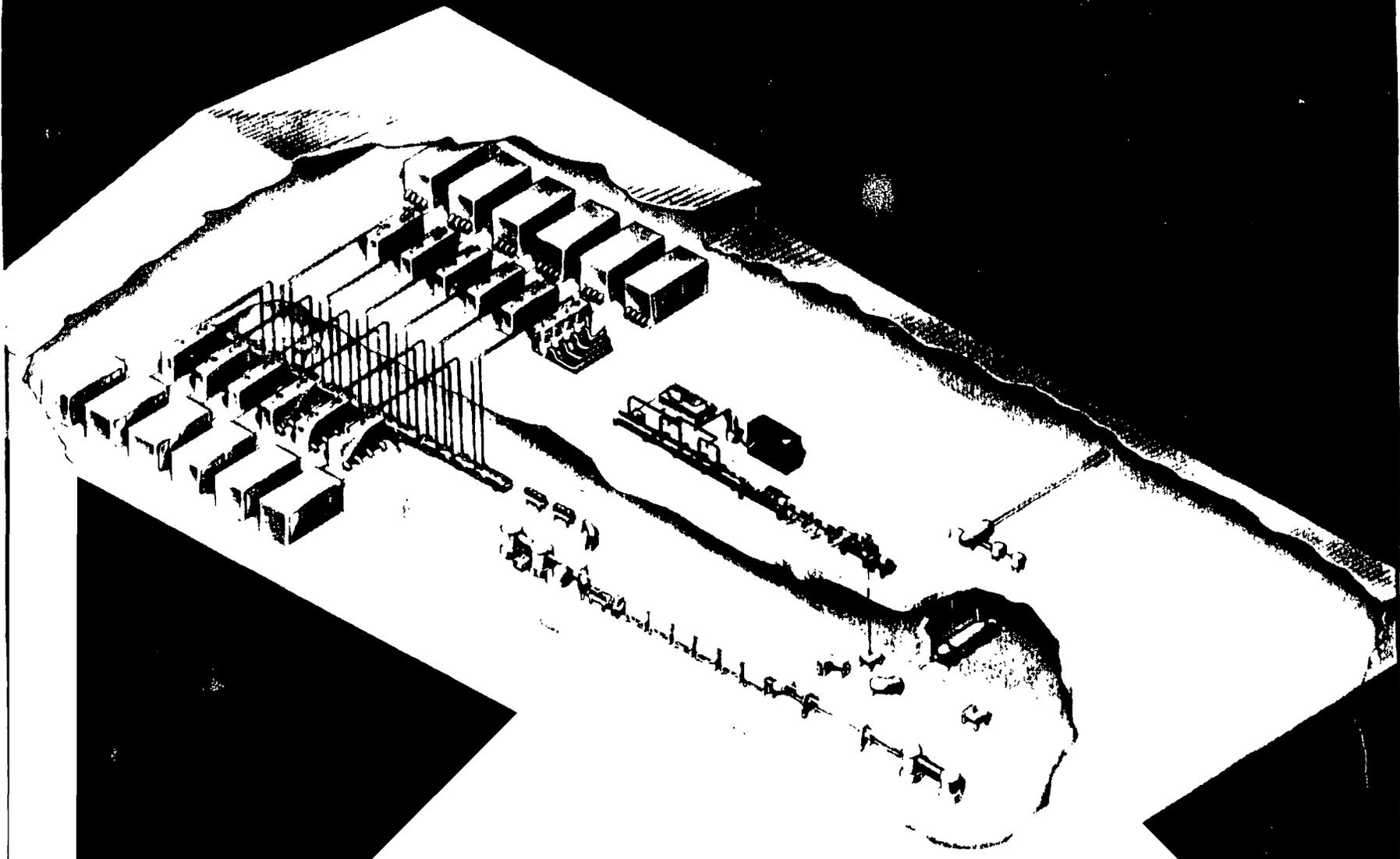
1 kW FEL and 2 MW FEL Linacs Use Proven Technologies
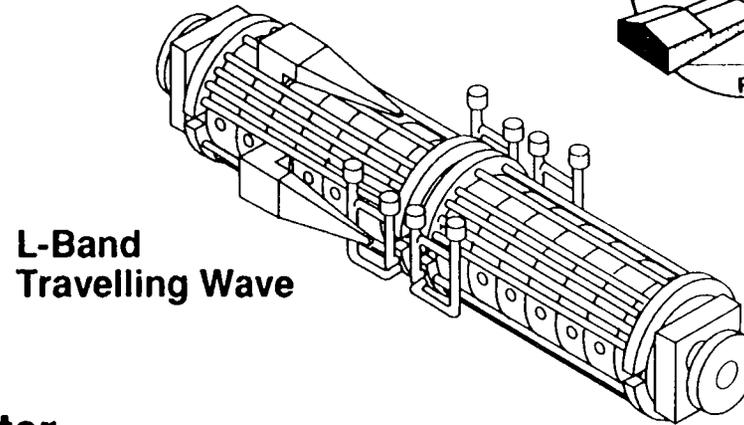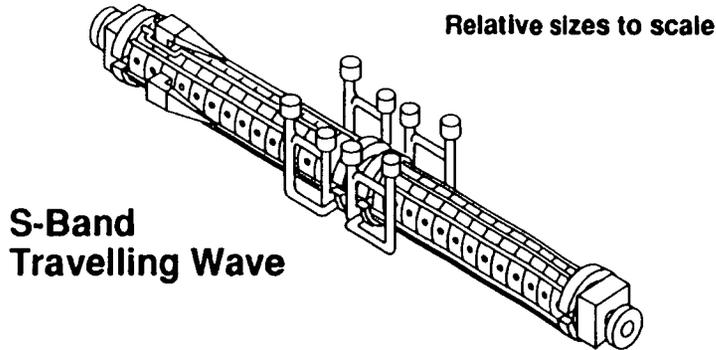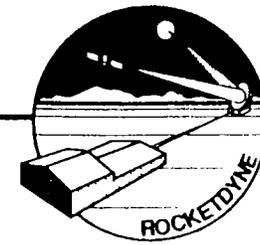
Relative sizes to scale

S-Band Travelling Wave

L-Band Travelling Wave

ROCKETDYNE

1 kW FEL		Parameter	2 MW FEL	
Nominal	Max		Nominal	Max
1.25 m	—	Length	1.25 m	—
56 MΩ/m	—	Shunt Impedance	39 MΩ/m	—
2856 MHz	—	Operating Frequency	1428 MHz	—
15 MV/m	30 MV/m	Gradient	8 MV/m	21 MV/m
1 nC	4 nC	Macropulse Charge	2.8 nC	10 nC
500 A	1000 A	Peak Current	500 A	1700 A
30 kW	60 kW	Thermal Load per Section	150 kW	200 kW

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Rocketdyne Division

1561-16b

397

2 MW FEL RF Power Uses Commercially Available Klystrons

1 kW FEL

Relative sizes to scale
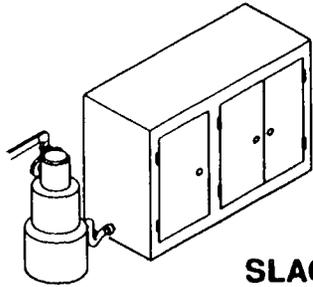
2 MW FEL

SLAC
5045 Klystron
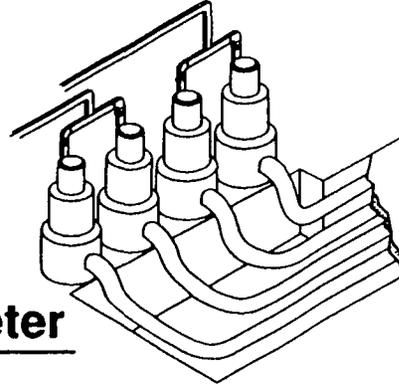
Thomson
2104-X Klystrons

1 kW FEL	Parameter	2 MW FEL
2856 MHz	Frequency	1428 MHz
65 MW	Peak Power/Klystron	8.4 MW
3.5 μsec	Macropulse Duration	100 μsec
360 Hz	Macropulse Rep Rate	360 Hz
1	Number of Klystrons/Modulator	4/8
90 kW	Average Power/Klystron	304 kW

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FEL's 3rd Harmonic Output Provides UV Light To Drive High Power Photocathode
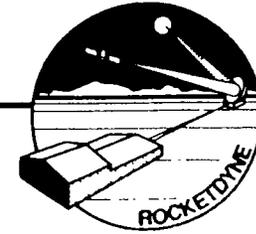
1 kW FEL		2 MW FEL
LaB_6	Cathode Material	LaB_6
352 nm	Light Wavelength	280 nm
5×10^{-4}	Quantum Efficiency	1×10^{-3}
1.0 nC	Electron charge/ μ-pulse	2.8 nC
7 μJ	Photon Energy/μ-Pulse	13 μJ
N/A	FEL 3rd Harmonic Output/μ-Pulse	40 μJ
13 Watts	Average Power on Cathode	670 Watts
6 Watts	Average Power Absorbed	330 Watts
7°C	Cathode Temperature	350 °C
3.4 A/cm^2	Thermionic Emission @ 1800 °K	3.4 A/cm^2
2210 °C	Cathode Melting Point	2210 °C
3 mm	Cathode Diameter	6 mm
20 A/cm^2	Macropulse Current Density	14.2 A/cm^2
0.026 A/cm^2	Average Current Density	0.51 A/cm^2

1561-17

2 MW FEL Wiggler Performance Scales Directly From 1 kW FEL

Relative sizes to scale
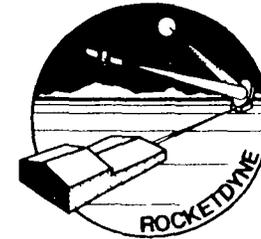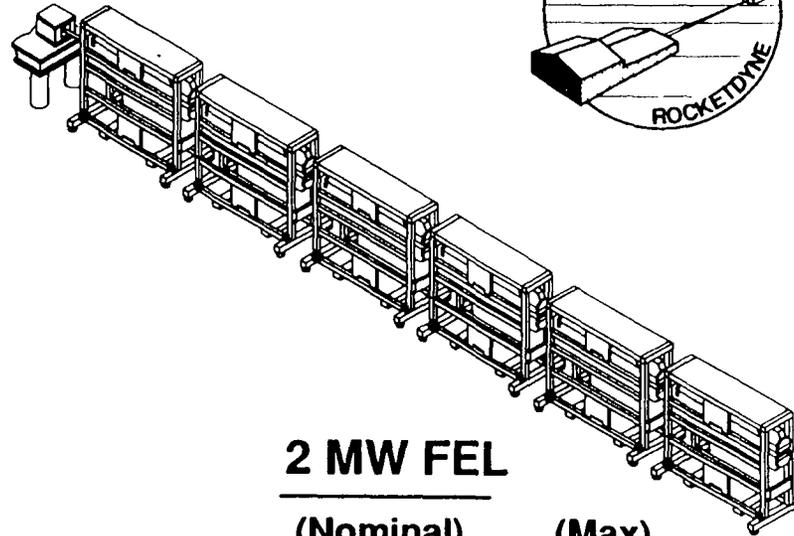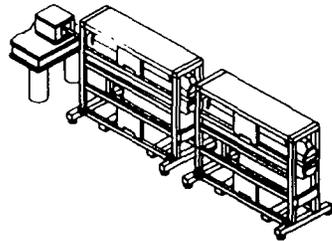
1 kW FEL		Parameter	2 MW FEL	
(Nominal)	(Max)		(Nominal)	(Max)
78 MeV	100 MeV	Electron Energy	120 MeV	—
500 A	1000 A	Electron Current	500 A	1700 A
4 m	—	Wiggler Length	12 m	—
2.4 cm	—	Wiggler Wavelength	3 cm	—
6.3 %	10%	Extraction Efficiency	15%	25%
2.7 GW	6.5 GW	Peak Optical Power	9.5 GW	30 GW

ROCKETDYNE

Rockwell International

2 MW FEL Optics Based on Components Rocketdyne
Built for HAPFEL and GBFEL
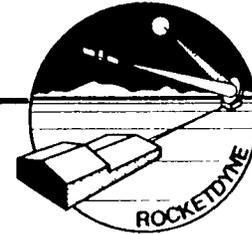
Relative sizes to scale
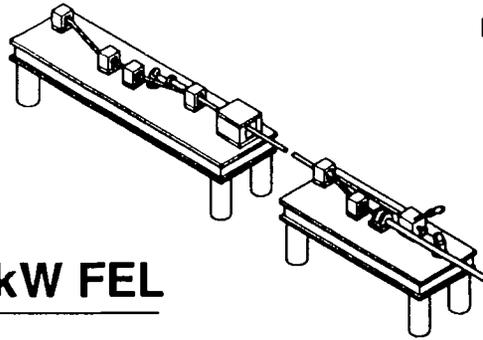
1 kW FEL

1 kW FEL		2 MW FEL
0.84 μm	Optical Wavelength	0.84 μm
300 MW	Optical Seed Power	500 MW
70%	Outcoupling Fraction	90%
6.3 %	Extraction Efficiency	15%
0.33 kW	Circulating Power	200 kW
0.7 J/cm^2	Macropulse Fluence on 1st Optic	100 J/cm^2
N/A	Damage Threshold	500 J/cm^2
Sapphire Flat	Outcoupler	Grating
Ring	Resonator Type	Grazing Incidence Ring
8 m	Mirror Spacing	37 m
15°	Incidence Angle on 1st Optic	3°
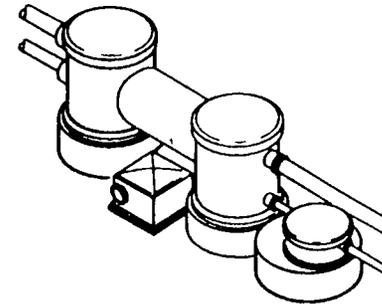
2 MW FEL

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Rocketdyne Division

401
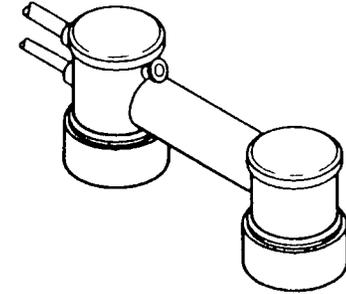
Power Beaming FEL Program

Funding requirements ($M)
year from start

	1	2	3	4	5	Step Totals
Step 1. 1 kW	2	2	–	–	–	4
Step 2. 150 kW	13	25	22	–	–	60
Step 3. 400 kW	–	10	25	25	–	60
Step 4. 2 MW	–	–	10	24	30	64
Totals	15	37	57	49	30	188

△ 1 kW   △ 150 kW   △ 400 kW   △ 2 MW

↑ *Proof-of-Principle*   ↑   ↑ *EOTV*   ↑

GEO Satellite   *Preliminary Lunar Power Capability*
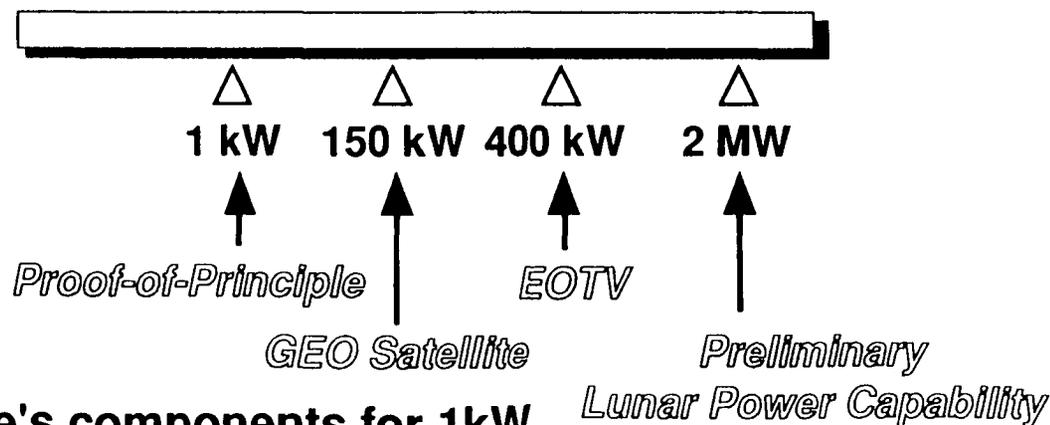
- Assumes Rocketdyne's components for 1kW

- Facility costs and schedule not included

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Laser Power Beaming Proof-of-Principle

Space Shuttle
- Sun power ➔ laser power
  - Intensity = 1/3 solar
  - Efficiency = 3 x solar
- 1.5m diameter spot

Atmospheric Effects
- Absorption – minimized with wavelength selection
- Turbulence – minimized with adaptive optics
  - Beacon
  - Deformable mirror
  - Fast steering mirror

Free Electron Laser
- Power – 1 kW

Optical System
- Beam director: 1.5m diameter
- Pointing accuracy: 1 milliarc-sec

403

1581-2

Laser Power Beaming Demonstration and Satellite Power Mission
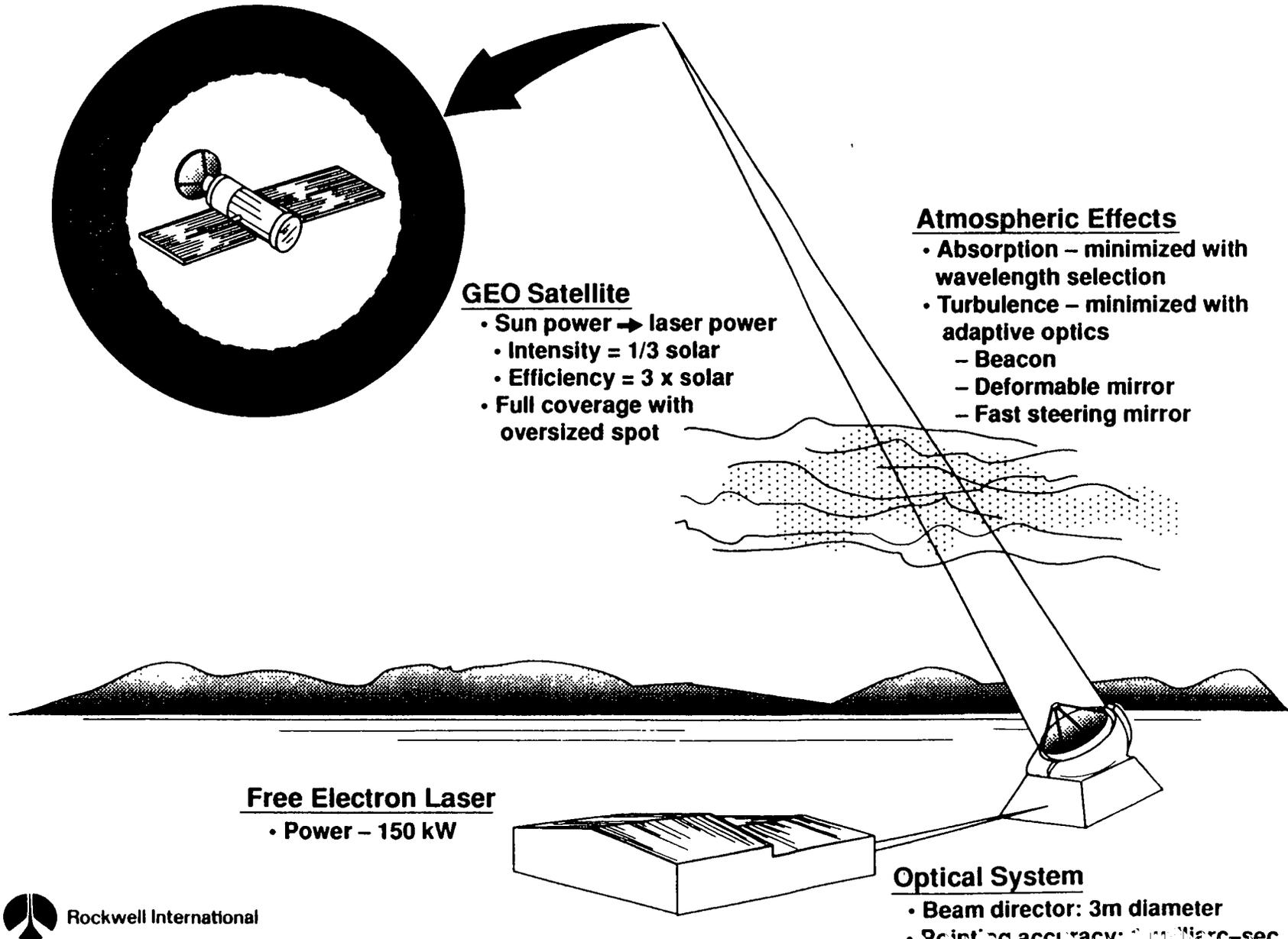
GEO Satellite
- Sun power ➜ laser power
- Intensity = 1/3 solar
- Efficiency = 3 x solar
- Full coverage with oversized spot

Atmospheric Effects
- Absorption – minimized with wavelength selection
- Turbulence – minimized with adaptive optics
  - Beacon
  - Deformable mirror
  - Fast steering mirror

Free Electron Laser
- Power – 150 kW

Optical System
- Beam director: 3m diameter
- Pointing accuracy: milliarc-sec

Rockwell International

Laser Power Beaming for Lunar Mission

Atmospheric Effects
- Absorption – minimized with wavelength selection
- Turbulence – minimized with adaptive optics
  - Beacon
  - Deformable mirror
  - Fast steering mirror

Lunar Base
- 6 MW in 65m diameter spot
- Efficiency 3 x solar
- Intensity 75% x solar
- 2–3 MW electric power

Free Electron Laser
Power ≳ 10MW

Optical System
- Beam director: 12m diameter
- Pointing accuracy: 1 milliarc–sec

Rockwell International
Rocketdyne Division

405

1581-3

OPTICAL ENERGY TECHNOLOGIES INC.
472 WESTOVER ROAD, STAMFORD, CT 06902
203-357-0626

S24-44

//5-524 1-15

THE MOON AS A SOLAR POWER SATELLITE
BY
GERALD FALBEL

At the last Space Exploration Initiative meeting I attended here, a lot of the papers discussed "Waypoints" in various space exploration programs. It is the purpose of this paper to present approaches which are "off the screen to the right", indicating what goal these waypoints should be on the way to. This is important, because, as we all realize, the day of the blank check for space exploration is over, and the taxpayers and Congress are asking more and more: What do we get out of this"?

I firmly believe that in this climate, the only space exploration project that Congress and the people will support is one which directly improves their economic position or their life style.

I therefore hereby propose a modified version of the Satellite Solar Power System, (SPS), using the moon as the "satellite". I believe this is the only space project that can show a direct cash return on investment, and thus meets the above criterion.

The original SPS was conceived by Dr. Peter Glaser of MIT in 1968, as a series of large, photovoltaic solar collectors orbiting at geosynchronous altitude, and converting and beaming the collected solar power to the earth surface using microwave energy (which can pass through cloud cover). This system was studied extensively under the joint sponsorship of the Department of Energy and NASA between 1977 and 1980.

These studies concluded positively as to this concept's feasibility in hardware, legal, environmental, health, and societal acceptance areas. Unfortunately, in the 1980's President Reagan, proclaimed "Morning in America" but didn't see the sun.

The overall advantages of a Solar Power Satellite are summarized in Table I.

Figure 1, shows the reference system configuration defined by these studies, consisting of a 55 km^2 photovoltaic orbiting collector and a 1 km^2 transmitting microwave antenna. Figure 2 shows a 10 x 13 km elliptical microwave rectenna below the geostationary satellite. Figure 3 shows the breakdown of the collected solar energy and the conversion efficiencies up to the output to the electrical power grid on the earth.

I will first discuss a direct transfer of this highly defined system to a lunar platform, and will then discuss what I believe should be the ultimate lunar SPS configuration.

Figure 4 shows the reference system broken into an array of five, 100 meter high, by 55 km long, 2/1 concentrating, Ga As collectors. Notice that the aluminized polyester concentrating mirror uses the lunar gravity and lack of wind to effectively save half the cost of the collector.

These collectors are located 6˚ from the lunar pole to account for the sun's + 5˚ elevation angle variation caused by the lunar orbit's inclination to the ecliptic plane.

A-11

SILICON CR=1
BLANKET AREA = 52.34 km^2
PLANFORM AREA = 54.08 km^2

650 m
TYPICAL

5200 m

470 m

10400 m

1050 m

5250 m

463 m

625 m

10500 m

525 m

625 m

GaAlAs CR= 2
BLANKET AREA = 28.52 km^2
PLANFORM AREA = 55.13 km^2
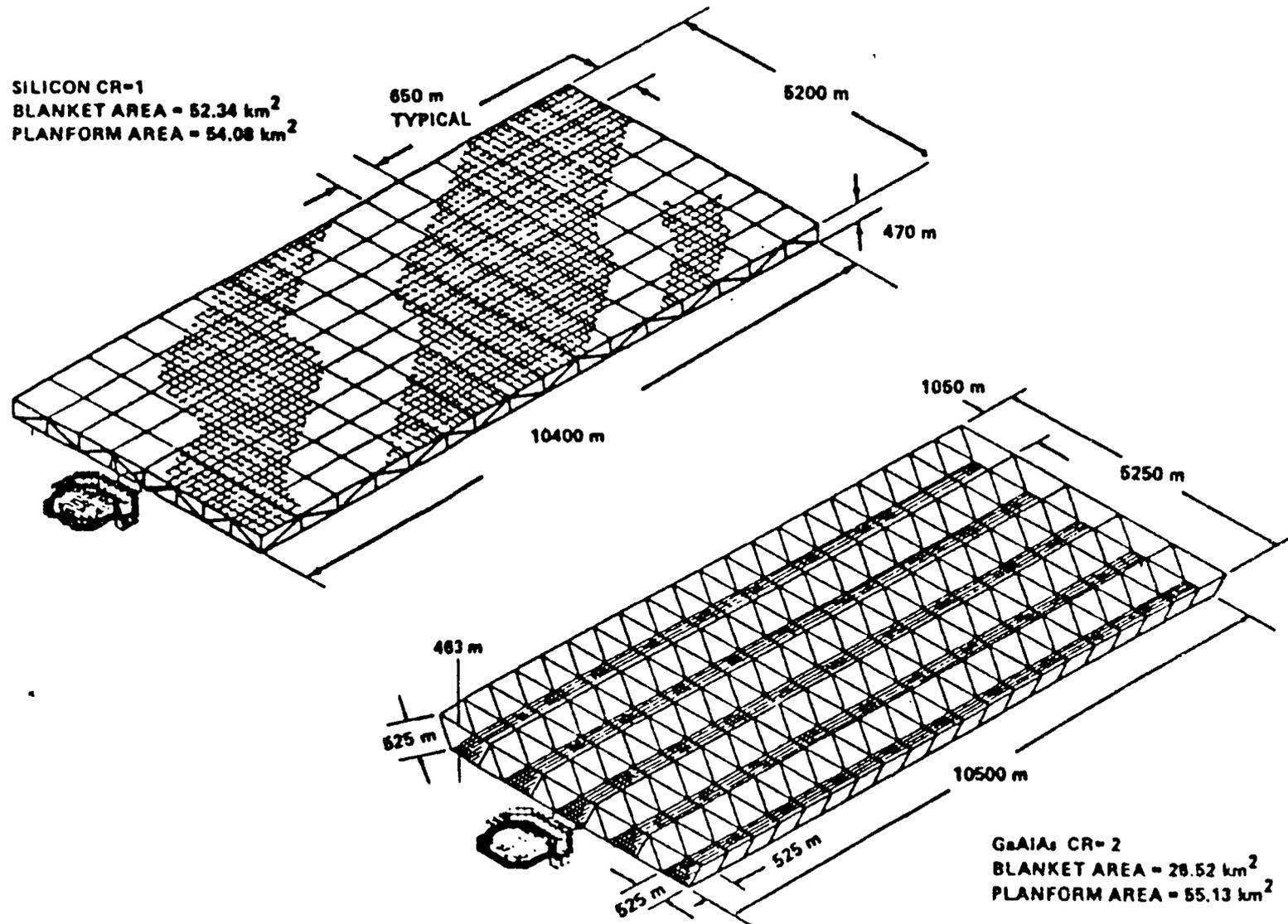
FIGURE 1   SPS REFERENCE SYSTEM - SATELLITE CONFIGURATION

Array
Structure

Solar Cell Array

Transmitting Antenna
Subarray

DC-RF
Power Amps

Antenna Waveguides

Half-Wave
Dipole
Antenna

Open-Screen
Ground Plane

Rectifying Antenna
10 km × 13 km at 35° lat.

5 km

10 km

High Power Density
Microwave Beam

1 km diam.

Low Power Density
Microwave Beam
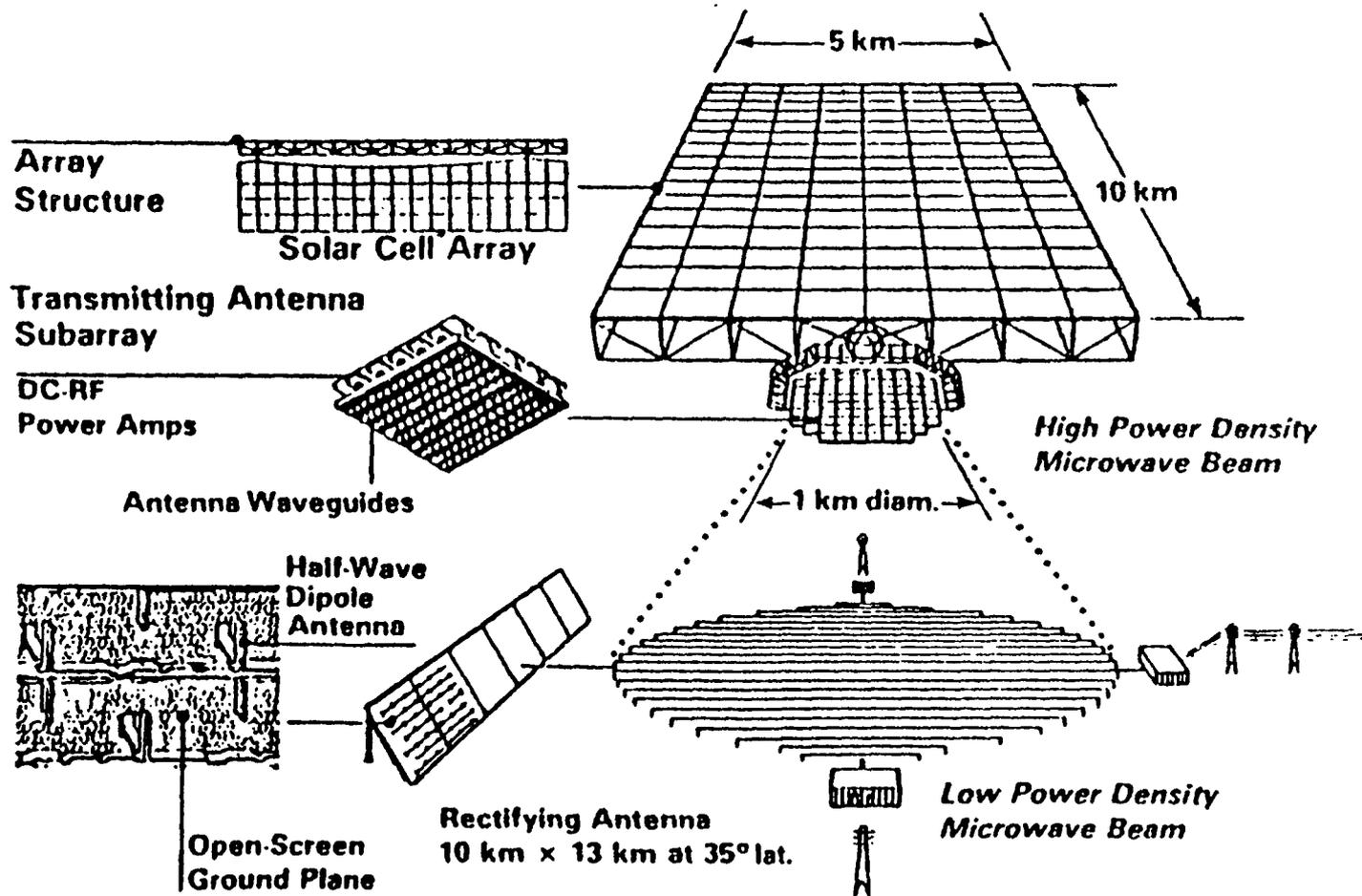
FIGURE 2 BASIC SPS REFERENCE SYSTEMS CHARACTERISTICS
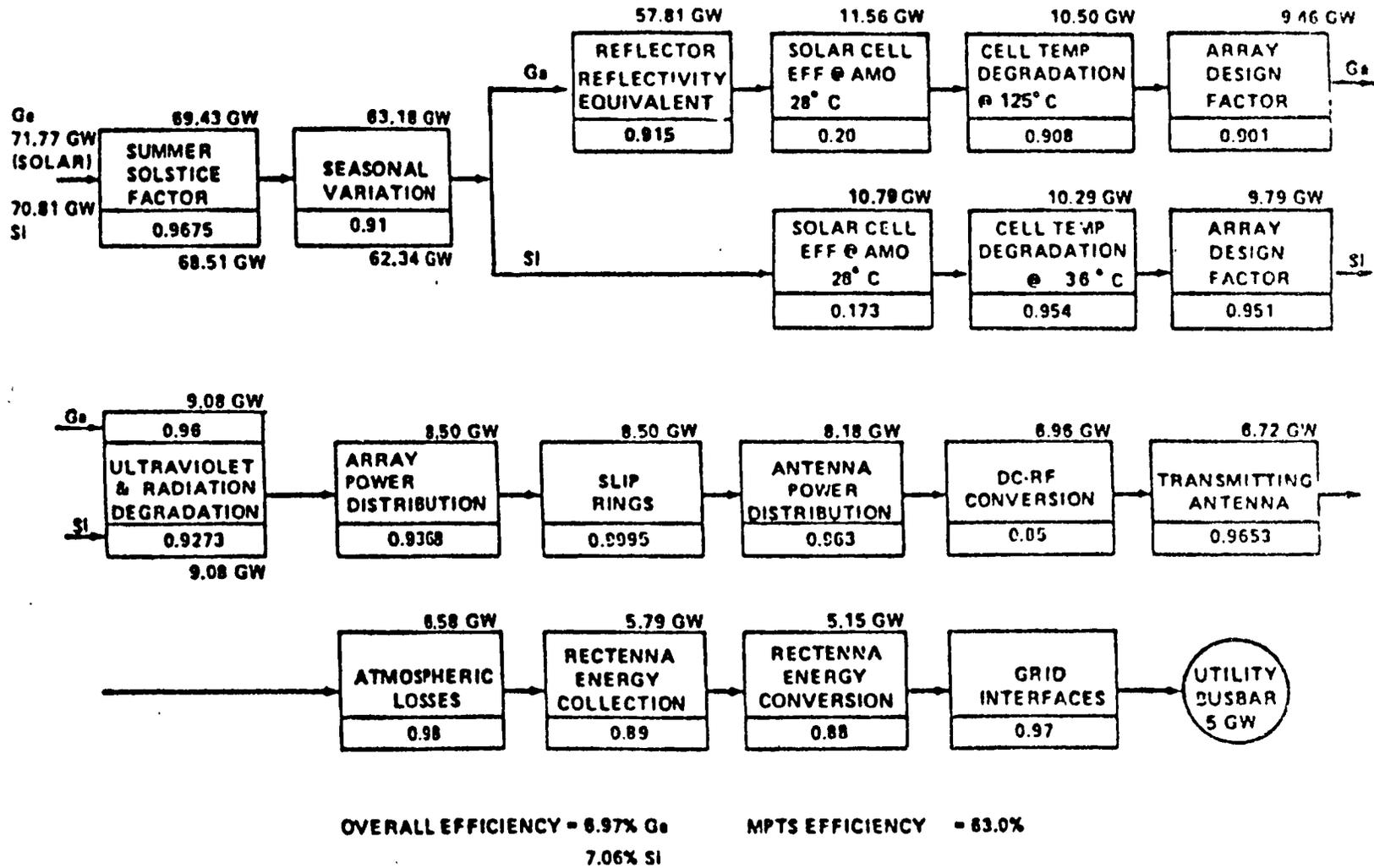
FIGURE 3   SPS EFFICIENCY CHAIN (GaAlAs CR2 and Si CR1)

50 METERS X 55 KM
GaAs SOLAR COLLECTOR

120 M

100M

LUNAR SURFACE

HEAT DISSIPATING COLLECTOR MOUNT

ALUMINIZED POLYESTER CATENARY REFLECTOR

ROLLERS FOR ALUMINIZED POLYESTER
TO EXPOSE NEW SURFACE, REPLACING
MICROMETEORITEDAMAGEDREFLECTOR

FIGURE 4   LUNAR PHOTOVOLTAIC CONCENTRATING COLLECTOR (5 COLLECTORS PER SITE)

TABLE I

OVERALL ADVANTAGES OF A SATELLITE POWER SYSTEM

1. SIGNIFICANT REDUCTION IN THE "GREENHOUSE EFFECT" CAUSED BY BURNING FOSSIL FUELS

2. A TOTALLY "CLEAN" GENERATION OF ELECTRIC POWER WITH NO NUCLEAR OR TOXIC WASTE

3. THE APPLICABLE TECHNOLOGY IS AVAILABLE TODAY AND REQUIRES NO SCIENTIFIC BREAKTHROUGH, AS DOES HYDROGEN FUSION.

4. RESUMPTION OF U.S. TECHNICAL LEADERSHIP IN SPACE, WHICH HAS BEEN LOST SINCE THE APOLLO PROGRAM.

5. APPLICATION OF U.S. ENGINEERING TALENT TO USEFUL AND CONSTRUCTIVE ENDS.

6. A DEFUSING OF THE OIL-DEPENDENT POWDER KEG OF THE MIDDLE EAST.

7. UNLIMITED AVAILABILITY OF CLEAN ELECTRIC POWER WORLDWIDE WILL BENEFIT DEVELOPING COUNTRIES, WILL SAVE THE RAIN FORESTS, AND WILL ELIMINATE ACID RAIN.

8. THIS PROGRAM WILL ACCELERATE THE DEVELOPMENT OF LOWER COST EARTH-MOUNTED PHOTOVOLTAIC SOLAR CELLS DUE TO QUANTITY PRODUCTION.
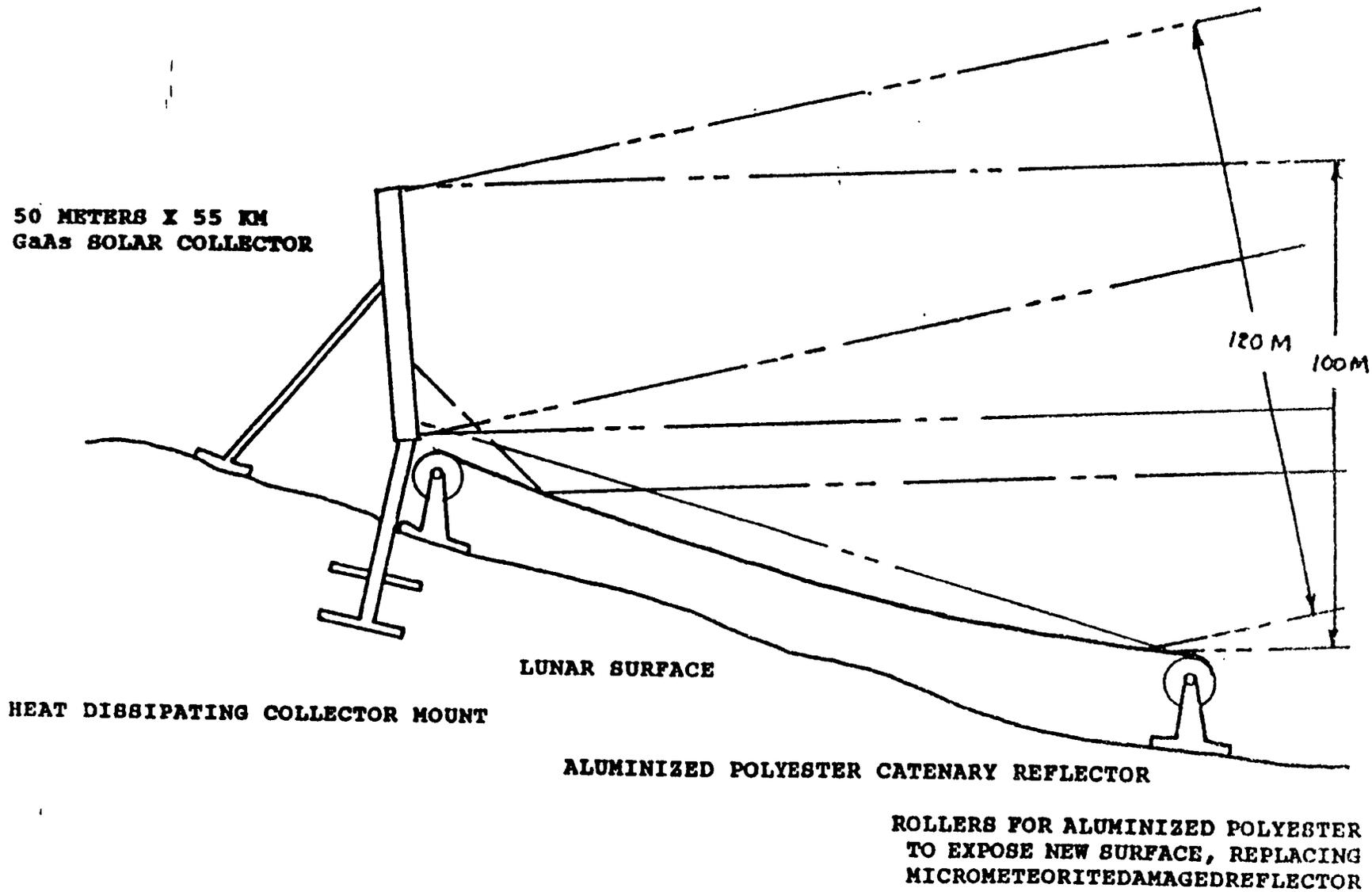
9. THE RECTENNAS RECEIVING THE MICROWAVE ENERGY TRANSMIT > 80% OF IMPINGING SUNLIGHT. IF THEY ARE MOUNTED ON FLOATING ISLANDS IN THE OCEANS, AND SYPHON OFF SOME OF THE RECEIVED MICROWAVE ENERGY, AND USE IT FOR DESALINIZATON OF OCEAN WATER, FOOD CAN BE GROWN IN ABUNDANCE TO FEED THE WORLD.

10. THIS PROGRAM IS IDEAL FOR INTERNATIONAL COOPERATIVE EFFORT WHERE EVERYONE BENEFITS, THUS MAKING WARS MORE UNLIKELY.

Table II lists the technical advantages of the lunar-based SPS compared to the geosy
chronous SPS. Table III lists its disadvantages, and the means of dealing with them

As is shown in Table III, the moon does not face the same surface on the earth as do
a geosynchronous satellite in Dr. Glaser's original proposal, but the entire earth c
benefit from this electrical energy, and, as shown in Figure 5, three or more
rectennas would be located 120° or 90° apart in longitude on the earth surface, and
the U.S. would once again become an energy exporting nation. In order to eliminate
the possibility of accidentally scanning a high power microwave beam across the eart'
surface while switching between ground rectennas, I propose two transmitting antenna
on the moon, one aimed at one rectenna, and the second aimed at the next rectenna to
which power would be switched as it comes over the earth's horizon. While it is
transmitting power, each transmitting antenna, (whose diameter may have to be as lar
as 10 Km to achieve the required diffraction-limited microwave beam size) will be
scanned relative to the moon's polar axis by approximately 1° in a 6 or 8 hour
period. This can be practically accomplished by achieving the effective 10 KM
diameter with an array of smaller antennas ganged together, using the same principle:
as is used in large baseline radio astronomy antennas, or preferably, by increasing
the ground rectenna area, or increasing the 2.45 GHz transmitting frequency, or both.

Of course, the moon rotates relative to the sun also, so at least three such solar
collectors will be required, located 120° apart in longitude around the lunar pole.
At any time, the combined output of two collectors will be equal to or greater than
the peak output of one collector directly facing the sun. The distance between the
collectors is about 250 KM, a distance amenable to standard high voltage conductive
transmission, microwave beamed power transmission, or superconductor transmission.

The minimum 5 Gwatt input of this system at the power grid represents 46 Billion
kilowatt hours per year, which, at the current average price of ten cents/kwhr.,
represents a yearly revenue of $4.6 Billion per year. If we complete the "necklace"
around the lunar pole, the collection area increases to 355 km^2, and the net revenue
increases to to $29.7 billion per year. If we use both the north and south poles of
the moon the net revenue increases further to $59.4 Billion per year. Of course, on
the solar cell fabrication facilities on the moon are in operation, the collection
area can increase indefinitely.

However, I believe that the system shown in Figures 6 and 7 represents a more effi-
cient and lower-cost-per-kilowatt design than the reference system defined by the DO
and NASA studies, which more efficiently uses the attributes of the lunar platform.

Studies of Stirling cycle solar energy conversion systems, conducted at Sandia and a
NASA Lewis Research Center have shown that end-to-end solar-to-electricity efficien-
cies of higher than 30%, or twice the efficiency of the photovoltaic reference
approach are feasible with high gain solar concentrators. The system shown in Figur
6, which girds the lunar equator, uses the lunar gravity and windless environment to
achieve a very low cost cylindrical concentrator composed entirely of aluminized
polyester hanging in a catenary, (which closely approximates a parabola) focusses the
sun on a linear series of pipes containing the Stirling cycle gas. This gas is heate
to increase its pressure, thus driving the Stirling engines distributed along this
linear concentrator, which generate the electricity. No solar tracking is required
because it is an optical characteristic of a cylindrical mirror that its line focus i
maintained independent of the incident angle of the incoming rays in the plane of the
cylindrical axis, and the maximum ± 5° angular variation of the sun in the orthogonal
plane is accommodated by the width of the black pipe collector. The availability of
an unlimited heat sink less than 100°K on the dark side of the moon, allows even

TABLE II

TECHNICAL ADVANTAGES OF A LUNAR-BASED SPS

1. THE MOON IS A MUCH MORE RELIABLE AND STABLE PLATFORM THAN A GEOSYNCHRONOUS SPACECRAFT.

2. THE MOON HAS UNLIMITED EXPANSION SPACE FOR ADDITIONAL SOLAR POWER CAPACITY.

3. THERE IS NO INTERFERENCE WITH VISIBLE ASTRONOMY DUE TO SPECULAR SUNGLINTS OFF THE SOLAR COLLECTORS, AS THERE IS WITH THE GEOSYNCHRONOUS SYSTEM.

4. RAW MATERIALS FOR INCREASED SOLAR COLLECTION CAPACITY ARE AVAILABLE DIRECTLY FROM THE MOON AND DO NOT HAVE TO BE LIFTED FROM THE EARTH.

5. COLLECTOR AND TRANSMITTING ANTENNA ASSEMBLY IS MUCH SIMPLER AND CHEAPER UNDER LUNAR GRAVITY THAN IN AZERO G ENVIRONMENT.

6. LOCATION OF THE COLLECTORS NEAR THE LUNAR POLE(S) PROVIDES:

   ● THE NATURAL ARRANGEMENT FOR A CATENARY (PARABOLIC) MIRROR OPTICAL CONCENTRATOR BELOW A NEAR VERTICAL COLLECTOR.

   ● THE COLD AVERAGE LUNAR SURFACE TEMPERATURE NEAR THE POLES MAKES AN EXCELLENT HEAT SINK FOR THE PHOTOVOLTAIC SOLAR ARRAYS, THUS MAKING THEM MORE EFFICIENT.
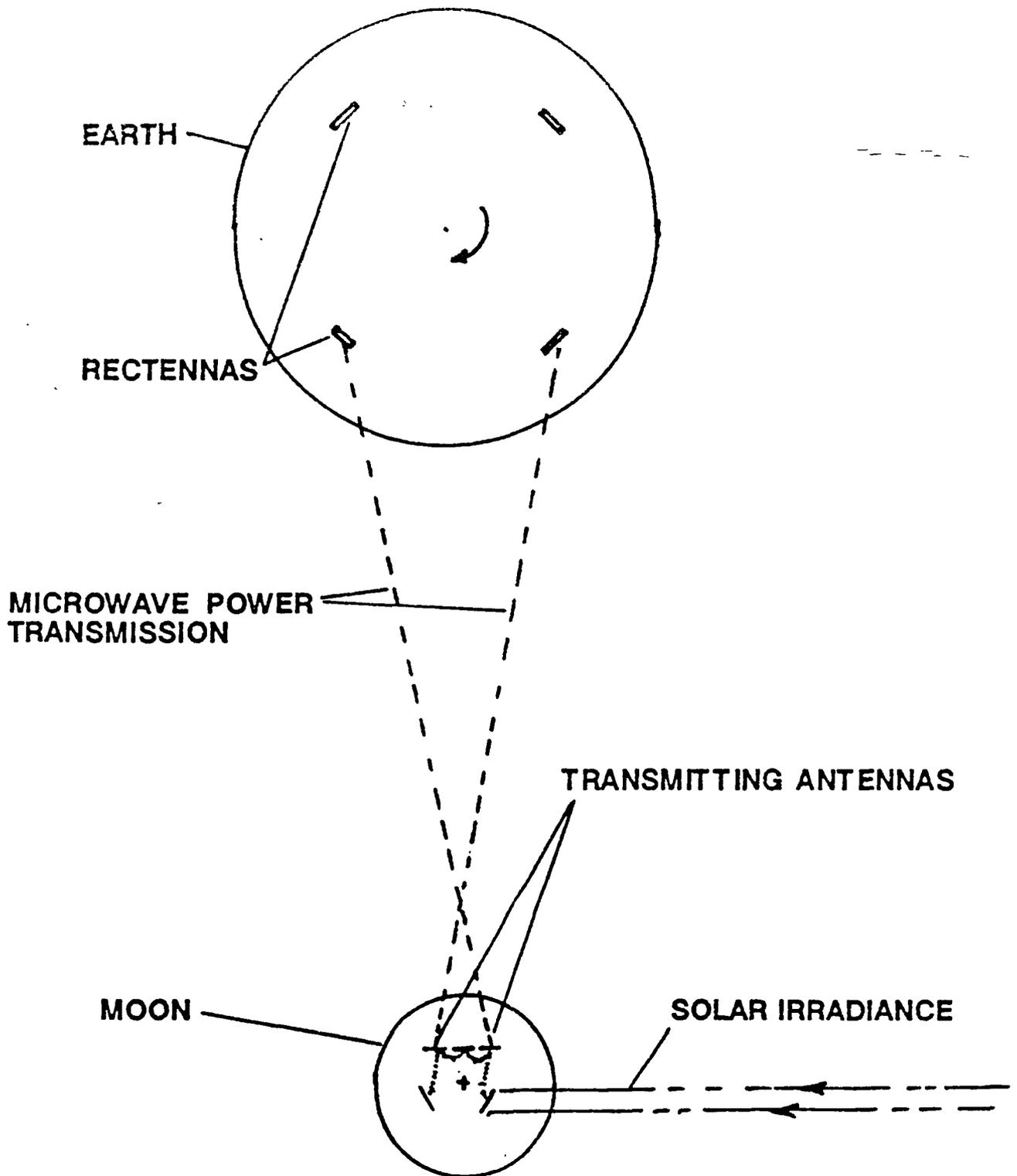
TABLE III

TECHNICAL DISADVANTAGES OF LUNAR SPS VS. GEOSYNCHRONOUS SPS

1. EARTH ROTATES UNDERNEATH TRANSMITTING ANTENNA

2. MOON IS APPROXIMATELY TEN TIMES FARTHER THAN GEOSYNCHRONOUS ALTITUDE, THUS REQUIRING TEN KM DIAMETER TRANSMITTING ANTENNA TO ACHIEVE THE SAME EARTH RECEIVING RECTENNA DIFFRACTION-LIMITED DIMENSIONS.

3. WITHOUT TRANSPONDING TRANSMITTING SATELLITES, THERE IS A DISCONTINUOUS POWER INPUT TO THE GRID AT ANY ONE EARTH LOCATION.

PROPOSED SOLUTIONS

1. LOCATE RECTENNAS AT THREE OR FOUR LONGITUDES ON THE EARTH, PROVIDING INTERNATIONAL POWER INPUT CONTINUOUSLY TO THE EARTH. (U.S. AGAIN BECOMES ENERGY EXPORTING NATION.)

2. DESIGN A LARGE BASELINE GANGED ARRAY OF SMALLER TRANSMITTING ANTENNAS, OR INCREASE THE RECTENNA DIMENSIONS, OR INCREASE THE TRANSMITTING FREQUENCY TO > 2.45 GHZ., OR ALL OF THE ABOVE.

3. ADD TWO TRANSPONDER SATELLITES, OPERATING 120° APART IN THE LUNAR ORBIT, AND ADD TWO MORE TRANSMITTING ANTENNAS ON THE MOON. (THESE TRANSPONDING SATELLITES WOULD BE FABRICATED ON THE MOON AND LAUCHED INTO ORBIT FROM THE MOON.)

EARTH

RECTENNAS

MICROWAVE POWER
TRANSMISSION

TRANSMITTING ANTENNAS

MOON

SOLAR IRRADIANCE

FIGURE 5 POWER TRANSMISSION FROM THE MOON TO EARTH
(HALF MOON CONDITION)

SUN ANGLE VARIATION
OVER YEAR

10°

RADIATION
SHIELDS
(HOT SIDE)

BLACK
RADIATORS
(COLD SIDE)

COLLECTOR PIPES

ROLLERS FOR ALUMINIZED
POLYESTER TO EXPOSE
NEW SURFACE, REPLACING
MICROMETEORITE DAMAGED
REFLECTOR

1.3 KM

LUNAR SURFACE

ALUMINIZED POLYESTER IN RELAXED POSITION TO SHUT OFF SOLAR
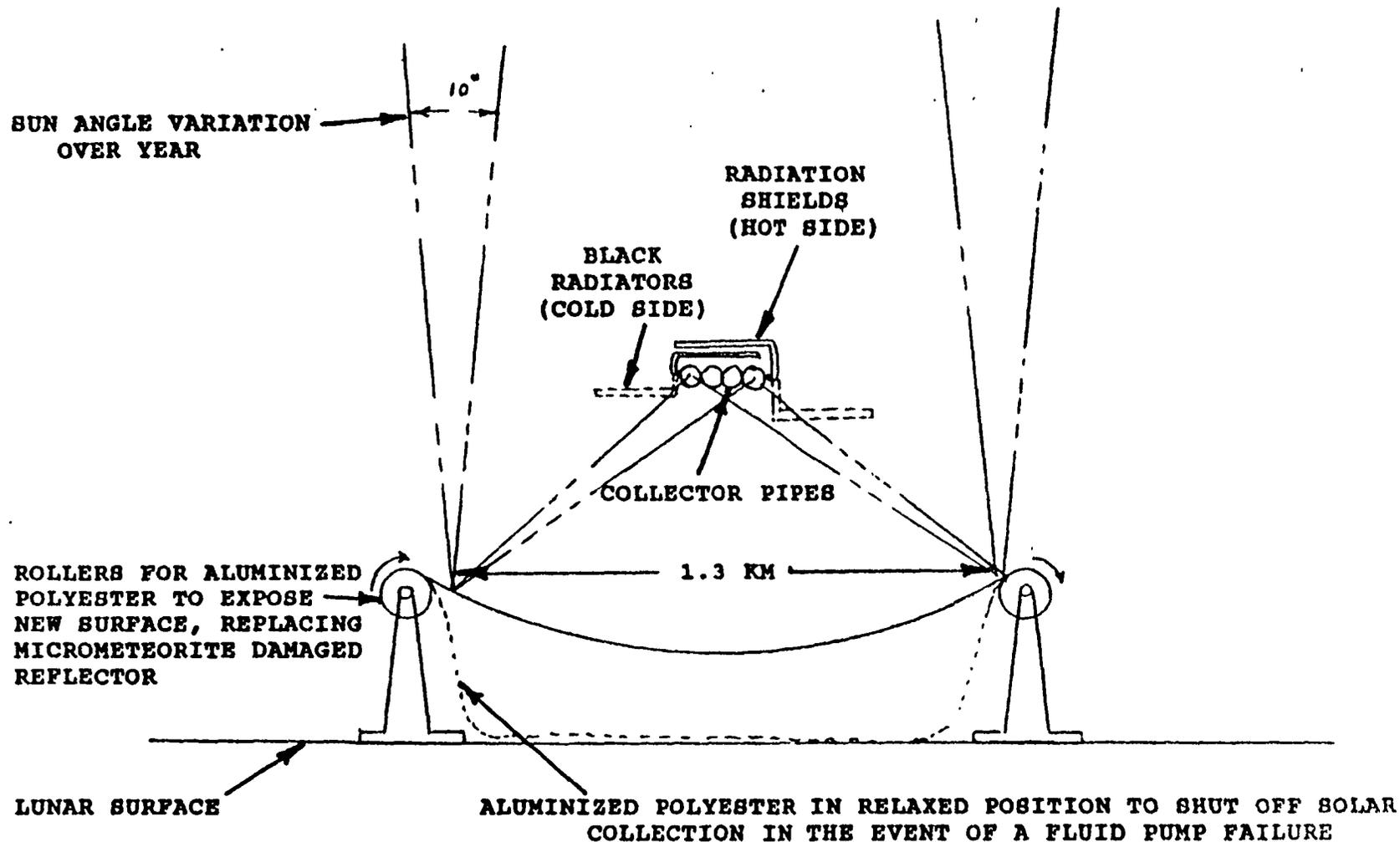COLLECTION IN THE EVENT OF A FLUID PUMP FAILURE

FIGURE 6   OPTIMUM LUNAR SOLAR COLLECTOR FOR EQUATORIAL MOUNTING

higher efficiencies for the Stirling cycle than have heretofore been achieved. The use of heat pipes to conduct the low temperature gases in the Stirling cycle to this unlimited heat sink should be able to achieve these high efficiencies. The equatorial location results in 10,000 km. of collector length, with 3,330 km^2 of net collecting area at any one time. As is shown in Figure 7, since the Stirling cycle efficiency is approximately twice that of the photovoltaic cell configuration, this system produces 5.31 Trillion kwhrs per year, generating a revenue of $558 billion.

In this ultimate system, the power distribution to the earth should be maintained 24 hours a day. This is accomplished through the use of two "transponder" satellites operating at libration points 120° apart in the lunar orbit, as shown in Figure 8. These transponder satellites would consist of rectennas receiving microwave energy from the moon, and two transmitting antennas performing the same function as the transmitting antennas to the earth from the lunar surface described above. These transponder satellites would be fabricated on the lunar surface, assembled in lunar orbit, and located and maintained in their orbital positions using electrically driven ion engines driven from electricity received by their rectennas.

Upon completion and expansion of this ultimate system, the revenues could be used to retire the national debt and ultimately replace the income tax with an electric bill equal to what they are already paying.

Therefore, this ultimate system can be said to give the American electorate what in recent years they have shown that they want, namely, a free lunch!

Now I will discuss how I propose to obtain the seed money to pay for this project. (Once the revenues from the electric power begin, these revenues can be fed back into the system to expand it, without further tax based input).

I propose to fund these intial phases of this project with, read my lips: No new taxes!

In order to accomplish this seeming feat of magic, we must first acknowledge that the "Emperor has no clothes"! By this I mean that we must finally acknowledge that for forty years the U.S. defense budget, having a cumulative total of over $4 trillion, has incorporated a large percentage of "make work" WPA programs for the Military Industrial Complex.

For those of you not familiar with "WPA", it stands for Works Progress Administration, a New Deal program initiated during the Great Depression to get the unemployed back to work on government sponsored programs. WPA paid artists to paint murals, laborers to replant forests, and contractors to build highways, such as the Merritt Parkway in Connecticut on which I drive every day. This parkway includes beautiful, artistic overpass bridges incorporating different and original artistic frescoes, which were the S level components of the thirties.

In fact, this useful version of WPA is alive and well today and residing in Japan. We call it "Japan Incorporated". If you harbor any remaining doubts relative to this WPA thesis, I will now dispel them with these two recent news items:

1. That Superhawk, Senator Edward Kennedy, is reported to be in favor of continuing to build the unneeded Sea Wolf Submarines.

2. The House and Senate Democrats have now passed a budget allocation continuing the fabrication of the second Sea Wolf Submarine over the objection of that Superdove, President Bush.

$$\underline{\text{COLLECTOR AREA}} \text{ X NET KWATTS COLLECTED BY REFERENCE SYSTEM X } \underline{\text{STIRLING EFFICIENCY}}$$
$$\text{REFERENCE AREA} \qquad\qquad\qquad\qquad\qquad\qquad\qquad\qquad \text{PHOTOVOLTAIC EFFICIENCY}$$

$$\frac{3,330 \text{ KM}^2}{55 \text{ KM}^2} \quad X \quad 5 \text{ BILLION} \quad X \quad \frac{2}{1} = 606 \text{ BILLION KW}$$

YEARLY ENERGY VALUE = 606 BILLION KW X 8760 HOURS/YEAR X $.10/KWHR

= **$531 BILLION**

FIGURE 7 ENERGY CALCULATION FOR THE ULTIMATE LUNAR SPS

LUNAR ORBIT

TRANSPONDER
AT 120° LIBRATION POINT

MOON

EARTH

BEAMED MICROWAVE
ENERGY
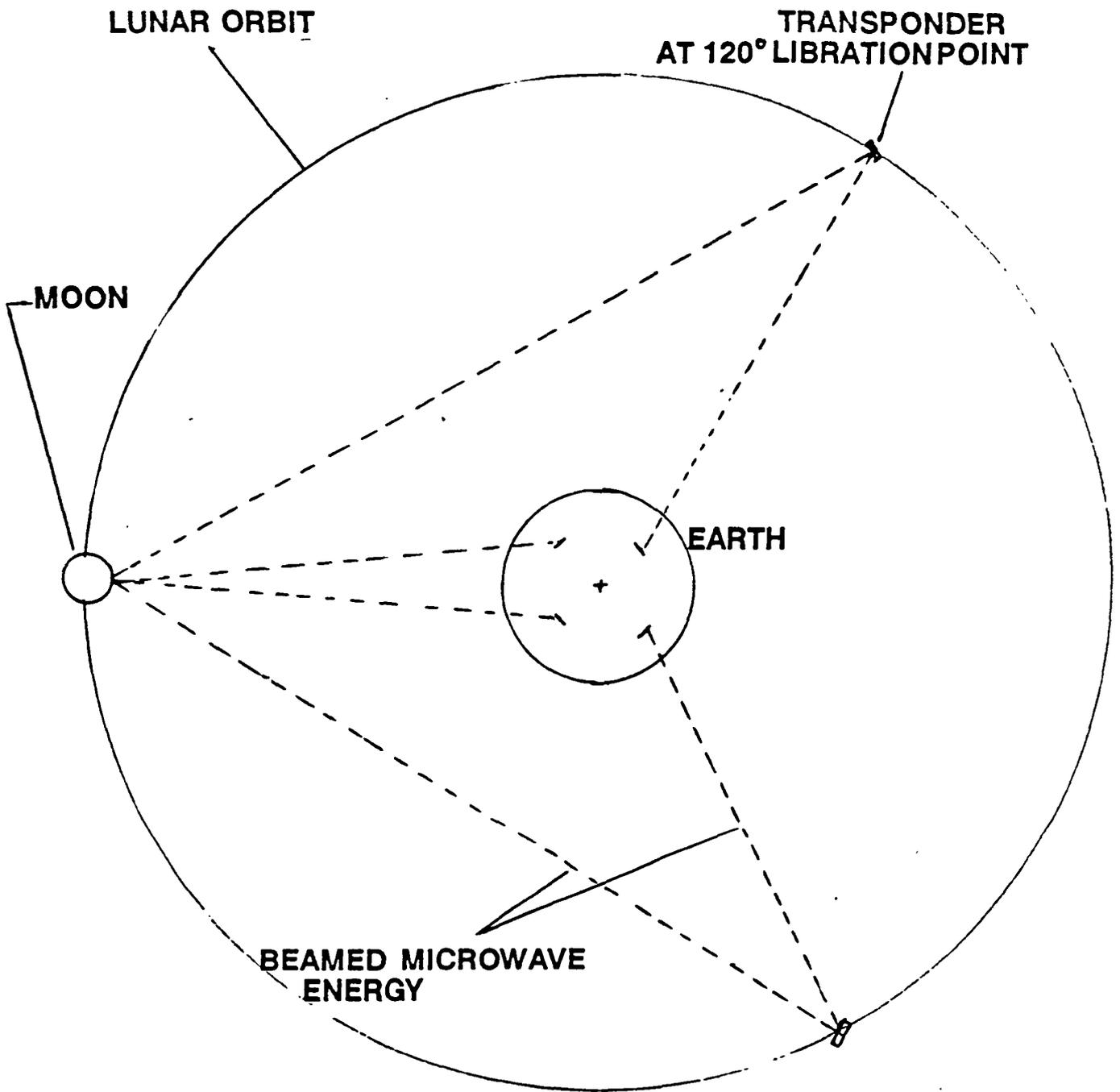
FIGURE 8 CONTINUOUS WORLD WIDE POWER TRANSMISSION

How would we transfer the funds presently going to useless defense programs to a lunar Solar Power Satellite? We would simply convert the defense contracts covering technical disciplines similar to those in the Solar Power Satellite, with the management of these contracts being retained by the present military responsible personnel. Therefore, no-one from the Defense Establishment would lose his or her job.

If you think this is impractical, I can recall for you two precedents:

1. General Leslie Groves, successfully managed the Manhattan Project for the development of the Atomic Bomb.

2. Rockwell International stipulated that its subcontractors for the B-1 bomber must come from all 50 states.

For over 12 years I have written letters expounding this misappropriation of U.S. engineering talent to U.S. Presidents, Senators, Congressmen, journalists, Presidential candidates, and Presidents Gorbachev and Yeltsin. I have never received a single direct response to any of these letters, nor any refutation of my arguments. This indicates to me the cataclysmic lack of the "Vision Thing" or in both the Executive and Legislative branches of our government.

In closing, I would like to read two quotations from these letters:

From a letter to President Gorbachev:

"When you address the "Military Industrial Complex", your attitude is that it is the incarnation of evil, intent only upon destroying the Soviet Union by military threats or application of military advantage. Since I am a member of the "Military Industrial Complex", I assure you that for me, or any one I know, this is not the case.

However, when it comes to putting bread on the table, most people (including me) will justify their work in the Defense Establishment as assigned tasks that must be done properly and whose parochial justifications may even be advocated so that work accomplished on an ongoing project is not "wasted" by its cancellation. I submit to you that this insidious "work ethic" is the underlying cause of the Arms Race, to a much greater extent than any military or political necessities."

From a letter to President Reagan:

"It might be of interest for you to know that I am, by profession, an electro-optical engineer, whose hey-day would come with the widespread implementation of the systems in "Star Wars". However, I consider such work a prostitution of my art, which, unfortunately, I, and a majority of my professional associates, do as a matter of economic survival."

This leads me to my final manifesto:

Engineers of the world, unite! You have nothing to lose but your net stockings and your hot pants!

"Fast Track" Lunar NTR Systems Assessment
for the
First Lunar Outpost and its Evolvability to Mars

Presented
by

Dr. Stanley K. Borowski
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to

SEI 3rd Technical Interchange Meeting
University of Houston - Clear Lake

May 5 & 6, 1992

"FAST TRACK" LUNAR NTR/STAGE ANALYSIS

OBJECTIVE: QUANTIFY NECESSARY ENGINE/STAGE CHARACTERISTICS TO PERFORM NASA'S "FIRST LUNAR OUTPOST" SCENARIO AND ASSESS THE POTENTIAL FOR EVOLUTION TO MARS MISSION APPLICATIONS

GOAL: ESTABLISH FEASIBILITY/ATTRACTIVENESS OF A COMMON LUNAR/MARS SPACE TRANSPORTATION SYSTEM BASED ON MODULAR NTR/STAGE COMPONENTS

RATIONALE: BY DEVELOPING NTR/STAGE TECHNOLOGIES FOR USE IN NASA'S "FIRST LUNAR OUTPOST" SCENARIO, NASA WILL MAKE A MAJOR DOWN PAYMENT ON THE KEY COMPONENTS NEEDED FOR THE FOLLOW-ON MARS SPACE TRANSPORTATION SYSTEM. A FASTER, CHEAPER APPROACH TO OVERALL LUNAR/MARS EXPLORATION IS EXPECTED

Fast Track NTR Assessment Team

- **NASA**

  - LeRC
    - ✳ • **Nuclear Propulsion Office**
    - ✳ • **Advanced Space Analysis Office**
    - • **Procurement Division**
    - • **Instrumentation and Controls Division**

    - • **Space Propulsion Technology Division**
    - • **Resources Analysis and Management Ofc.**
    - ✳• **Cryogenic Fluid Technology Office**
    - • **Safety Assurance Office**

  - MSFC
    - • **Program Development**
    - • **Research and Technology Office**

    - • **Propulsion Laboratory**

- **DOE**
  - Idaho National Engineering Lab (INEL)
  - ✳- Los Alamos National Lab (LANL)
  - Sandia National Lab (SNL)

  ✳ Indicates Organizations involved in Engine/System Analysis & Design

- **Industry**
  - ✳- Rocketdyne
  - Analytical Engineering Corporation
  - ✳- General Dynamics

  - Aerojet
  - Sverdrup
  - ✳- Westinghouse

NERVA-Derivative Engine Parameters

Parameters	Graphite	Composite
T_{ex} (°K)	2550	2700
p_c (psia)	785	785
ε (-)	200:1	200:1
Isp (s)	870	915
M_{engine} (kg)*		
- 25 klbf	3830	3966
- 50 klbf	5601	6009
- 75 klbf	6863	7361

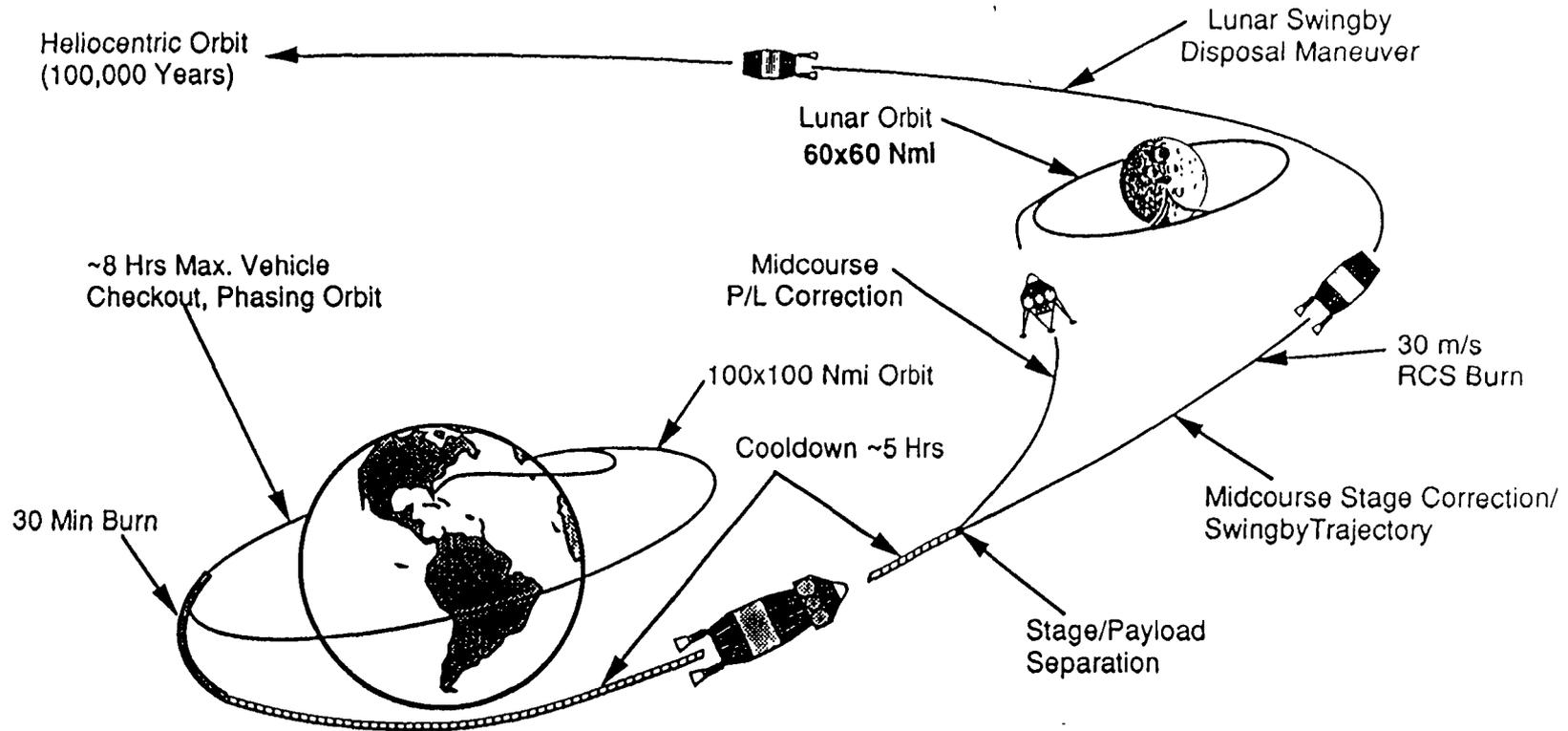* Includes internal dome shield and dual turbopumps

"Assumed" Engine Operating Ranges/Constraints

- **Engine thrust levels** : **10 klbf - 75 klbf**

- **Baseline fuel form** : **Coated-UC$_2$ particles in graphite (UC-ZrC) C "composite" (backup)**

- **Max. Single Burn Duration*** : **30 minutes/engine**

- **Max. Total Burn Duration** : **≤ 60 minutes/engine (lunar)**
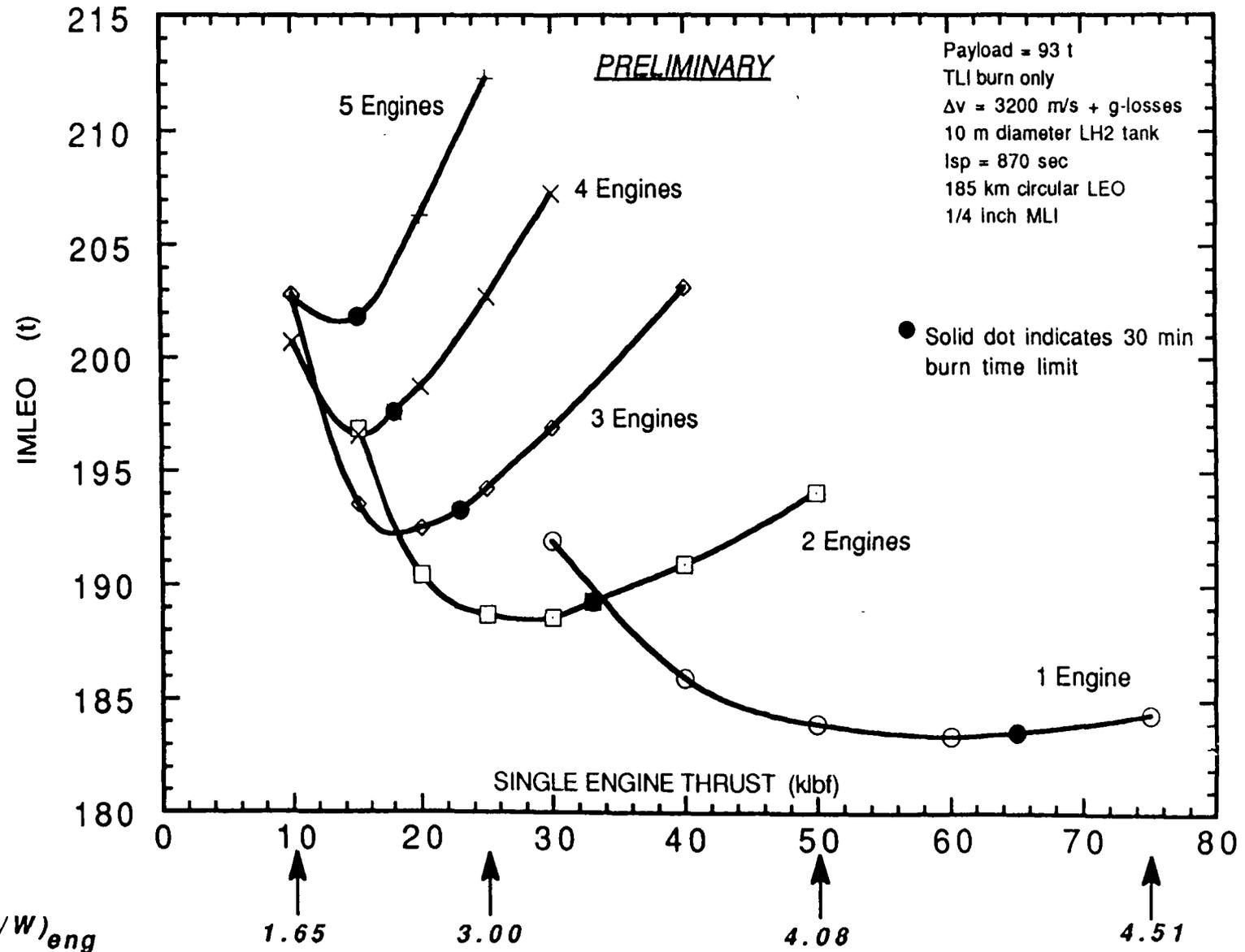 : **≤ 120 minutes/engine (Mars)**

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*The maximum single burn time specified to provide enhanced mission success probability and to provide margin (in terms of total burn duration) for the remaining engine(s) in case of "engine-out" occurrence.

"Fast Track" Lunar NTR Mission Scenario
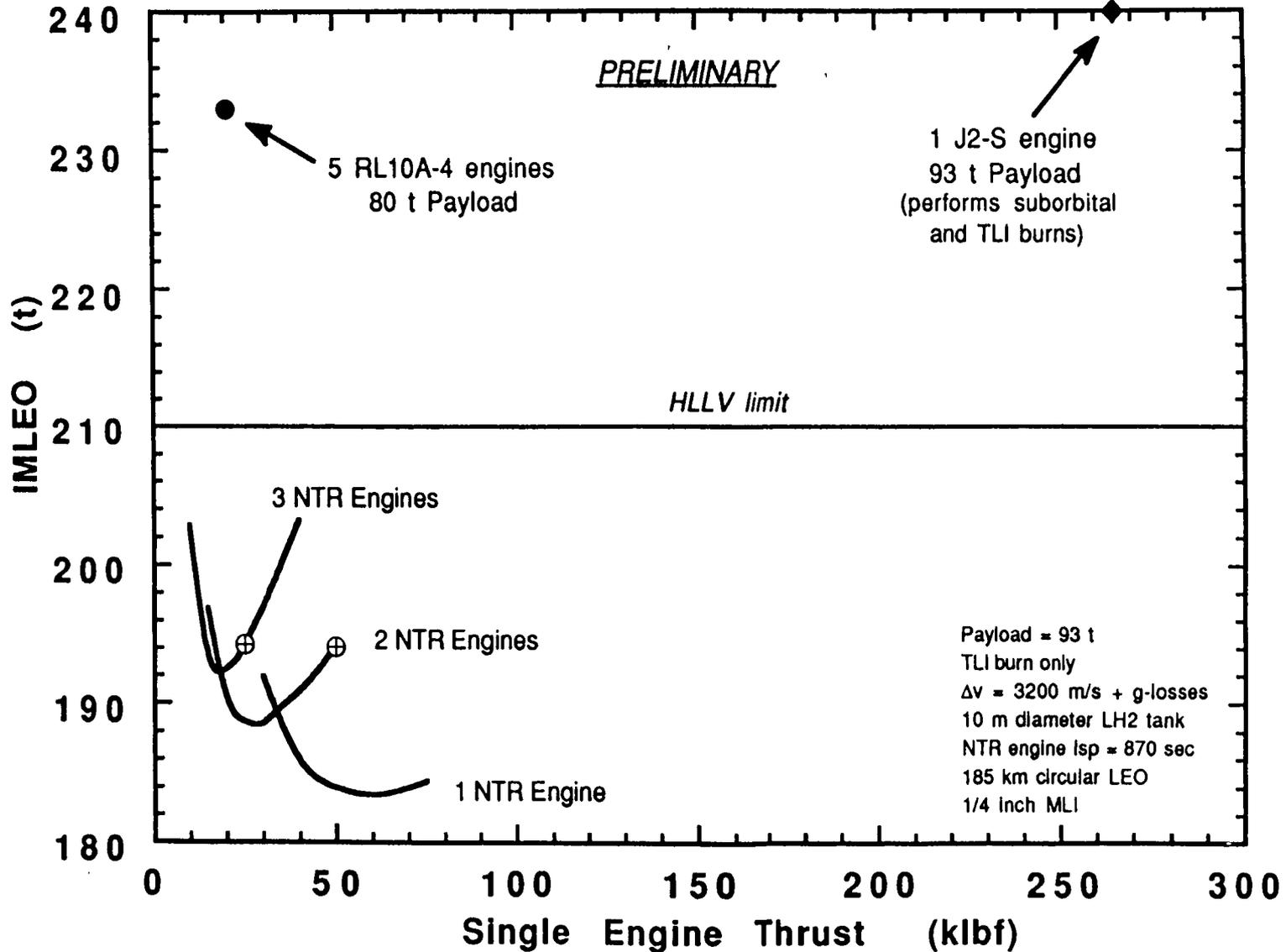## (Mission Profile Analogous to FLO)

Heliocentric Orbit
(100,000 Years)

Lunar Swingby
Disposal Maneuver

Lunar Orbit
60x60 Nml

~8 Hrs Max. Vehicle
Checkout, Phasing Orbit

Midcourse
P/L Correction

30 m/s
RCS Burn

100x100 Nmi Orbit

Cooldown ~5 Hrs

Midcourse Stage Correction/
SwingbyTrajectory

30 Min Burn

Stage/Payload
Separation

- **Single TLI Burn**
- **Cryogenic NTR TLI Stage Requires Minimal Insulation**
- **Payload Separation before Mid-course Correction**

Fast Track Lunar NTR (93 t to TLI)

NASA — LEWIS RESEARCH CENTER

PRELIMINARY

Payload = 93 t
TLI burn only
Δv = 3200 m/s + g-losses
10 m diameter LH2 tank
Isp = 870 sec
185 km circular LEO
1/4 inch MLI

● Solid dot indicates 30 min burn time limit

5 Engines
4 Engines
3 Engines
2 Engines
1 Engine

IMLEO (t)

SINGLE ENGINE THRUST (klbf)

$(T/W)_{eng}$

1.65   3.00   4.08   4.51

NUCLEAR PROPULSION OFFICE

NTR is Enabling for FLO

First Lunar Output: 93t to TLI and 210 t HLLV

PRELIMINARY

1 J2-S engine
93 t Payload
(performs suborbital
and TLI burns)

5 RL10A-4 engines
80 t Payload

IMLEO (t)

HLLV limit

3 NTR Engines

2 NTR Engines

1 NTR Engine

Payload = 93 t
TLI burn only
Δv = 3200 m/s + g-losses
10 m diameter LH2 tank
NTR engine Isp = 870 sec
185 km circular LEO
1/4 inch MLI

240

230

220

210

200

190

180

0   50   100   150   200   250   300

Single Engine Thrust (klbf)

Payload	93 t to TLI
Tank	10 m X 14.8 m
Insulation	1/4 inch MLI
Propellant	67 t
Propulsion	3 X 25 klbf NDR
	870 sec
IMLEO	194 t

Benefits of Modular, NTR-Based Lunar/Mars Space Transportation System

- Past/On-going studies* conducted by the NPO indicate that NTR propulsion is also <u>enabling</u> for lunar applications when demanding mission scenarios and/or significant payload delivery capability is being contemplated.

- Study results also indicate the attractiveness of evolving a "proven" lunar NTR transportation system for Mars application through the addition of a "modular propellant tank" for use either as a structural and/or drop tank.

- By using this "building block" approach, a variety of lunar/Mars vehicles can be configured to meet specific mission requirements.

- Analysis is proceeding aimed at quantifying vehicle building block and "best compromise" engine characteristics based on current ExPO mission requirements and a range of HLLV assumptions.

*References

(1)   S. K. Borowski, "The Rationale/Benefits of Nuclear Thermal Rocket Propulsion for NASA's Lunar Space Transportation System," AIAA paper 91-2052, 27th Joint Propulsion Conference, Sacramento, CA, June 24-26, 1991.

(2)   S. K. Borowski, "An Evolutionary Lunar-to-Mars Space Transportation System Using Modular NTR/Stage Components," AIAA paper 91-3573, Conf. on Advanced SEI Technologies, Cleveland, OH, Sept. 3-6, 1991.
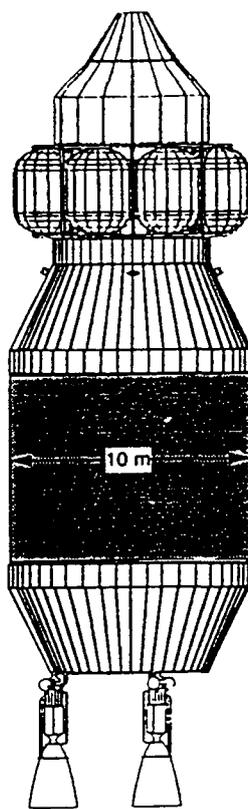
Four Burn Reusable Lunar NTR Vehicle

Comparison of Lunar NTR Vehicles

"90 Day Study"
Fully Reusable Vehicle

Cargo

Cargo

←— 10.0m —→
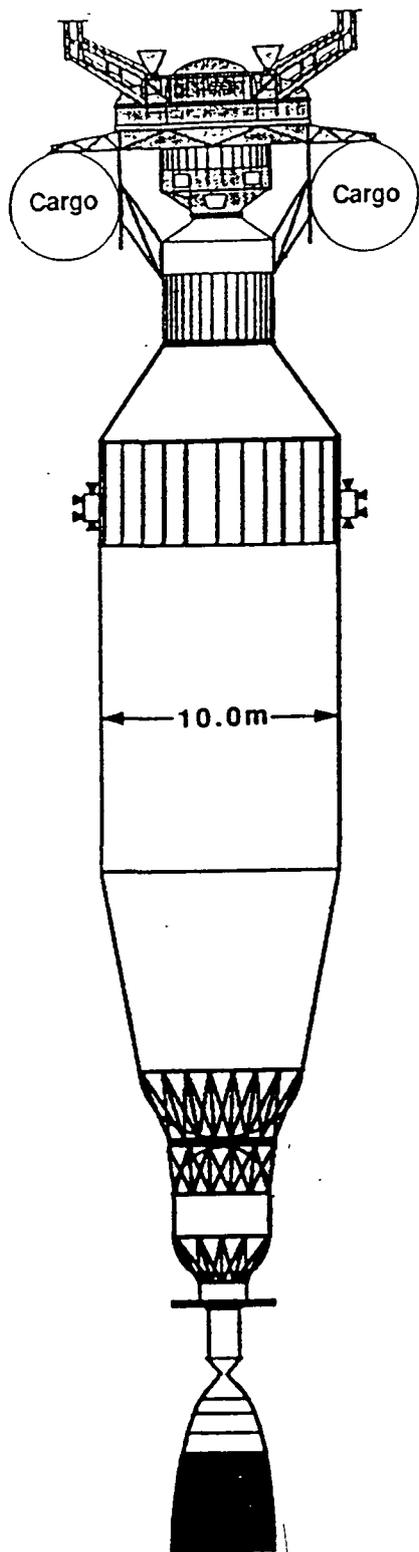
"First Lunar Outpost"
Expendable TLI Stage

Permanent Lunar Base
Reusable Vehicle

10 m

14 m

3 X 25 klbf Engines

2 X 50 klbf Engines

1 X 75 klbf Engine

LEWIS RESEARCH CENTER
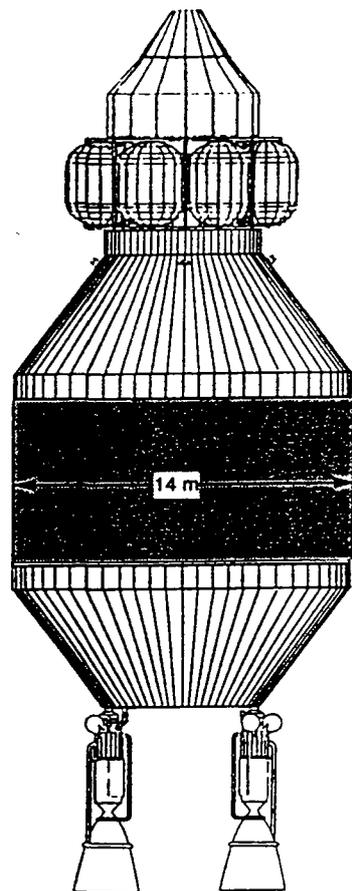
2005 MARS CARGO VEHICLES
Single Tank

PRELIMINARY

3 Engines
Single Burn TMI

2 Engines
Single Burn TMI

1 Engine
Single Burn TMI

3 X 40 klbf
Triple Burn TMI

IMLEO (t)

2005 Mars Cargo Vehicle (81.1 t Payload)
Single 14 X 17.25 m Tank, No External Shield
Single Perigee Burn and Triple Perigee Burn TMI
407 km circular LEO (w/gloss)
250 X 1 sol Mars Orbit
Isp = 915 sec

⊕ 2 X 50 klbf
Triple Burn TMI

Single Engine Thrust (klbf)
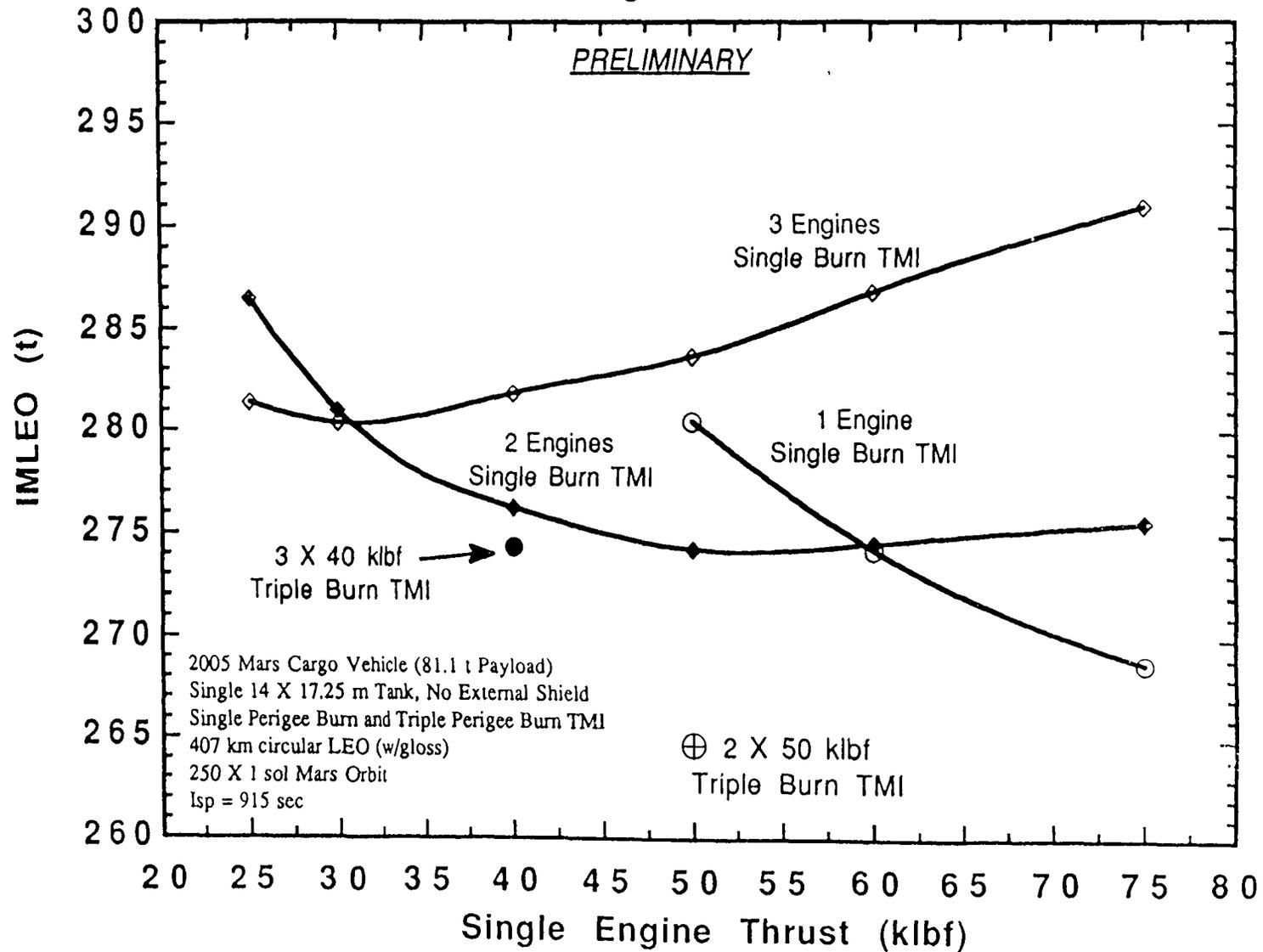
MARS NTR Vehicle "Building Block" Assumptions

- **Modular components** are sized to match boost capability of planned HLLV's (233t to 220 nmi w/ 14m x 30m payload volume) to reduce launch cost/maximize performance

- **Main "Core" Propulsion Module**
  Includes:
  - NTR Cluster
  - Tank Structure
  - Contingency (engine/shields: 15%; remainder: 10%)
  - Interstage/Docking Structure
  - Thermal/micro Protection
  - External Shields
  - RCS

  - Dimensions: 14m x 17.25m length (cylindrical tank w/ $\sqrt{2}/2$ ellipsoidal bulkheads)
  - Total Mass: 233t (41.32 t hardware; 148.7t LH_2; 15t NTO/MMH)

- **"Modular" Tanks**
  Includes:
  - Tank Structure
  - Thermal/micro Protection
  - Interstage/Attachment Structure
  - Contingency (10%)

  - Dimensions: 14m x 12.5m length (cylindrical tank w/ $\sqrt{2}/2$ ellipsoidal bulkheads)
  - Total Mass:
    - Drop Tanks: 18.8t (w/ 2" MLI) and 98t LH_2
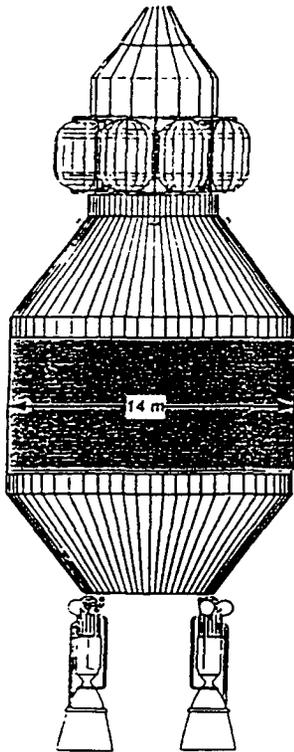   - "In-Line" Tank: 21.9t (w/ 4" MLI & VCS) and 98t LH_2

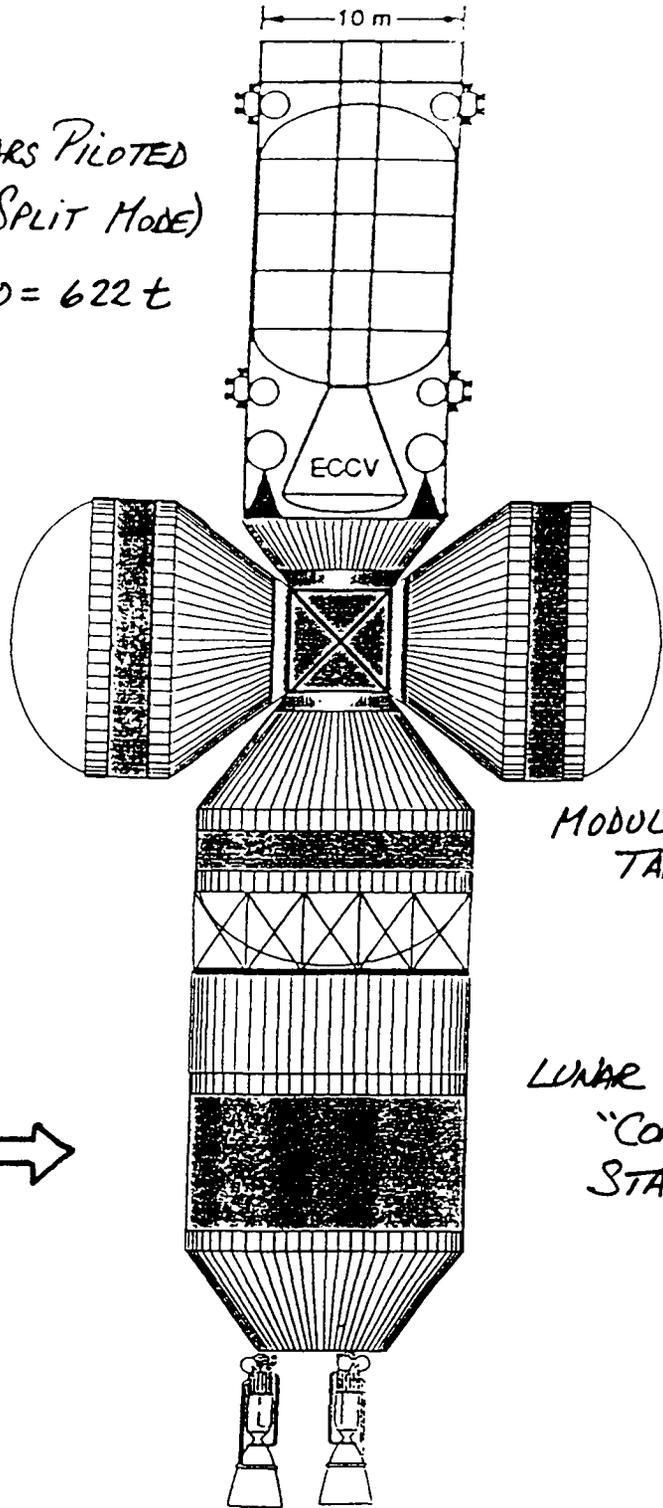# Comparison of NTR Vehicles

(Isp = 915 s)

⊢— 10 m —⊣

*2007 Mars Piloted
("No MEV" Split Mode)*

$IMLEO = 622 t$

ECCV

*Reusable Lunar
or
Expendable 2005 Mars
Cargo*

14 m

Modular LH₂ Tanks

Lunar Vehicle "Core" Stage

2 X 50 klbf Engines

3 X 50 klbf Engines

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435

Summary/Conclusions

- NTP **enables** 93 t to TLI (cryo/storable lander option) for the "First Lunar Outpost" within original HLLV limits (200-210 t)

- LeRC has performed an initial assessment of the performance capability for single and multi-engine NTP concepts over a wide range of thrust levels, HLLV configurations (10 and 14 m diameter), and mission profiles.

- Mission Profiles examined to date:
    - Expendable TLI (One Burn) Mission
    - Reusable Lunar (Four Burn) Mission
    - 2005 Mars Cargo Mission
    - 2007 Mars Piloted Mission

- Preliminary results indicate that the 50 klbf engine is the "best compromise" engine based upon IMLEO and burn time considerations.

- Initial analysis indicates both the attractiveness and feasibility of developing a "single, modular, NTR-Based" Space Transportation System (STS) for Lunar/Mars missions.

- The implementation of a NTR-Based Lunar STS will position NASA favorably for Mars missions in the 2005-2010 time frame. An **accelerated schedule** and **reduced cost** is anticipated for the Lunar/Mars STS because of a single development effort.

Nuclear Thermal Rocket by 2000: A DOE Perspective

By

Steven D. Howe

Los Alamos National Laboratory

and

Marland Stanley

Idaho National Engineering Laboratory

For the

3rd SEI Technical Interchange Meeting

May 5, 1992

*LOS ALAMOS NATIONAL LABORATORY*
Space Nuclear Propulsion Technology

N92-33335

PREMISE

- A NUCLEAR PROPULSION SPACE TRANSPORTATION SYSTEM IS REQUIRED FOR THE MANNED MARS MISSION AND CAN SUPPORT A WIDE VARIETY OF FUTURE SPACE MISSIONS.

- AN NTR STAGE CAN SIGNIFICANTLY BENEFIT LARGE- SCALE LUNAR BASE IMPLEMENTATION AND SUPPORT.

- THE ROVER/NERVA PROGRAM DEMONSTRATED THAT A SAFE, RELIABLE NTR CAN BE DEVELOPED AND OPERATED FOR SUFFICIENT RUN TIMES, AT DESIRABLE TEMPERATURES, AND WITH MULTIPLE RESTARTS.

- OTHER REACTOR CONCEPTS AND FUEL FORMS SHOULD BE INVESTIGATED AS THE SCHEDULE PERMITS AND WOULD REQUIRE APPROPRIATE EXPERIMENTAL VALIDATION BEFORE SUPPLANTING THE BASELINE TECHNOLOGY

- REFURBISHING EXISTING FACILITIES MAY REDUCE COST AND SHORTEN SCHEDULE SIGNIFICANTLY.
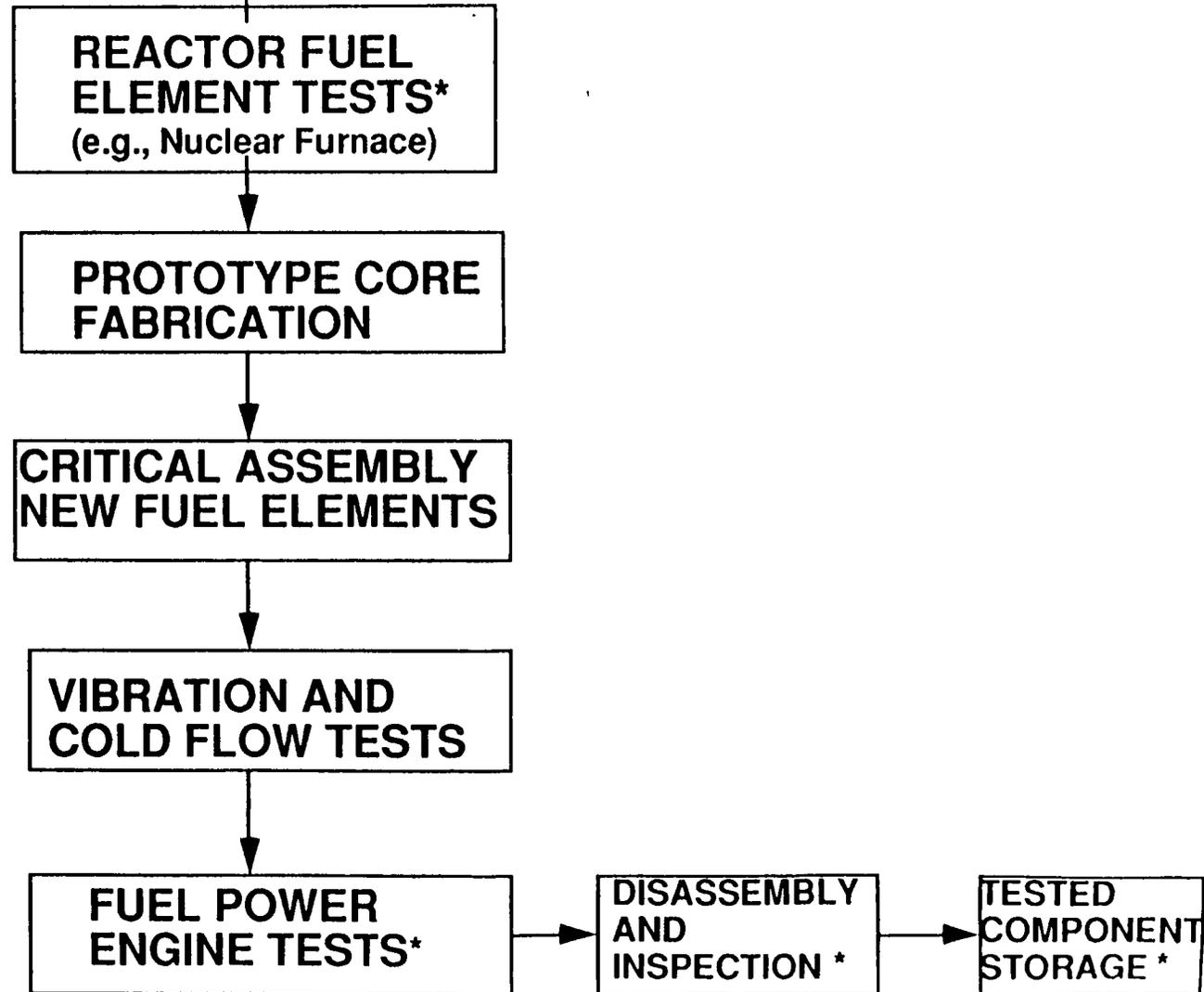
FUEL DEVELOPMENT PATH *
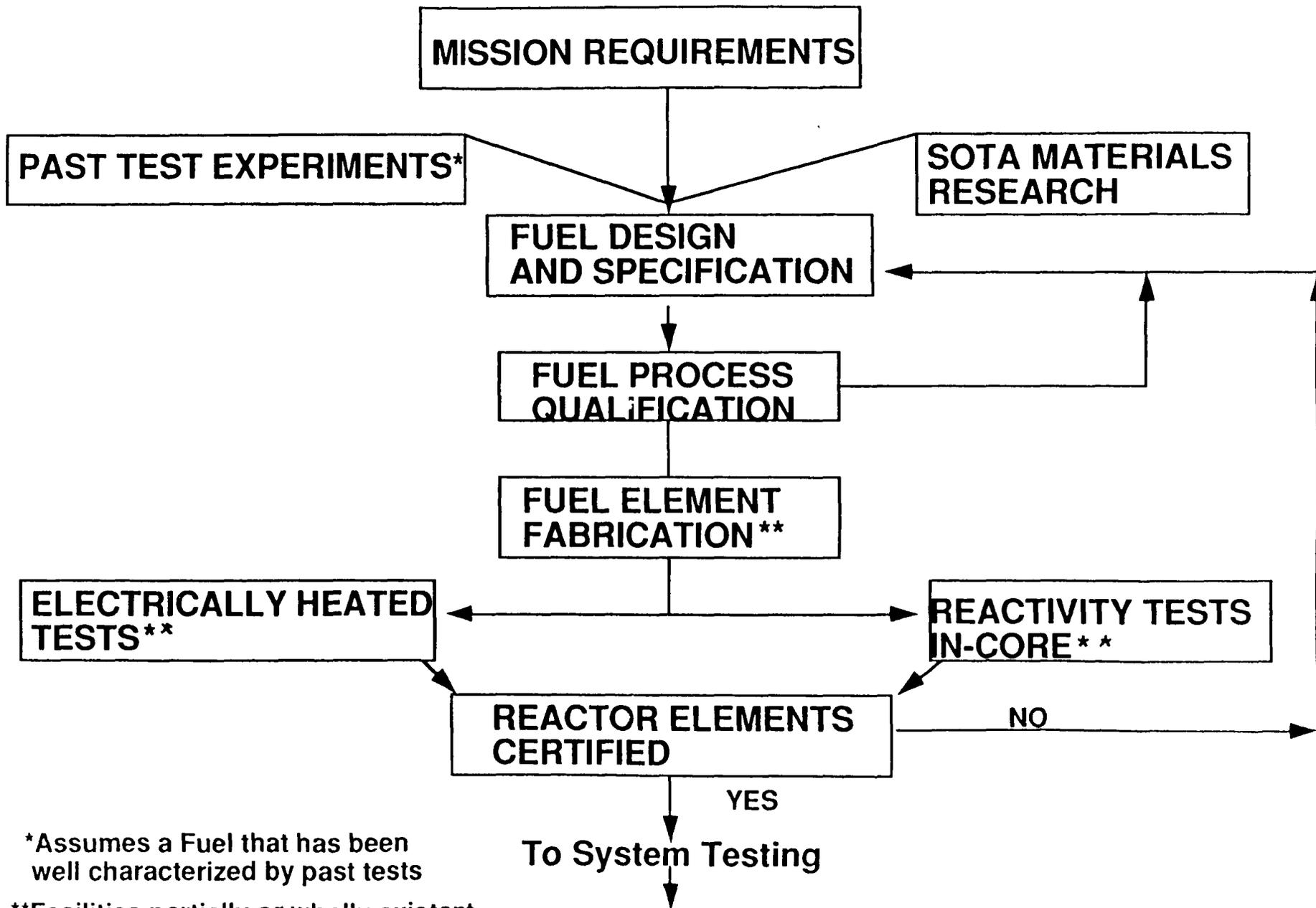
*Assumes a Fuel that has been
   well characterized by past tests

**Facilities partially or wholly existant

439

SYSTEM TESTING PATH

FROM FUEL DEVELOPMENT

```
┌─────────────────────────┐
│ REACTOR FUEL            │
│ ELEMENT TESTS*          │
│ (e.g., Nuclear Furnace) │
└─────────────────────────┘
             │
             ▼
┌─────────────────────────┐
│ PROTOTYPE CORE          │
│ FABRICATION             │
└─────────────────────────┘
             │
             ▼
┌─────────────────────────┐
│ CRITICAL ASSEMBLY       │
│ NEW FUEL ELEMENTS       │
└─────────────────────────┘
             │
             ▼
┌─────────────────────────┐
│ VIBRATION AND           │
│ COLD FLOW TESTS         │
└─────────────────────────┘
             │
             ▼
┌─────────────────────┐   ┌──────────────┐   ┌─────────────┐
│ FUEL POWER          │──▶│ DISASSEMBLY  │──▶│ TESTED      │
│ ENGINE TESTS*       │   │ AND          │   │ COMPONENT   │
│                     │   │ INSPECTION * │   │ STORAGE *   │
└─────────────────────┘   └──────────────┘   └─────────────┘
```

*Potential refurbishable sites exist

SUMMARY

- NASA AND DOE TEAMS HAVE RECENTLY INVESTIGATED AN "NTR FAST TRACK" PROGRAM PLAN TO DEVELOP A FLIGHT- READY ENGINE BY THE 2000-2002 TIMEFRAME.

- BASELINE ASSUMPTION OF THE STUDY RELIED ON UPGRADING THE ROVER/NERVA FUEL FORM AND USING TESTED TECHNOLOGIES TO PRODUCE AN ENGINE WITH POTENTIAL FOR Isp BETWEEN 900-925s.

- IMPROVED FUEL RECOVERY AND CHARACTERIZATION CAN BE ACHIEVED WITH A COMBINATION OF ELECTRICAL TESTS AND REACTIVITY MEASUREMENTS. (MAJORITY OF FACILITIES CURRENTLY EXIST.)

- COMPLETE FUEL VALIDATION COULD BE ACCOMPLISHED IN A GENERIC FUEL ELEMENT TEST REACTOR TO SIMULATE ENGINE OPERTIONAL CONDITIONS. (CURRENTLY INVESTIGATING APPLICABILITY OF LOFT AT INEL.)

- FULL-POWER, GROUND TESTING APPROACHES ARE BEING EXPLORED - LANL AND INEL ARE CURRENTLY INVESTIGATING THE REFURBISHMENT AND UPGRADE OF THE PREVIOUS ROVER/NERVA FACILITIES-EMAD AND ETS-1.

SEI FACILITY ACTIVITIES - INEL

- DOE Construction Short Form Data Sheets submitted for SEI Facilities with FY-94 Budget Request

  - Formal submittal for Test Reactor Hydrogen Loop (HFIR and ATR) to DOE-NE

  - Informal submittal of Fuel Element Test Facility, Reactor Test Stand, and Engine Test Stand at undetermined site made to DOE-NE

- LOFT Containment Building re-activation study completed (using INEL internal funds)

- Prepared draft environmental compliance plan for DOE's SEI ground test facility

- Supporting NASA-LERC non-nuclear test facility evaluations

- Supporting NASA-LeRC "Fast-Track" proposal

- Top Level Scoping Evaluation of use of SNTP PIPET for SEI NTP fuels testing

- Limited Evaluation of use of ETS-1 and E-MAD at NTS for SEI engine testing (internal LANL and INEL funds)

- Supporting review of EIS for SNTP

- Developing overall nuclear test selection strategies and plans

Engine Maintenance and Disassembly Facility
Jackass Flats, Nevada
Preliminary Status Report 4/15/92

- **General Description:**
  Designed for the assembly, disassembly, and maintenance of a NERVA-type engine. A T-plan, multi-story structure, 280ft by 350ft divided into 7 separated sections based on specific functions and material traffic flow
  - Cold Assembly Area
  - Hot Maintenance and Disassembly Area
  - Post Mortem Cells
  - High and Low Level Cells
  - Operating Galleries
  - Shop and Service Areas
  - Office Area

- **Preliminary Inspection Results:**
  - Building generally in excellent shape
  - All major equipment items from hot-cell windows and manipulators to machine tools present and in good shape.
  - Overhead cranes in good shape and functional.
  - Electrical system grounding and labeling in compliance and functional.
  - No PCB electrical equipment on site.
  - Plumbing system in place and functional.
  - HVAC major components in good shape and functional.
  - Large shielding doors in place and in good shape.
  - RR system in place complete with engines, load cars, and turntables.
  - New cooling tower needed.
  - Sprinkler system needed.
  - Covering of asbestos flooring needed.
  - Roof leaks need patching.
  - Water tank leak needs patching.
  - Seismic assessment needed but no obvious problems.

M V Hynes, J-DO

Engine Test Stand No. 1 Complex
Jackass Flats, Nevada
Preliminary Status Report, 4/15/92

- **General Description:**
  Designed for ground developmental testing of a downward firing NERVA-type
  engine in a flight simulated environment. The ETS-1 complex includes:
  - A 160ft, 100t aluminum structure supporting a 77,000 gal LH2, vacuum jacketed run tank
    with associated below grade pipe chase and process piping, exhaust duct vault, and a
    3ft wide by 40ft high by 100ft long concrete shadow shield.
  - A below grade control point building supporting 2000 channels of diagnostics
  - A cryogenic dewar and high pressure gas vessel tank farm with interconnecting process piping
  - An engine compartment radiation shield.
  - A diffuser/ejector exhaust duct.
  - A 2.5 Mgal water storage tank.
  - Required I&C, electrical and water systems, HVAC and other support systems.

- **Preliminary Inspection Results:**
  - Complex facilities generally in good shape.
  - Aluminum superstructure in good shape.
  - 250,000 gal LH2 tank in good shape.
  - 77,000 gal LH2 run tank in good shape.
  - Process piping in place.
  - Engine compartment radiation shields in good shape.
  - RR track in place.
  - Electrical switch-gear in good shape.
  - Significant scavenging of HP gas tanks --- one remaining.
  - Above ground buildings need significant repairs.
  - Below grade control point building needs significant upgrading.
  - Some flame-proof electrical boxes missing at test stand.
  - Some stairway sections missing.
  - Shadow shield bracing for seismic shock needs upgrading.
  - Seismic assessment needed.
  - Move LH2 dewar from Test Cell C for longer run times.

M V Hynes, J-DO

Los Alamos

557-12

115527

P. 15

Space Defense Initiative Technologies and Hardware Can Help Resolve Certain Space Exploration Initiative Weight and Performance Issues

N92-33330

Many Aerojet Programs Have Contributed to Advanced Technologies and Hardware

Program and POP	Objective
Advanced Liquid Axial Stage (89-92)	Space Based Interceptor - Advanced Liquid Propulsion and Structures Technologies
Missile Integrated Stage (90-94)	Low Cost Booster/Interceptor
Liquid Propellant Sustainer (90-94)	Gelled Technology for Interceptor
High Endoatmospheric Def. Int. (87-93)	Ground Based Interceptor
SCIT-DACS (87-92)	Kill Vehicle Propulsion
THAADS (92-)	Theatre Missile Defense Propulsion
GBI (90-)	Ground Based Interceptor
Brilliant Pebbles (90-95)	Advanced Booster and Kill Vehicle Propulsion Systems and Structures
Endo LEAP (90-)	Endoatmospheric Interceptor Controls & Cooling

446

SDI Programs' Technical Focus

Lightweight   –   • **High Mass Fraction Stages**
                                  • **Heavy Use of Composites**
                                  • **Advanced Propellants**

Low Cost   –   • **Highly Producible Designs**
                                  • **Integrated Propulsion Modules**

High Performance –   • **Ultrafast Engine Responses**
                                  • **Front-End Cooling for In Atmospheric Flight**
                                  • **Advanced Propellants**

SDI Technology Provides Order of Magnitude Savings on Weight

Current State of the Art

37.9 In
96.3 cm

33.3 In
84.6 cm

Wt = 290 lbm (132 kg)

ALAS

14.07 In
35.7 cm

17.3 In
93.9 cm
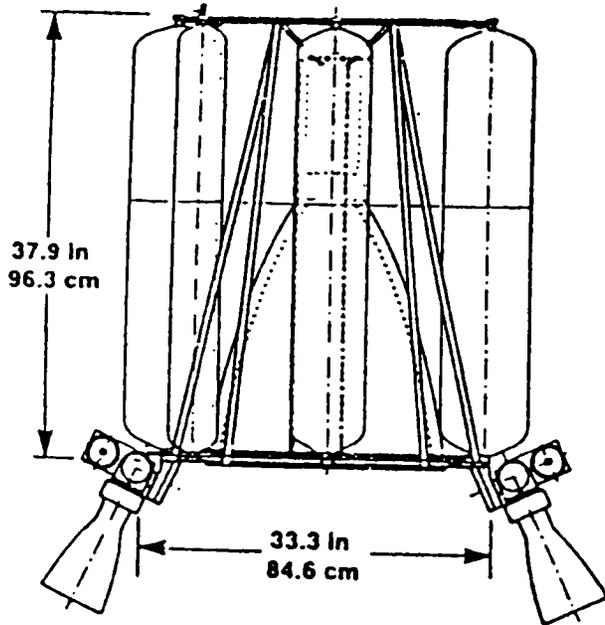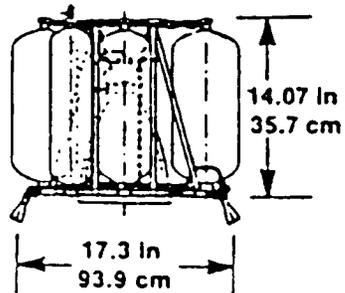
Wt = 38.3 lbm (17.4 kg)

		Current S.O.A.	ALAS	Weight Impact
SIGNIFICANT STAGE DESIGN DRIVERS	Material	All Metal	Carbon Composites	High strength to weight composites are more weight efficient than best metals
	Propellants	N_2O_4/ N_2H_4	ClF_5/N_2H_4	High density oxidizers result in denser, smaller packages
	Isp, sec $\left(\dfrac{N\text{-sec}}{kg}\right)$	310 – 320 (3040- 3140)	340 – 360 (3330-3530)	Higher ISP results in less required propellant for same mission
	F/Wt	50	500 – 1000	Decreases engine weight an order of magnitude
	Response Time, sec	0.010 – 0.030	0.001	Improves control of stage — saves using another set of smaller control engines
	$\dfrac{Press\ Vol}{weight}$ $\dfrac{in}{(cm)}$	6×10^5 (15.2×10^5)	$1\text{-}2 \times 10^6$ $(2.5 - 5 \times 10^6)$	Halves the tank weight

Benefits are Realized in Several Areas

- New Engines

- Structures

- Tanks

- Advanced Propellant

Emerging Composites Technologies Result in Numerous Propulsion Benefits

Subsystem	Conventional Technology	ALAS Technology	Benefit
ALAS Axial Engine	• Refractory Nozzle • Low Density Graphite Chambers • Metal Structural Shell	• Braided Carbon Axial Nozzle • Carbon Structural Shell	• Nozzle Weight Reduced 90%
Propellant Tanks	• All Metal Designs Usually Titanium • Glass – Overwrapped Thick-Wall Metal Liners (Pressure Load Is Shared Between Liner and Overwrap	• Carbon Fiber Overwrapped with Very Thin Wall Liners (Pressure Load Is Not Shared Between Liner and Overwrap)	• ~60% Weight Savings from 1 lbm to .45 lbm Order of Magnitude Savings in Cost $10,000 vs ≤$1000
ACS Engine	• Refractory Nozzle	• Free Standing Graphite Nozzle	• Nozzle/Chamber Weight Reduced from 2 lbm to < 2 lbm
Composite Structure	• All Aluminum Bolted/Welded Configuration	• Injection Molded Carbon Rings • Braided Rings • Stamped Struts • Plastic Welding	• Weight Savings – from 2 lbm to 5 lbm

Advanced Liquid Axial Stage

15.0 in.
(38.1 cm)

Integrated Injector/ Valve

Inconel Heat Exchanger

10,000 psia He Bottle 2 Places

Carbon Overwrapped Aluminum Lined Fuel Tank 2 Places

8.94 in. (22.71 cm)

Ox Tank 2 Places

Carbon-Epoxy Structure

20 lbf ACS Engine Fast Response 4 Places

High Performance Carbon Composite, Restartable Axial
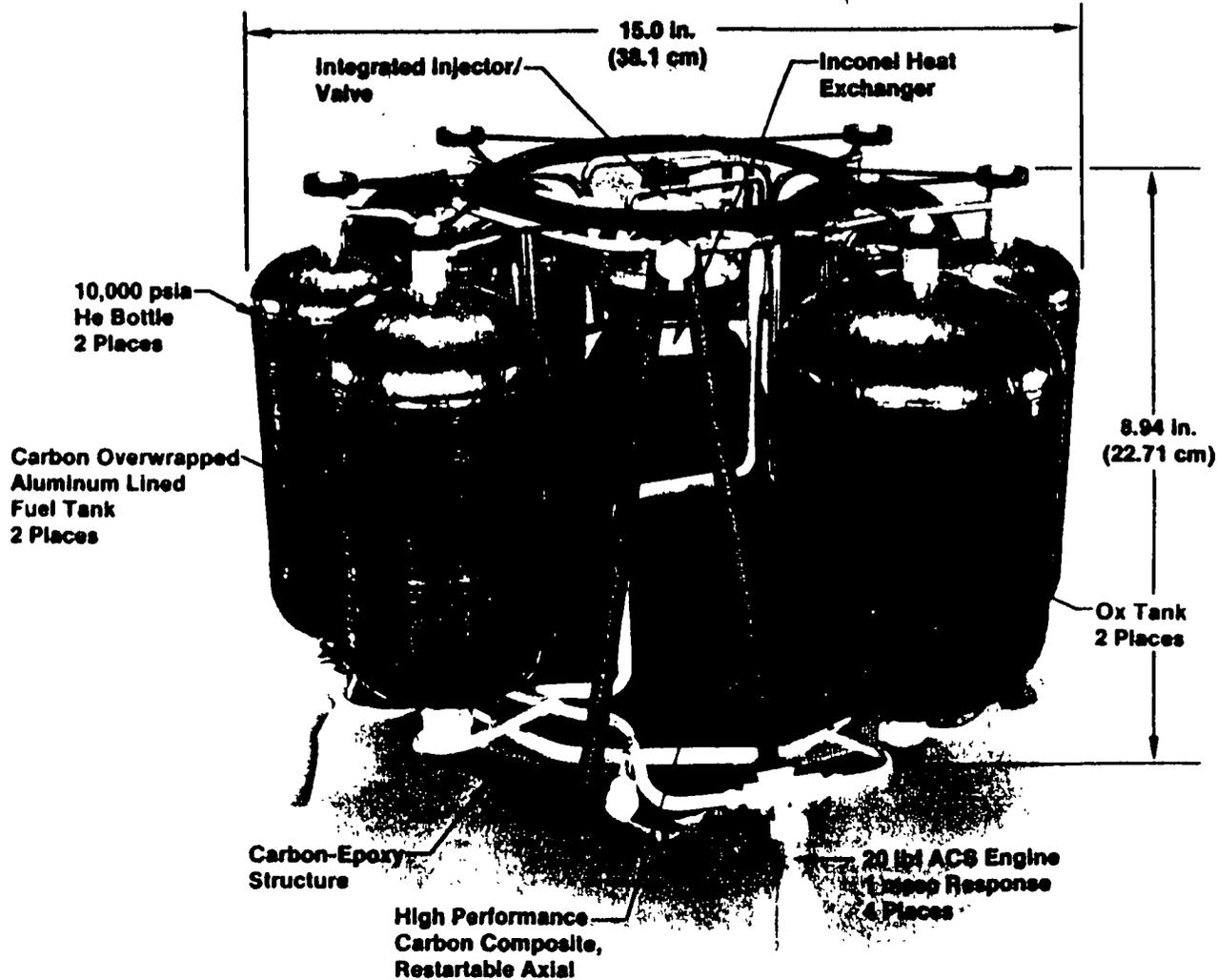
Propellant and Pressurant Tank Accomplishments

Features

- 10^6 psi (7000 MPA) Carbon Fiber

- Yielding .006 in (.015 cm) Al Liner

- No Liner Welds

- Passive Propellant Management Device

Status

- Fiber/Resin System Demonstrated

- .006 in (.015 cm) Liners Made

- Long Term CIF Material Storage Demonstrated

- He Containment Demonstrated With 0.010 in (.025 cm) Liner/@ 10,000 psi

- Prototype PMD Made

- First Burst Tests at 14,100 and 16,860 psia

New Family of Lightweight Engines Has Been Developed

Program	Engine Type	Pc	Tests
ALAS	Axial	775	150 Tests 1989-91
ALAS	ACS	500	110 Tests 1989-91
SCIT	Divert	500	20 Tests 1989-92
LDI	Axial/Divert	300-600	23 Tests 1992 (On-going)
GBI	ACS	500	To Be Tested July 1992
BP	Divert	500	To Be Tested Early 1993
BP	ACS	300	To Be Tested Mid 1993

ALAS Has Demonstrated High Performing Helium Tanks

Welded 2219/1100 Liner

Spun 2219 Liner
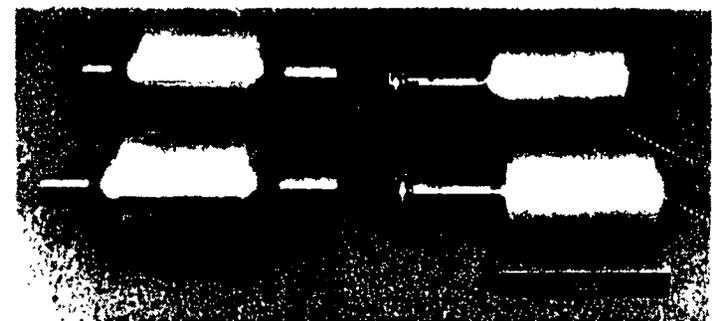
Spun 6061 Liner

- **32 Helium Tanks Fabricated**

- **0.010 in. Liner Wall Thickness Demonstrated**

- **PV/W = 1.2×10^6 Achieved**

- **Helium Permeability 1.0×10^{-9} sccs at 10,000 psi after 20 Cycles Demonstrated**
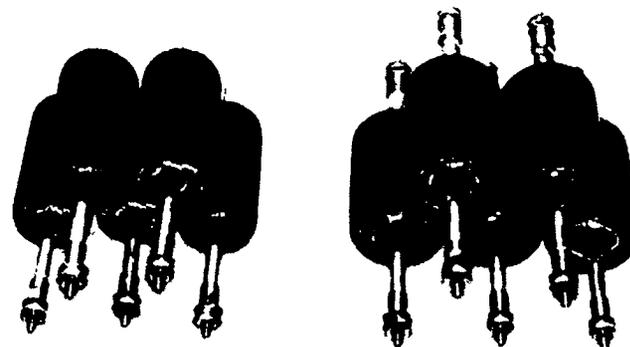
Specification

	Phase I	Phase II
Volume, in^3	40	335
Diameter, in	3.2	6.3
Operating Pressure, psi	10,000	10,000

GENCORP

AEROJET

454

Propulsion Division

14 15 32

Propellant Hoop And Helical Fibers Have Been Selected
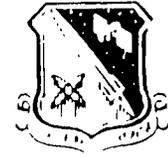
Fiber	Tank Application	Modulus (MSI)	Tank Weight Comparison, %		Delivered Fiber Stress (KSI)	Avg
			Fu	Ox		
T-400 (3K Tow)	Helical	36.4	+8	+4	367 368 370	368
T-650[1] (3K Tow)	Helical	35.0	—	—	591 605 591	596
T-650 (6K Tow)	Helical	42.0	+10	+5	596 609 603	603
Apollo 53-750 (12K Tow)	Helical	53.0	+3	-1	615 666 660	647
T-1000H	Hoop	42.0	+6	+5	919 901 791*	910
T-1000GB[3]	Hoop	42.0	—	—	909 901 961	924 [2]

*Not Included in Average

ALAS0228

[ ] Selected

Selection Criteria

(1) Minimum Weight Design

(2) Higher Strength

(3) Cheaper and Available

ALAS Developed An Advanced Carbon Composite Structure

5 Structures Fabricated
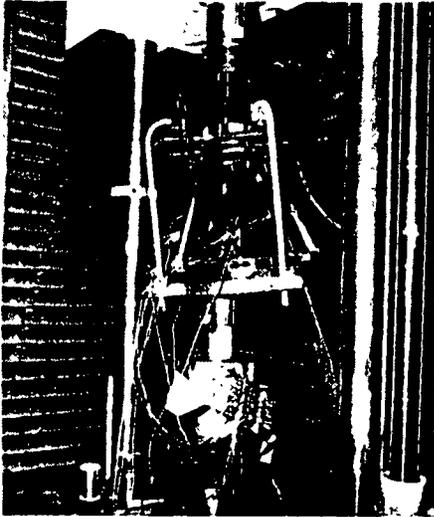
KKV Deflection Test
0.018 in. Deflection at
Flight Load
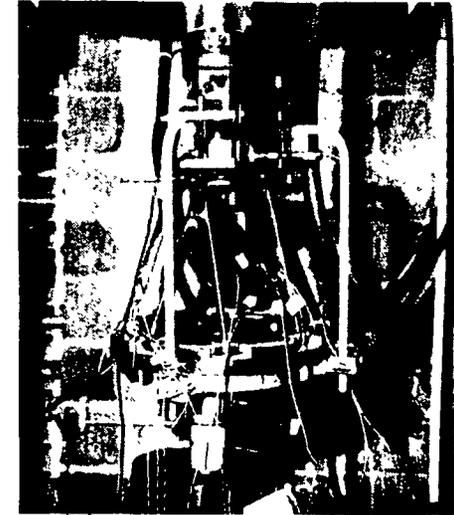
Compression Test
- **Ultimate Failure at 5000 lbf**

SLOSH Tensile Test
- **Strut Demonstrated**
  at 2X Load

Component Tests
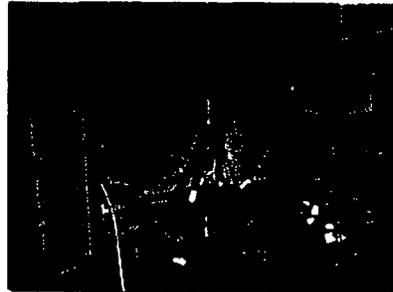
Main Strut Component Test Set-Up
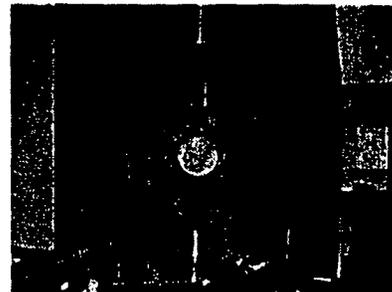
Forward Ring Component Test Set-Up

ALAS Aft Ring Component Test Set-Up

ALAS Structure
Estimated Weight Summary

- **Forward Ring, lbs**   .147

- **Aft Ring, lbs**   .230

   ACS Supports, lbs   .0178

   Tank Support Inserts, lbs   .0086

- **Struts, Structure, lbs**   .328

- **Struts, Engine, lbs**   .041

- **Tank Retaining Pins, lbs**   .011

                            Total, lbs .757

Note: Change in Tank Mounting Method Provides .0195 lbs Total Tank Weight Saving

Integrated Concept Employs Optimum Material for Each Component

Component	Material	Rationale
Helium Tank Mount	High Strength Graphite Fiber/High Elongation Resin [±45°] Layup	Best Balance of Stiffness/Strength
Longeron	High Modulus Graphite Fiber/BMI Resin [±45°/0°/±45°] Layup	Stiffness Driven Producible BMI for Thermal Capability
Aft Ring*	High Strength Graphite Fiber/High Elongation Resin	Best Strength/Weight Ratio for Launch Looks
Forward Flange*	Beryllium	Stiff Isotropic Machined Part Ribs/Bosses

*Detailed Structural Analysis and Dynamics Must Be Done

CLF_5 Offers Improved Performance Without Undue Safety/Toxicity Issues

- **Performance**
    - **High specific impulse - 340-360 sec delivered**
    - **High specific gravity - 1.8 vs. N_2H_4 = 1.04**
- **Safety**
    - **No untoward incidents in 5 years of recent testing**
        - **Over 300 rocket engine tests**
        - **Over 25 different engines**
        - **Stage test (loading and firing)**
    - **Handles like N2O4 - and tested with same precautions** (Amines are more trouble)
    - **Strong reaction with hydrocarbons - must be clean**
        - **Lox cleanliness level is appropriate**
- **Toxicity**
    - **Only about two-four times as toxic as N_2H_4**
    - **About 4-8 times safer than Titan III launch**
        - **Titan III fuel load = 105,000 lb of N_2H_4/UDMH**
        - **CLF_5 on Atlas ~ 6,500 lb**
        - **Equivalent N_2H_4 = 13,000-26,000 lb**

LOW PRESSURE - HIGH PERFORMANCE

BOOSTER ENGINE TECHNOLOGY

FOR

ULTRA-LOW COST LAUNCH VEHICLES

**PRESENTED TO THE THIRD SPACE EXPLORATION
INITIATIVE TECHNICAL INTERCHANGE MEETING
UNIVERSITY OF HOUSTON, CLEAR LAKE PARK**

MAY 5 & 6, 1992
R. L. SACKHEIM
DEPUTY DIRECTOR
PROPULSION TECHNOLOGY
AND FLUID MECHANICS
CENTER, APPLIED TECH. DIVISION
TRW SPACE & TECHNOLOGY GRP.

N92-33387

TRW

Approach

Low-cost booster engine
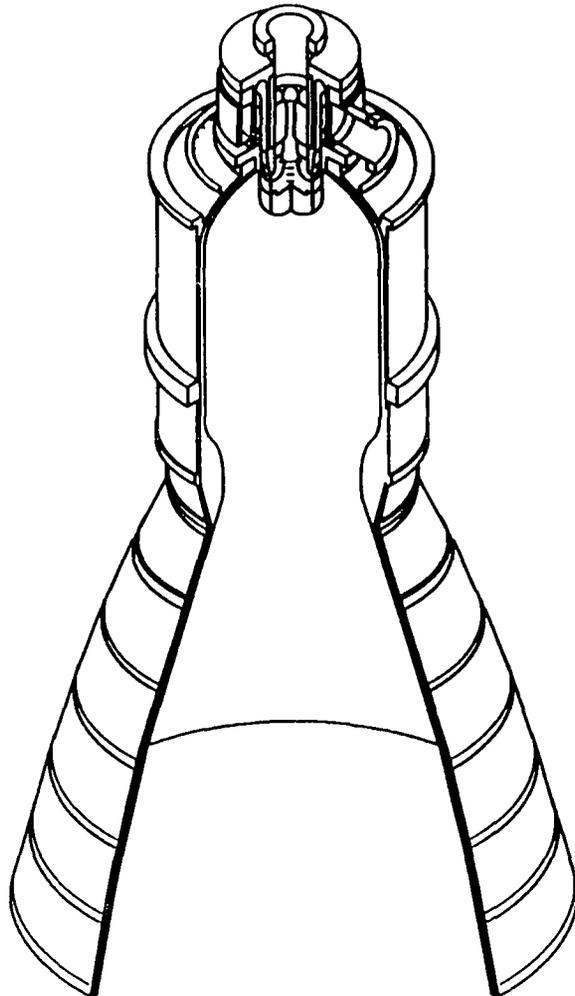
- Strategic advantages
  - Reduced booster engine cost with enhanced reliability and safety

    Enabling technology for low-cost assured access to space

    Reduced launch costs are needed to achieve national space program goals

    Would enhance competitive position of U.S. commercial space industry

- TRW development activities
  - TRW is a leader in pioneering low-cost booster engine technology
  - Current work builds on TRW technology work started in late 1960s
  - A cooperative program is in place to demonstrate low-cost booster technology at small scale (16.4 Klbs thrust)
    - LOX/LH$_2$ propellants

    - TRW (IR&D)

    - NASA Lewis Research Center (National Space Act)

Background

- TRW pioneered a minimum-cost booster engine concept in mid 1960s

    - Low pressure, pressure fed
        - $P_C \sim 300$ psia

    - Ablative cooled

    - Commercial manufacturing processes

    - Pintle injector
        - Produces inherently stable combustion

    - Relaxed margins

- Several minimum cost engines were fabricated and tested

    - 250 K
        - NTO/UDMH

    - 50 K
        - IRFNA/UDMH
        - LOX/Propane
        - LOX/RP-1

    - 35 K
        - IRFNA/UDMH

    - All were stable

- Efforts terminated in early 1970s

TRW PINTLE INJECTOR BACKGROUND

- TRW PINTLE INJECTOR WAS ORIGINALLY DEVELOPED UNDER NASA APOLLO PROGRAM AS THE LUNAR MODULE DESCENT ENGINE (LEMDE)

    - P_C = 100 PSIA

    - NTO/A-50

    - DEEP THROTTLING (10:1)

    - 10,500 LBS FULL THRUST

    - ABLATIVE COOLED

    - PRESSURE FED

- A FIXED THRUST VARIANT OF LEMDE (TR 201) FLEW 75 SUCCESSFUL FLIGHTS AS THE LOW COST, PRESSURE FED SECOND STAGE ENGINE OF DELTA

- TRW LOW COST BOOSTER ENGINE WORK IS BASED ON THE LEMDE CONCEPT
    - SIMPLIFIED PINTLE INJECTOR DESIGN
    - COMMERCIAL MANUFACTURING PROCESSES
    - LOW COST ABLATIVE

- PINTLE INJECTOR ADVANTAGES INCLUDE:

  - INHERENT COMBUSTION STABILITY ON A WIDE RANGE OF PROPELLANTS WITH GOOD TO EXCELLENT COMBUSTION PERFORMANCE HAS BEEN DEMONSTRATED

    - NTO/UDMH
    - NTO/A-50
    - LOX/LH$_2$
    - HDA/USO

    - LOX/PROPANE
    - LOX/RP-1
    - IRFNA/MMH (GELS)

    - NTO/MMH
    - NTO/N$_2$H$_4$
    - LOX/N$_2$H$_4$
    - IRFNA/UDMH

  - MULTI-PROPELLANT CAPABILITY WITH THE SAME ENGINE DESIGN
    - IRFNA/UDMH, LOX/RP-1 AND LOX/LH$_2$ INJECTORS CAN BE A COMMON DESIGN

    - INJECTOR CAN BE DESIGNED FOR EASY RECONFIGURATION TO ACCOMMODATE PROPELLANT CHANGES

  - DIAL-A-THRUST CAPABILITY
    - INJECTOR CAN BE DESIGNED FOR EASY RECONFIGURATION TO ACCOMMODATE DIFFERENT THRUST LEVELS

  - DEEP THROTTLING CAPABILITY
    - MOVING SLEEVE MAINTAINS PROPELLANT VELOCITIES FOR EFFICIENT THROTTLING

    - 19:1 THROTTLING HAS BEEN DEMONSTRATED WITH LOW COST TECHNOLOGY

OXIDIZER FAN

HOLLOW FUEL FAN
FORMED BY SHEET
IMPINGEMENT

FUEL SHEET

INJECTOR PINTLE
SHOWING FUEL
PENETRATION

SECONDARY SLOT

PRIMARY
SLOT

TIP

PINTLE

SLEEVE

FUEL

OXIDIZER

VIEW A-A
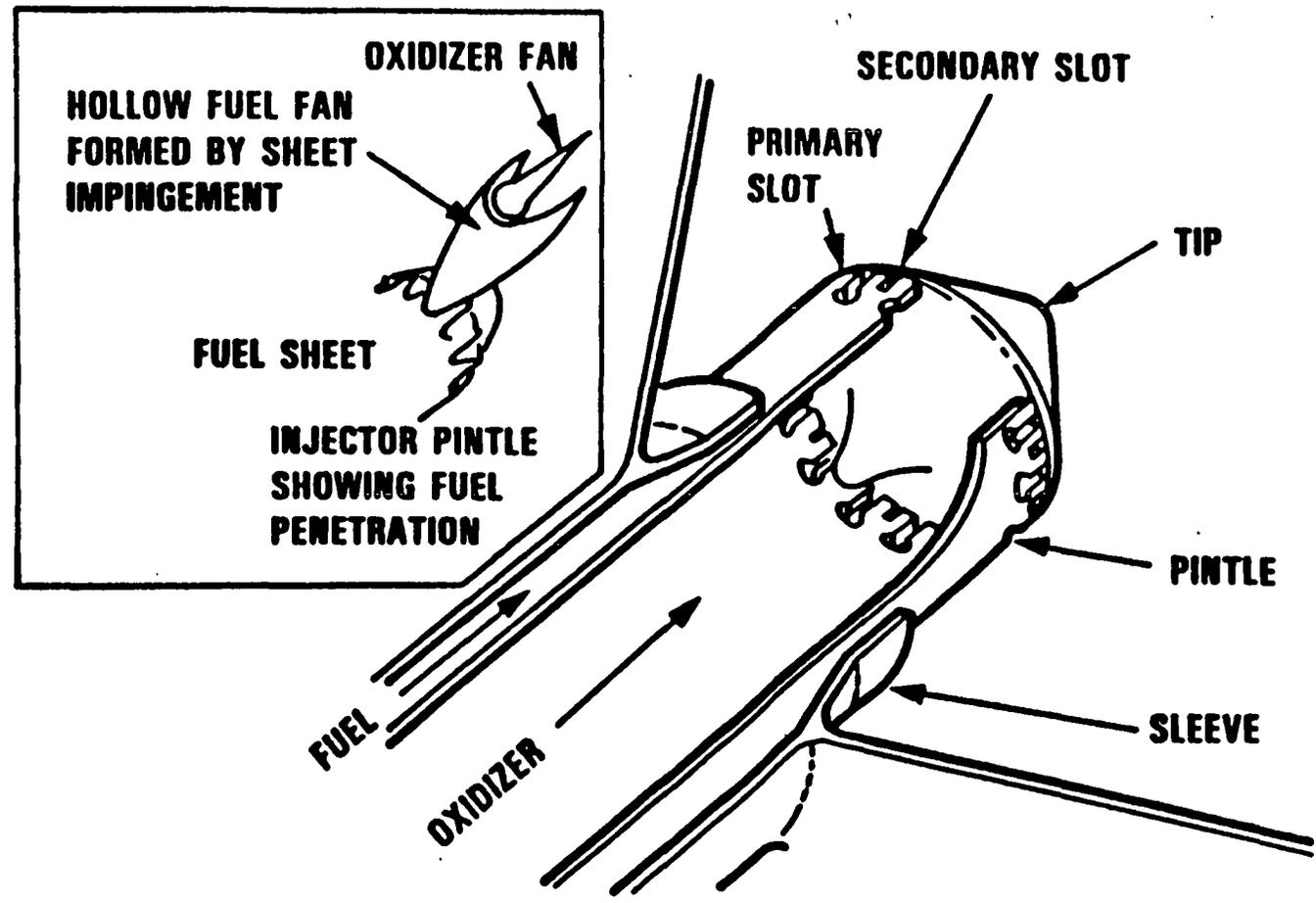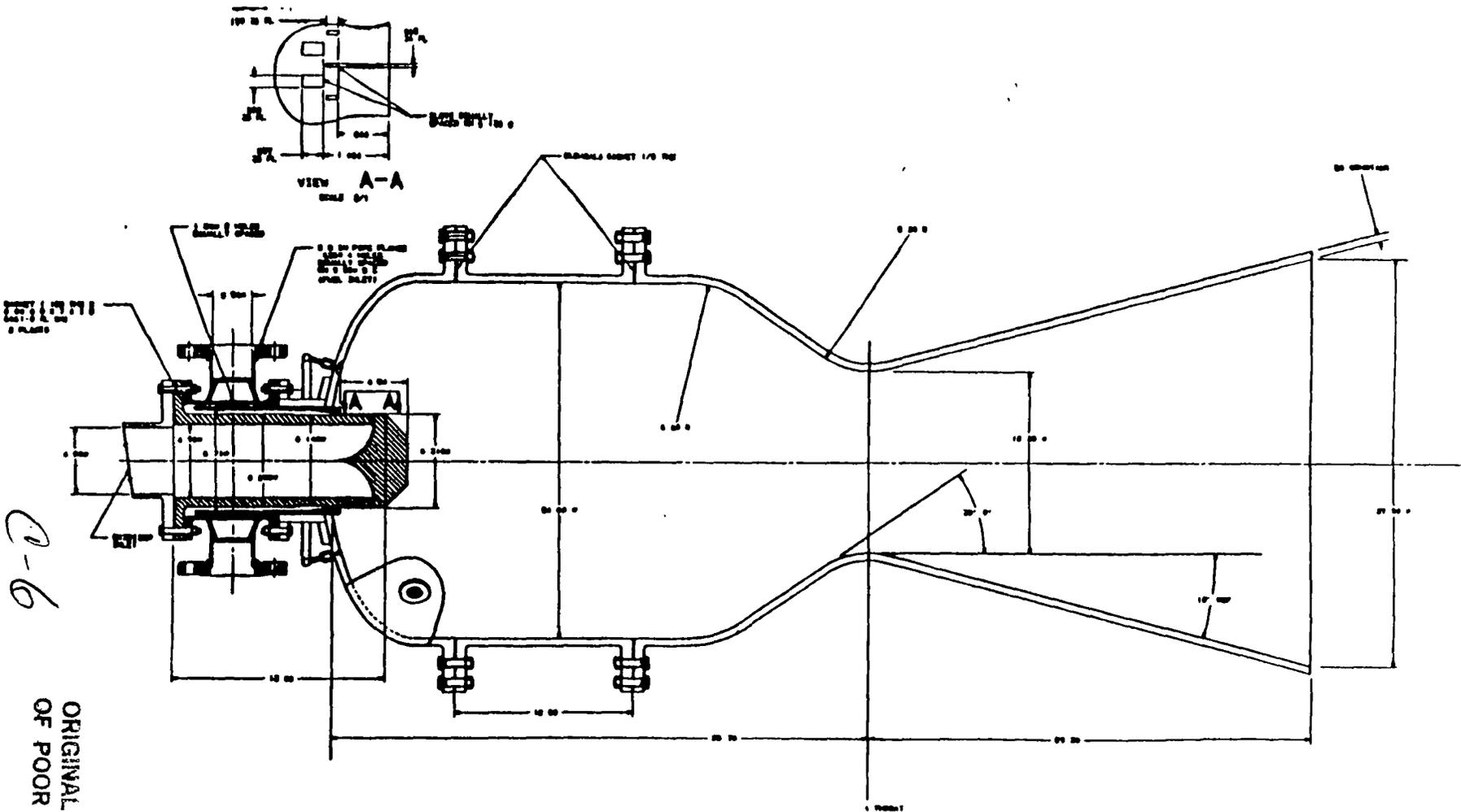
COMPARISON OF BOOSTER ENGINE PERFORMANCE

	ENGINE DESIGNATION							
	F-1	H-1	RS-27	TRW CONCEPTS*				SSME
PROPELLANTS	LOX/RP-1	LOX/RP-1	LOX/RP-1	LOX/RP-1	LOX/RP-1	LOX/LH$_2$	LOX/LH$_2$	LOX/LH$_2$
P_C (PSIA)	982	700	700	300	660	300	660	3270
ε	16.0	8.0	8.0	7.0	10	7.0	10.0	77.5
O/F	2.27	2.23	2.25	2.67	2.24	6.6	6.6	6.0
$I_{SP_{SL}}$ (SEC)	265.4	263.4	263.3	233.9	263.4	304.2	339.5	370.8
$I_{SP_{VAC}}$ (SEC)	304.1	295.8	294.7	291.5	301.6	379.2	388.6	455.3

* $\eta_{I_{SP}}$ = 93.1% FOR LOX/RP-1

$\eta_{I_{SP}}$ = 96.0% FOR LOX/LH$_2$

LAYOUT OF 16.4 K LOX/LH₂ HEAT SINK ENGINE

TRW
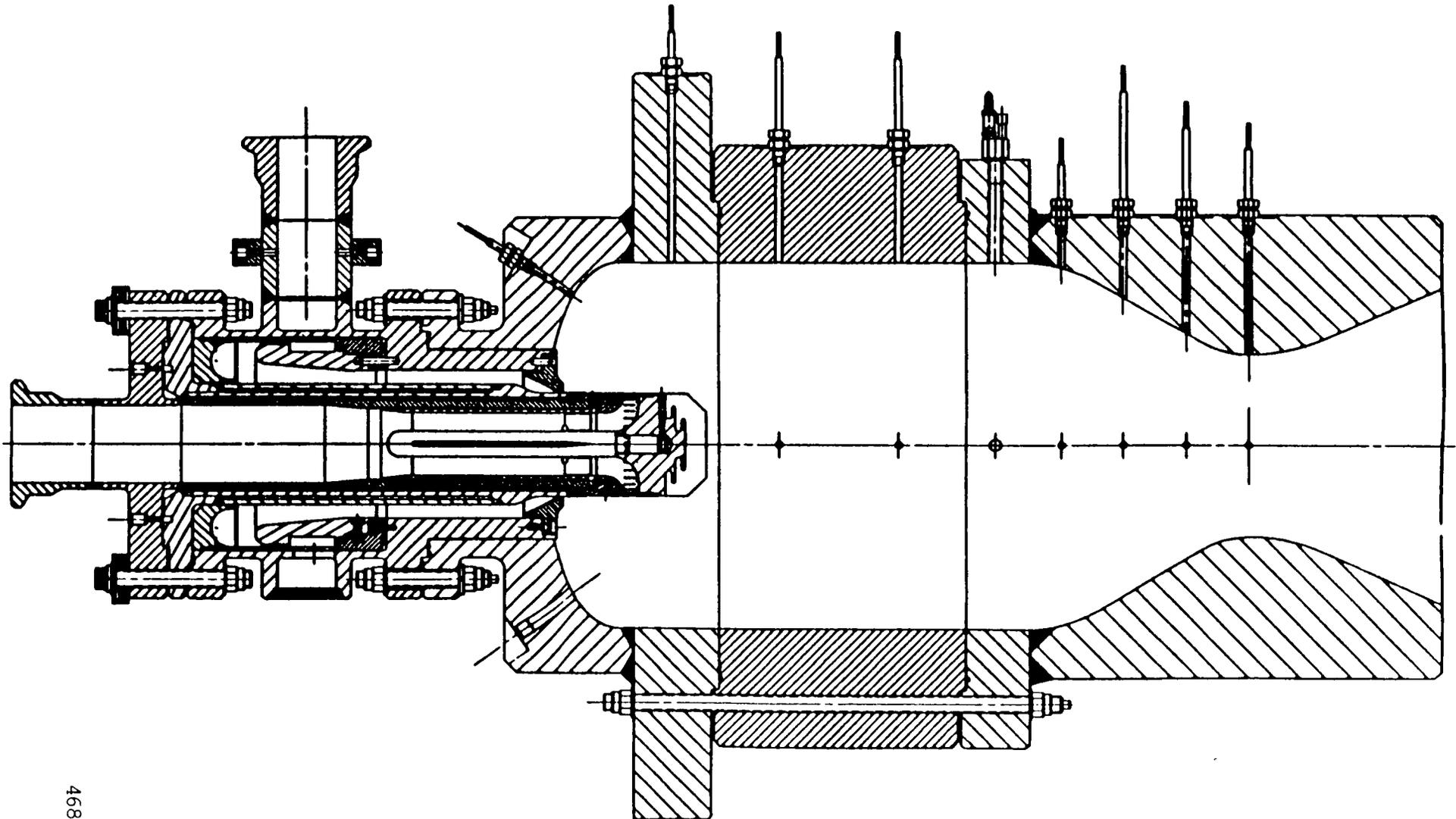
BOMB TEST SUMMARY

- FIVE BOMB TESTS
    - 20 GRAINS, TANGENTIAL
    - 20 GRAINS, RADIAL
    - 40 GRAINS, RADIAL
    - 40 GRAINS, TANGENTIAL/20 GRAINS, RADIAL
        - SIMULTANEOUS FIRING

- LARGEST OVERPRESSURE ~ 200 PSI

- Pc RECOVERY WITHIN 40 m sec

- ABSOLUTELY STABLE

STATUS OF 16.4 K LOX/LH2 TESTS AT LeRC (5 MARCH 1992)

- 67 HEAT SINK CHAMBER TESTS COMPLETED (UP TO 2.5 SEC DURATION)
  - INJECTOR PARAMETRIC OPTIMIZATION
  - OFF-DESIGN OPERATION
- ONE 20-SEC ABLATIVE CHAMBER FIRING PERFORMED
- HAVE DEMONSTRATED
  - OPERATION AT EXTREME OFF-DESIGN POINTS
    - O/F FROM 4.9 TO 11.8 (DESIGN IS 6.6)
    - 100% DOWN TO 60% RATED FLOWS
  - OPERATION ON SATURATED LOX
  - OPERATION WITHOUT LHe INJECTOR PRECHILL
  - ABLATIVE PERFORMANCE
  - HIGH COMBUSTION PERFORMANCE
    - > 98% C* FOR HIGHEST PERFORMING PROPELLANT RING COMBINATION
      - BLOCKAGE RATIO = 48%, ΔP = 150 PSIA LOX RING
      - ΔP = 50 PSI LH2 RING
    - THIS RING IS EXPECTED TO ACHIEVE 95% C* ON LOX/RP-1
  - ABSOLUTELY NO INDICATION OF SPONTANEOUS COMBUSTION INSTABILITIES
    - P_{rms}/P_c < 1% (LEMDE QUAL. DATA 1.2% < P_{rms}/P_c < 2.0%)
    - CONTRIBUTION TO P_{rms} AT FREQUENCIES > 300 HZ IS NEGLIGIBLE
      - VIRTUALLY NO COMBUSTION TURBULENCE EVIDENT
  - NO HARD STARTS OR SHUT DOWNS
  - FIVE BOMB TESTS WITH NO INDUCED COMBUSTION INSTABILITIES

FEASIBILITY OF LUNAR COMMUNICATIONS USING THE TDRS II

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Washington, DC 20024

ABSTRACT

The Space Network's Tracking and Data Relay Satellite (TDRS) II geostationary satellite constellation may be capable of providing lunar communiocations for the Space Exploraion Initiative. This paper summarizes geometrical coverage constraints, link budgets, TDRS II delivery schedules and life cycle costs for various Space Network architecture options.

1 INTRODUCTION AND BACKGROUND

The Space Network is a constellation of equatorial geosynchronous Tracking and Data Relay Satellites (TDRSs) which relay the communications from low earth orbiting user spacecraft to the White Sands Complex (WSC) - a ground station and processing facility at the NASA complex in White Sands, NM. The nominal Space Network constellation [1,2] in the Space Station era consists of four spacecraft located at 41° W, 46° W, 171° W and 174° W longitudes (see Figure 1). A TDRS spacecraft has two independently steerable 4.9 meter dishes, each of which supports Single Access S-band and Ku-band services (see Figure 2). Single Access services are circularly polarized with a user-selected polarity.

In the later half of this decade, the Space Network will begin a transition from the TDRS to the TDRS II. The TDRS II will extend communications service support to users with orbital altitudes up to Geostationary Earth Orbit (GEO) while remaining backward compatible with existing TDRS services. In addition, TDRS II will introduce a new Single Access service at Ka-band.

Space Exploration Initiative lunar missions require communications support for bases and rovers on the lunar surface, as well as spacecraft in-transit between the earth and the moon. A communications architecture based on existing Space Network concepts can provide much of the basic communications needs of the lunar initiative. Modification of the Space Network satellite design or constellation architecture, possibly including the addition of a terrestrial terminal, can allow the Space Network to provide lunar communications.

This paper summarizes previous estimates of geometrical coverage constraints, link budgets, TDRS II delivery schedules and life cycle costs for various Space Network architecture options [3,4,5]. Geometrical coverage constraints are determined by comparing the coverage provided by constellation architecture options with possible lunar user locations. Link budgets are based on existing Space Network service requirements, projected lunar user requirements, and estimates of lunar user characteristics. Projected TDRS II availability is compared to estimated need dates. Life cycle costs for comparing ground costs in the architecture options are based on WSC upgrade costs. Satellite costs are based on NASA estimates.
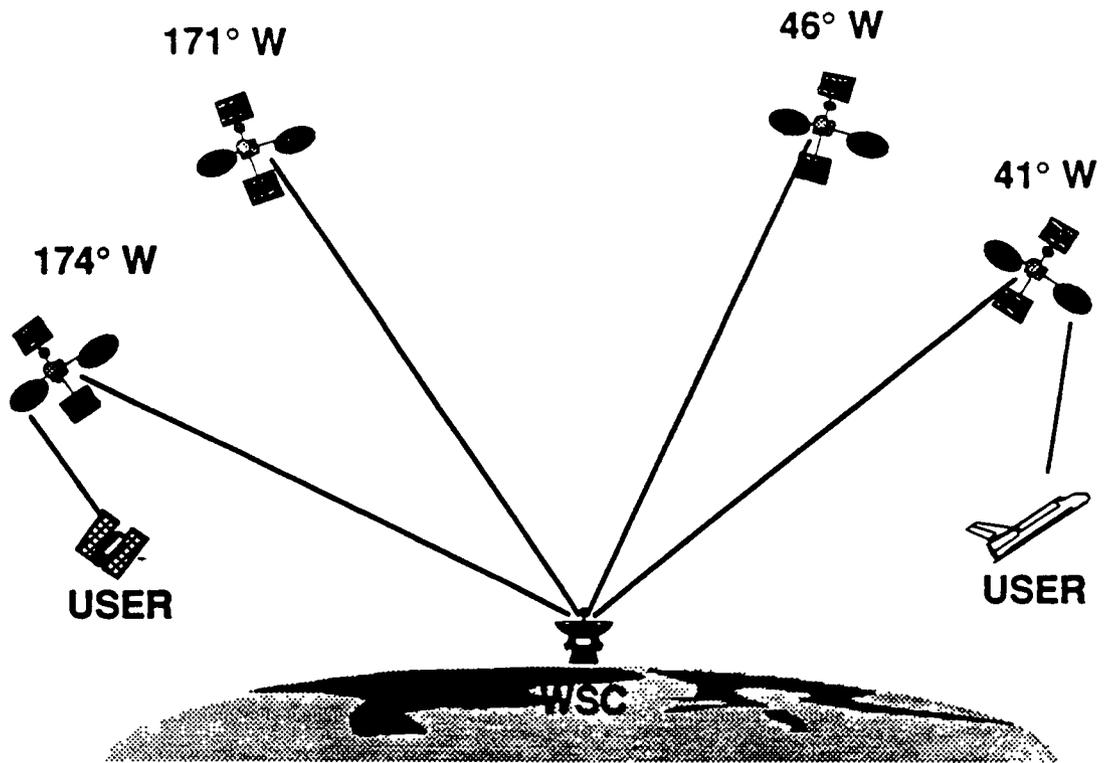
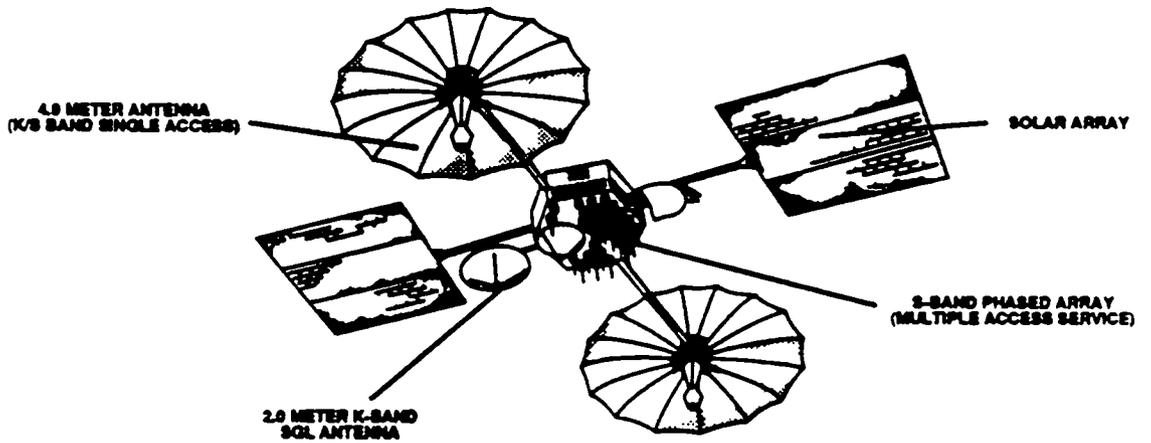1

Figure 1. Space Network Architecture

Figure 2. The Tracking and Data Relay Satellite

2 SUMMARY OF RESULTS

2.1 General Field of View Considerations

The Field Of View (FOV) of a constellation and/or ground stations is constrained by the limits on the maximum antenna scan angles and the placement of the satellites or earth terminals. Satellite antenna scan angle limits in elevation or azimuth result in areas behind the satellite which are outside of the FOV (See Figure 3A). Earth terminal FOV is limited at low elevation angles due to terrain masking and absorption and scintillation due to increased atmospheric transit ranges (see Figure 3B). Earth intrusion effects a satellites FOV just as terrain masking effects a ground terminals FOV

The impact of the FOV constraints on lunar support depends on the orbital ephemeris - orbital parameters - of the lunar user's trajectory. Elevation angle requirements are determined by the declination extrema of the TDRS II and the user in-transit or on the lunar surface. The maximum declination of a user in an orbit is the inclination angle of the orbit. TDRS II orbital inclination is assumed to be no greater than 7°. Lunar users may be on the surface of the moon or in a transfer orbit between a low earth parking orbit and the moon. The extrema in declination of a user on the lunar surface is determined by the ephemeridies of the moons orbit, while the in-transit user's declination extrema is determined by the cis-lunar transfer orbit. The moon's orbit is nearly circular and has an inclination of up to 28.5° north and south. The apparent angular extent of the moon as viewed from the earth is about 0.5°.

The selection of a cis-lunar transfer orbit is governed by mission constraints. Circumlunar trajectories with retrograde lunar orbits and direct return trajectories were often used during Project Apollo. Injection into these orbits occurred from low earth parking orbits after near due east launches from Kennedy Space Center. Inclinations of these orbits are typically below 34.3°. Communications support for unconstrained lunar and in-transit longitudes requires full circle azimuth coverage. The maximum in-transit user declination drives the TDRS II elevation angle requirement.

Earth intrusion outages occurred in groups of two to seven clustered around the time the moon crosses the equatorial plane. The onsets of the outages within a group are separated by between 24 and 26 hours. Outage durations are up to 82 minutes. The frequency of onsets increases with TDRS II inclination, decreases with lunar inclination, and decreases as the relative geocentric azimuth of the ascending nodes of TDRS II and the moon increases.

2.1.1 Near Baseline Architecture

The baseline architecture is the nominal Space Network constellation. Three Zones Of Exclusion (ZOEs, i.e., areas outside of the line-of-sight due to FOV restrictions or intrusion of the earth into the FOV) occur in the baseline architecture at altitudes beyond GEO; one ZOE is located "behind" the constellation (i.e., in the region above the western hemisphere), and the other two are due to earth-intrusion into the line-of-sight from the TDRS II to the user (see Figure 4). A fourth ZOE occurs at low altitudes (below 1200 km), but does not uniquely impact lunar communications support.
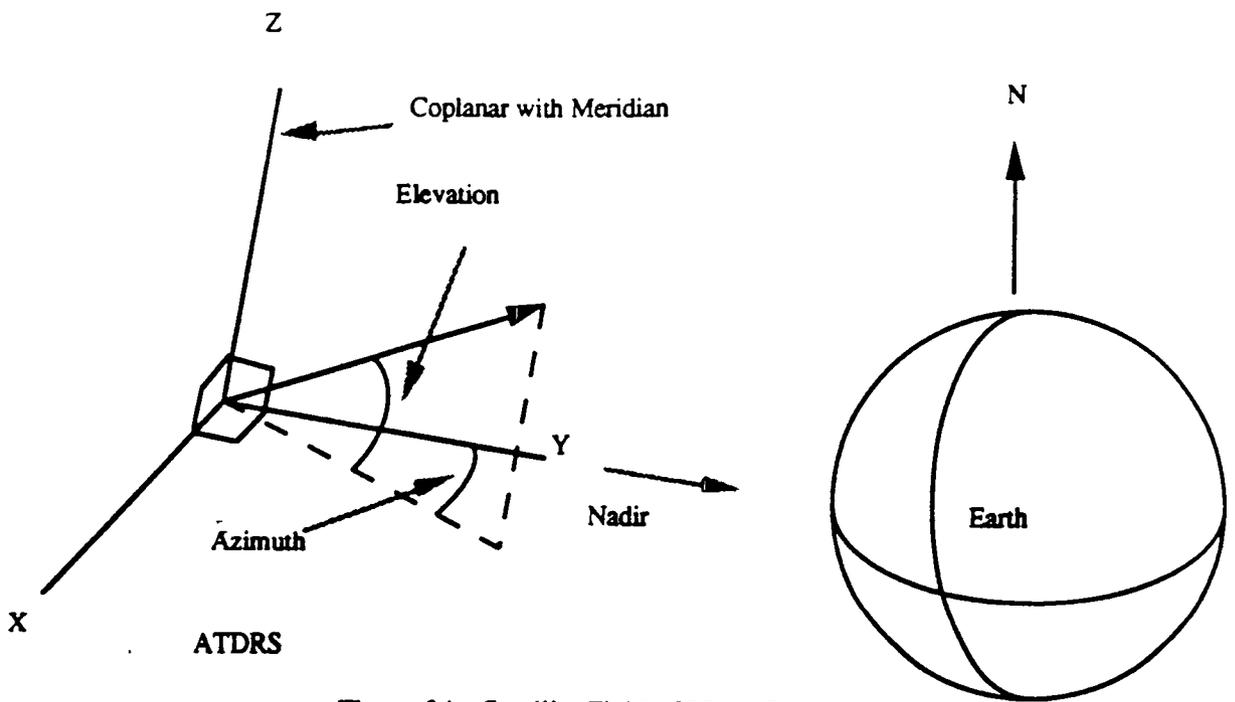
3

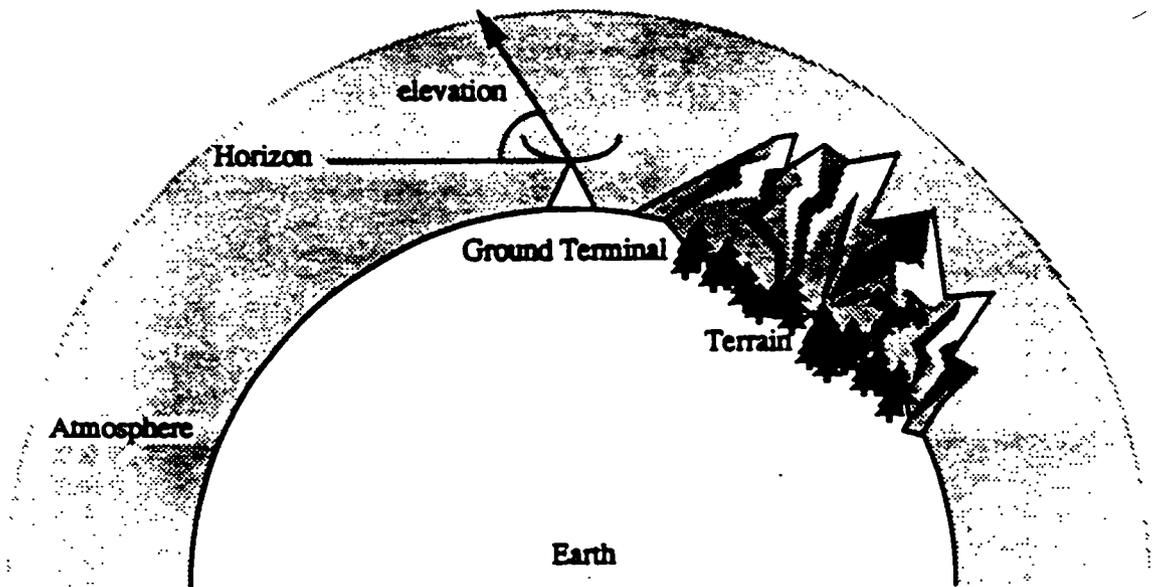Figure 3A. Satellite Field of View Constraints

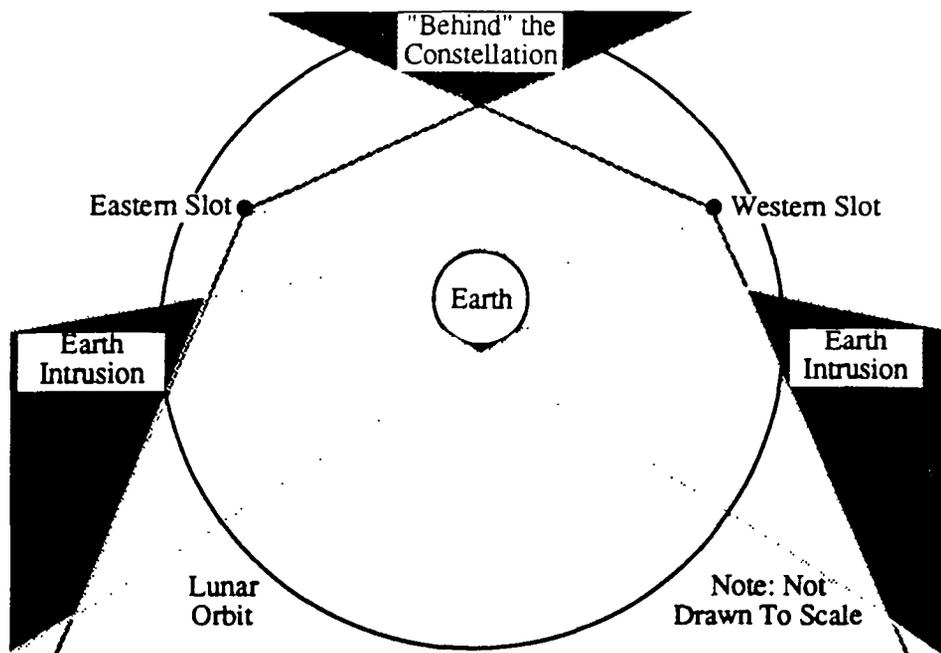Figure 3B. Earth Terminal Field of View Constraints

4

Figure 4. Baseline Space Network Constellation Field Of View Constraints

The connectivity provided by the baseline Space Network architecture may be characterized by the mean communications link availability. The baseline architecture provides 80% mean link availability for users on the lunar surface. For in-transit users, the outage due to earth intrusion (which accounts for two to three percent of the lost communications link availability at lunar ranges) will increase until it approximately doubles at GEO altitudes. This will be counteracted by the decrease in the communications link availability lost "behind" the constellation, which disappears at GEO altitudes. This availability assumes an increase in the TDRS II elevation angle scan limits to 39°. The nominal TDRS II maximum elevation scan angle is sufficient except when the TDRS declination and the user declinations are at their extrema.

2.1.2 Augmented Architecture

In the "augmented" architecture, the ZOEs are closed by augmenting the near-baseline architecture with a ground station on the earth and an operational satellite in a central location (see Figure 5).

The near-baseline TDRS II architecture of four operational satellites can provide a communications link availability of 80% for lunar users. The communications link availability is increased to 97-98% by the addition of a ground station at the WSC. The ground station provides coverage when the user is "behind " the constellation. The WSC ground terminal and the baseline TDRS II constellation can cover the ZOE "behind" the constellation with minimum elevation angles above 16.2° for lunar surface users and 11.4° for cis-lunar transit users. (The use of the same frequencies for both space-to-space and space-to-ground links may result in regulatory conflicts.)

5

The addition of a TDRS II satellite at a central location (i.e., an orbital slot between 102.5° W and 112.5° W longitude) increases the link availability to 100%. The additional satellite covers lunar users when line-of-sight to all baseline Space Network TDRS IIs is blocked by earth-intrusion. The central satellite would also prevent traffic overload caused by the addition of the lunar users. The WSC ground terminal and the central satellite can, by themselves, provide 84% availability for a single user and 51% for two independent users. (The reduced coverage for two users is due to limits on the overlap of the two TDRS II antennas.)
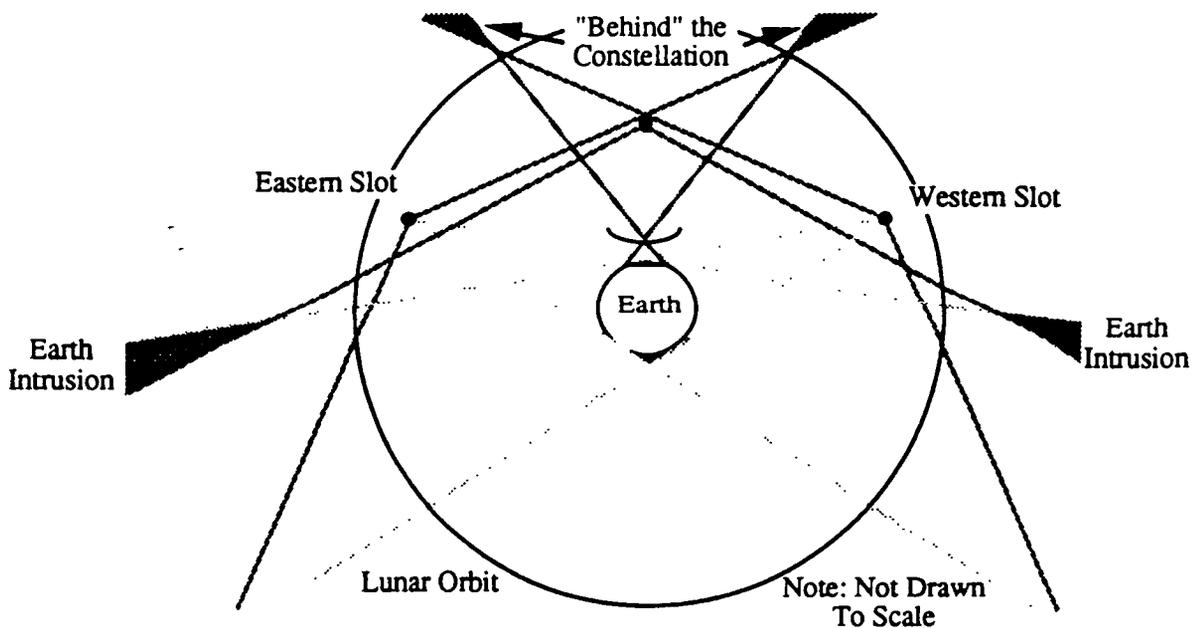
Figure 5. Augmented Space Network Constellation Field Of View Constraints

2.1.3 New Configuration

In the "new configuration" architecture, one or more satellites are placed at remote locations, i.e., below the horizon as viewed from the Space Network ground terminals in White Sands, NM. These remote satellites provide coverage of ZOEs which occur in the baseline architecture (see Figure 6). The placement of one satellite in an orbital slot between 36.8° E and 106.2° E longitude will provide coverage of the ZOE "behind" the baseline constellation -including about six degrees of operational overlap - except for the earth-intrusion into the line-of-sight of the remote satellite. This assumes an increase in the TDRS II elevation angle scan limits to 38°. The nominal TDRS II maximum elevation scan angle is sufficient except when the TDRS declination and the user declinations are at their extrema.

The earth-intrusion into the remote satellite's line-of-sight and one of the two earth-intrusion ZOEs in the baseline architecture may be eliminated by the proper placement of the remote satellite. If the satellites in the baseline architecture used for lunar support are those in the 41° W and 174° W longitude orbital slots and the operational overlap is relaxed from six to one degree., then a remote satellite placed between 28.8° - 29.5° E longitude or between 115.5° - 116.2° E longitude will satisfy these conditions. An increase in the azimuth antenna scan angles of the satellite may be used to increase the operational overlap and/or increase the orbital slots which satisfy these conditions. Two remote satellites can provide complete coverage and operational overlap.

The new configuration architecture presumes the use of direct inter-satellites links for relaying high volume user data from the remote TDRS IIs (those not within the line-of-sight of the WSC) to a relay TDRS II which is within line-of-sight of the WSC. (The inter-satellite link is an option for the future system growth reserve in the TDRS II Phase B program, and has been suggested if closure of the low altitude ZOE is required.). The new configuration architecture also results in the introduction of ground stations outside the continental United States to support the telemetry, tracking and control of the remote satellites.
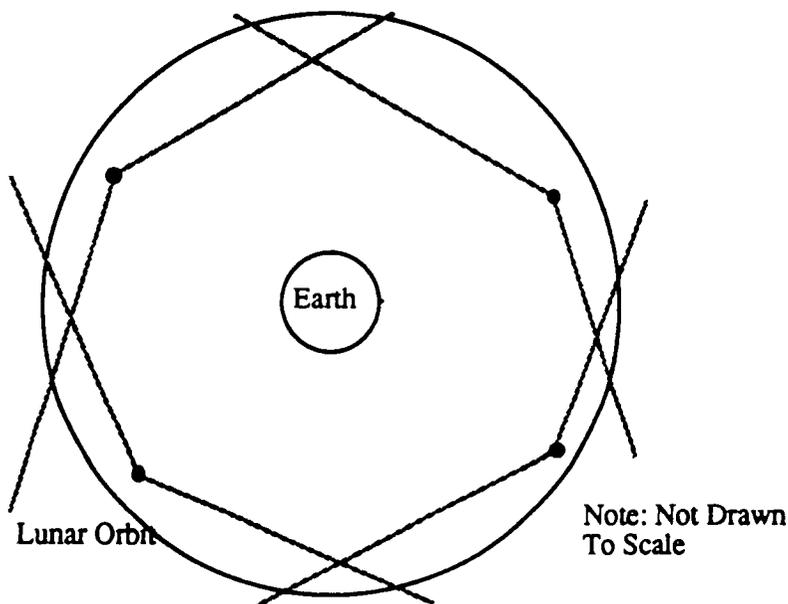

Figure 6. New Space Network Constellation Field Of View Constraints

2.2 Services

Projections of the data rates for lunar surface support have shown maximum forward rates of 25 Mb/s and return data rates of 350 Mb/s. The rates required for the first lunar outpost may be significantly less [5]. Link analyses show that the TDRS II can support S-band, Ku-band, and Ka-

477

band single access services at lunar distances (see Table 1). Rate 1/2 coded QPSK modulation, which is not specified for the present forward link, is suggested to increase the forward data rate supportable by TDRS II for lunar users. One coded Ka-band return service can provide a return data rate of 150 Mb/s. The 350 Mb/s maximum data rate projected for a lunar user is not one of the Phase B TDRS II options. Coded return services are not presently specified for TDRS II Ka-band return signal access services at data rates above 150 Mb/s although there may be sufficient bandwidth in the channel to support rate 1/2 coded 350 Mb/s return signals.

In S-band, a 1.7° Half-Power Beam Width (HPBW) antenna pattern from a 5 meter antenna would produce a beam diameter of over 10,000 km at lunar ranges. This would provide coverage of the entire lunar surface. At Ku-band, a 0.2 ° HPBW produced by a 5 meter antenna would produce a spot beam with a radius in excess of 700 km on the lunar surface. Similarly, at Ka-band, a 0.1 ° HPBW produced by a 5 meter antenna would produce a spot beam with a radius in excess of 350 km on the lunar surface. Efficient use of the nominally single access services by multiple lunar users may require the use of some multiplexing or multiple access technique.

2.3 Schedule

The first TDRS II satellite's availability date is assumed to be in 1997, and the following satellites are assumed to be available for launch every twelve to eighteen months The projected date for the first robotic precursor mission is in the late 1990's, the first test flight of the manned portion of the lunar exploration initiative is in the early 2000's. Support from the robotic precursor could be provided by a WSC ground station and/or the first TDRS II. The manned lunar mission could be supported by three TDRS IIs (see Figure 7).

Space Network replenishment is based on a launch strategy in which on-orbit replacement satellites are available when needed (as spares) with a probability of 80%. If operational use is made of the planned on-orbit spare, then no additional TDRS IIs are required in the augmented architecture. In the event of a failure of an on-orbit TDRS II in one of the eastern or western slots, either the central satellite would need to be moved to replace the failed TDRS II, resulting in a decrease in lunar communications coverage to 97-98%, or there would be decreased services in the eastern or western slots until another TDRS II is launched. If an augmented architecture of one operational satellite in the central location in addition to the baseline architecture of four operational satellites plus one spare is implemented, at least 2 additional satellites would likely be required by the end of the TDRS II era (2013) in order to ensure an 80% availability.

2.4 Cost

The augmented architecture requires the addition to the WSC of three direct lunar downlink antennas (two operational plus one spare) and a TDRS II equipment chain (which normally supports two independent links, each with a primary and hot spare). Commonality of design was assumed in order to eliminate non-recurring development costs, and collocation to minimize operational costs. The resulting projected ten-year life-cycle cost for this architecture is less than $125 million. This does not include the additional costs of the communications links between the WSC and the terrestrial end-users.

User Characteristics

Band	S-low	S-high	Ku	Ka-low	Ka-med	Ka-high
Antenna Diameter, m	1.0	2.0	4.9	1.0	2.0	4.9
Antenna Efficiency, %	55	55	55	55	55	55
System Noise Figure, dB	5.5	5.5	5.5	5	5	5
Transmitter Power, dBW	13	13	16	13	13	16

TDRS II Forward Link

Band	S-low	S-high	Ku	Ka-low	Ka-med	Ka-high
Coding Rate	None	None	1/2	1/2	1/2	1/2
Data Rate, Mb/s	0.02	0.08	1	1	5	25
RF Wavelength, cm	15.0	15.0	2.2	1.3	1.3	1.3
TDRS II EIRP, dBW	45.5	45.5	45.5	56.0	56.0	56.0
Space Loss, dB	-211.5	-211.5	-228.2	-232.5	-232.5	-232.5
User G/T, dB-K	-4.9	1.2	25.7	16.9	22.9	30.7
Eb/No, dB	12.3	12.3	10.2	8.2	7.4	8.1
Margin, dB*	0.2	0.2	3.3	1.3	0.5	1.2

(*Based on a 10^-5 BER, SGL S/N = 16 dB, and a 2.5 dB implementation loss;
    S- and Ku-band includes 3 dB of pointing loss)

WSC Forward Link

Band	S-low	S-high	Ku	Ka-low	Ka-med	Ka-high
Coding Rate	None	None	1/2	1/2	1/2	1/2
Data Rate, Mb/s	0.02	0.08	1	1	5	25
RF Wavelength, cm	15.0	15.0	2.2	1.3	1.3	1.3
WSC EIRP, dBW	45.5	45.5	45.5	66.0	66.0	66.0
Atmospheric Loss, dB	0.0	0.0	-2.8	-7.4	-7.4	-7.4
Space Loss, dB	-210.7	-210.7	-227.4	-231.7	-231.7	-231.7
User G/T, dB-K	-4.9	1.2	25.7	16.9	22.9	30.7
Eb/No, dB	15.6	15.6	9.6	12.4	11.4	12.2
Margin, dB***	3.5	3.5	2.7	5.5	4.5	5.3

TDRS II Return Link

Band	S-low	S-high	Ku	Ka-low	Ka-med	Ka-high
Coding Rate	1/2	1/2	1/2	1/2	1/2	1/2
Data Rate, Mb/s	0.1	0.4	75	20	80	350
RF Wavelength, cm	13.6	13.6	2.0	1.2	1.2	1.2
User EIRP, dBW	37.7	43.7	71.1	58.8	64.9	75.7
Space Loss, dB	-212.3	-212.3	-229.0	-233.5	-233.5	-233.5
User Prec, dBWi	-174.6	-168.6	-157.8	-174.6	-168.6	-157.8
Min. Req. Prec, dBWi **	-174.9	-168.9	-161.2	-175.0	-169.0	-162.6
Margin, dB	0.3	0.3	3.4	0.4	0.4	4.7

(**Based on TDRS DG2 QPSK BER 10^-5;
    S- and Ku-bands include 3 dB of pointing loss)

WSC Return Link

Band	S-low	S-high	Ku	Ka-low	Ka-med	Ka-high
Coding Rate	1/2	1/2	1/2	1/2	1/2	1/2
Data Rate, Mb/s	0.1	0.4	75	20	80	350
RF Wavelength, cm	13.6	13.6	2.0	1.2	1.2	1.2
User EIRP, dBW	37.7	43.7	71.1	58.8	64.9	75.7
Space Loss, dB	-211.5	-211.5	-228.2	-232.7	-232.7	-232.7
Atmospheric Degrad., dI	0.0	0.0	-2.8	-11.6	-11.6	-11.6
WSC G/T, dB-K	5.3	5.3	20.0	40.3	40.3	40.3
Eb/No, dB	10.1	10.1	10.0	10.5	10.5	14.8
Margin, dB***	3.2	3.2	3.1	3.6	3.6	7.9

(***Based on QPSK BER 10^-5 and 2.5 dB implementation loss;
    S- and Ku-bands include 3 dB of pointing loss)
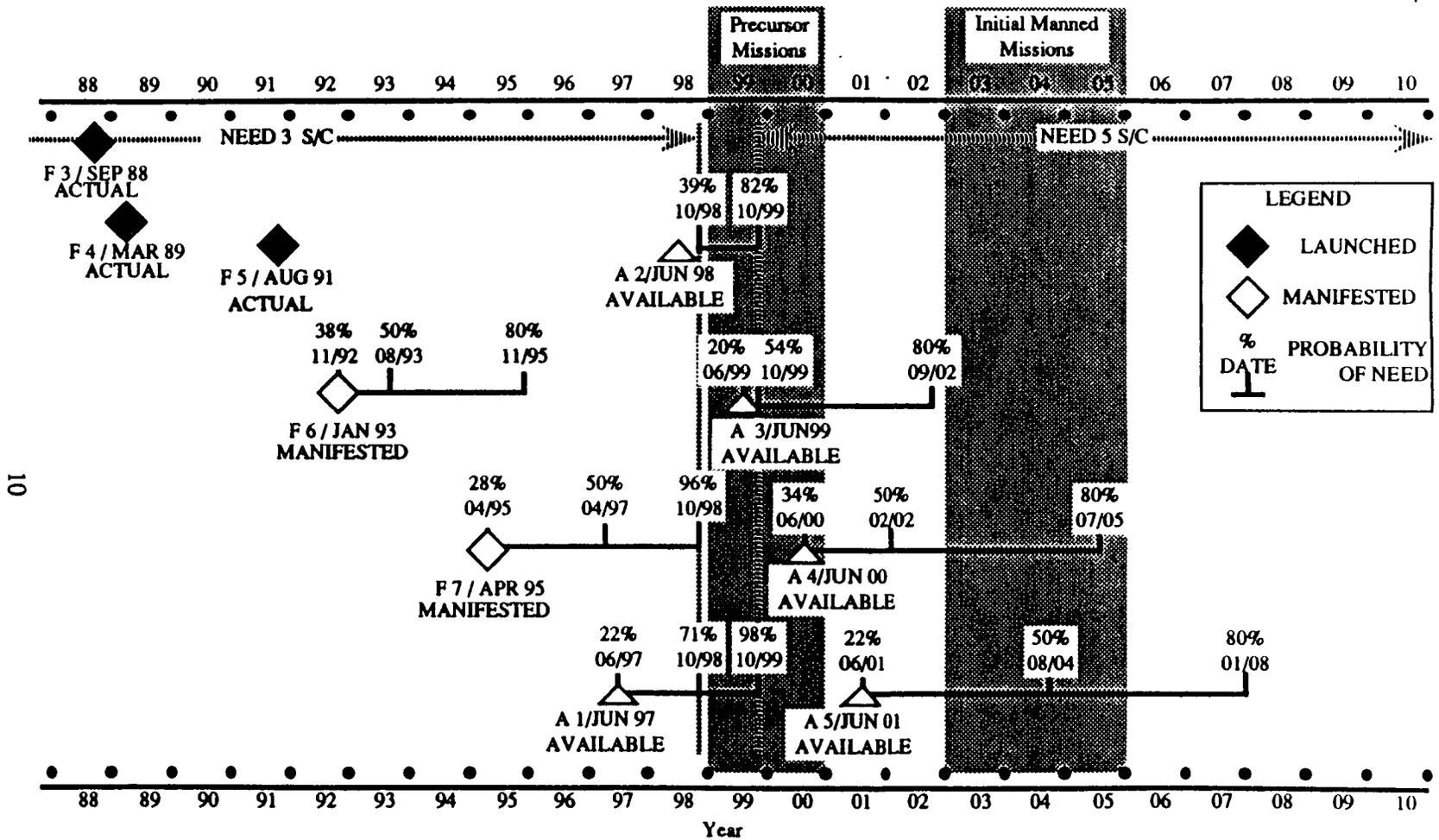
Table 1: Sample Link Budgets for Lunar Users

Figure 7. Schedule

In an augmented architecture with one operational satellite in the central location in addition to the baseline architecture of four operational satellites plus one spare the additional ground segment acquisition and ten-year operations and maintenance costs of two antennas (prime and backup) and a redundant equipment chain are about $100 million. The additional space segment costs are approximately $400 million to $600 million for two additional satellites and launches.

The new configuration architecture requires two additional remote ground terminals in order to provide TT&C support for the two TDRS II satellites which are relocated in order to close the ZOEs. Data is relayed to the WSC either by inter-satellite links between the TDRS II, a ground hop using the TDRS II, and leased commercial lines. Based on the incremental changes to the baseline TDRS II Phase A designs, the cost of this architecture is estimated to fall in the range of $150 to $250 million.

3 CONCLUSIONS

Several options for using the Space Network in support of lunar communications are available (see Table 2). The use of the Space Network as a basis for the lunar exploration initiative offers a low risk, evolutionary approach with maximum use of existing ground-based processing facilities developed in support of the Space Station Freedom program. In addition, the use of a single location for the ground terminal (i.e., the NASA facility at White Sands, NM) offers low operational costs.

The TDRS II mechanical design is well-suited for the lunar support if the augmented architecture is used. Maximum antenna elevation scan angles should be increased from 31° to 39° to ensure no drop-outs occur. The nominal TDRS II scan angle is sufficient except when the TDRS declination and the user declinations are at their extrema.

The addition of error correcting codes on the forward links and the highest data rate on the return links may be required.

The impact of the additional loading associated with the lunar initiative on the capacity of the TDRS II constellation, and the ability of the constellation to support the navigation requirements for lunar in-transit and surface operations requires further investigation.

The availability of a GEO slot between 102.5° W and 112.5° W longitude for operational use of the central spare in the augmented architecture should be investigated.

Possible regulatory conflicts associated with the use of the same frequencies for both space-to-space and space-to-ground links in the augmented architecture should be investigated.

Efficient use of the bandwidth available from the TDRS II may require multiplexing multiple low-rate users onto a single TDRS II single access service. Further investigations of the options and their implications on Space Network design and operations is required.

The impact of an operational spare on WSC operations should be considered.

Expansion room in the WSC infrastructure (e.g., common communications, depot, training facilities) sufficient to support the addition of the direct downlinks and the operational spare should be considered as a portion of the TDRS II development.

Table 2. Comparison of Space Network Constellation Architecture Options

Parameter	Near-Baseline	Augmented	New Configuration
S/C Antenna Scan Angles ±Az(E-W)/±El(N-S),°	77/39	77/39	77/38
Ground Segment	Nom	1 Ground Terminal	Remote Ground Terminals
Link Availability	80	100	100

4 LIST OF REFERENCES

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NASA

SATWG QFD Objectives

(1) Initiate a Cooperative Process for Continual Evolution of an Integrated, Time Phased Avionics Technology Plan

- Involve customers, technologists, developers, managers

(2) Demonstrate Application of QFD to SATWG Technology Planning Process

- "Customer-Focused" Effort
- Training & Forum for Teamwork
- Inputs to a "Total" Multi-Project, Multi-Year Plan Via QFD

(3) Demonstrate Computer Network Technology To Augment QFD Process

- Reduce Travel Costs and Make More Convenient to a Geographically Dispersed Team

SATWG Networked Quality Function Deployment

Don Brown

N92-23339

483

B-4

NASA

SATWG Networked
Quality Function Deployment (QFD)

Applying state of the art concurrent engineering tools to NASA Strategic Avionics requirements for the space infrastructure.

> QFD - Involves the development and prioritization of requirements for space systems, followed by analysis of technology areas to address these requirements.

> Voice and data networking are being used to facilitate the QFD process and involve technology users, developers, and program managers

General requirements have been defined for space systems, as well as program specific requirements.

These requirements are being correlated to system design characteristics that can be altered to meet requirements

> This analysis is useful when considering how to most effectively expend resources for the most value added technology advancement.

NASA

SATWG Networked QFD Participation

Facilitators:

-Martin Marietta, Denver

The QFD Team:

-Ames Research Center
-Johnson Space Center
-Kennedy Space Center
-Langley Research Center
-Lewis Research Center
-Marshall Space Flight Center

Other Sources of Input:

-Project Offices (NLS, ACRV, Commercial ELV)

-SATWG Panels (C&T, VHM, SAAP, GN&C)

SATWG Networked Quality Function Deployment

U.S. Govt.

SATWG QFD NETWORK

A-1a Matrix
"What" Evaluation

3/26/92

Code	Row label
A1	Level of Importance (Customer#1)
A2	Level of Importance (Customer#2)
A3	Level of Importance (Customer#3)
A4	Level of Importance (Customer#4)
A5	Level of Importance (Customer#5)
B	Industry Now
C1	1994-1996 Plan (CELV, ...)
C2	1996-1998 Plan (....)
C3	1998-2000 Plan (ACRV,....)
C4	2000-2002 Plan (NLS,....)
C5	2002-2004 Plan (....)
C6	2004-2006 Plan (....)
D	Ratio of Improvement
E	Sales Point
F	Absolute Weight
G	Demanded Weight

Customer Demands ("What's"):

Demand	NLS*	ACRV*	CELV*	B	plan	plan	plan	D	E	F	G
1.1 Be Insensitive to Schedule Chgs	4	3	3	1	7	7	7	7.0	1.0	3.0	3.7
1.2 Accommodate P/L Variability	4	1	4	1	3	1	7	1.0	1.0	1.0	1.2
1.3 Improved Maintainability	4	4	5	1	7	8	9	8.0	1.0	4.0	4.9
1.4 Robust System Performance	3	5	2	1	7	7	7	7.0	1.0	5.0	6.1
1.5 Launch & Perf Mission on Demand	4	5	2	1	5	9	7	9.0	1.0	5.0	6.1
2.1 Minimize Cost of P/L Increases	3	1	4	1	8	1	7	1.0	1.0	1.0	1.2
2.2 Meet Recur Cost Goal	5	1	5	1	6	1	7	1.0	1.0	1.0	1.2
2.3 Meet Non-Recur Cost Goal	4	3	3	1	8	7	9	7.0	1.0	3.0	3.7
2.4 Minimize Cost of Failure	4	4	5	1	7	9	7	9.0	1.0	4.0	4.9
2.5 Improve Design Process (PDT's)	4	1	4	1	6	8	7	8.0	1.0	1.0	1.2
2.6 Minimize Production Cost Risk	3	1	5	1	3	1	7	1.0	1.0	1.0	1.2
2.7 Minimize Ops Cost Risk	5	4	5	1	7	7	9	7.0	1.0	4.0	4.9
3.1 Be Reliable	5	5	4	1	5	7	7	7.0	1.0	5.0	6.1

Cust#1 Wt (W1)=	0.0%
Cust#2 Wt (W2)=	100.0%
Cust#3 Wt (W3)=	0.0%
Cust#4 Wt (W4)=	0.0%
Cust#5 Wt (W5)=	0.0%

$A = A1 \cdot W1 + A2 \cdot W2 + A3 \cdot W3 + A4 \cdot W4 + A5 \cdot W5$

D = Plan/Now (user selectable)

$F = A \cdot D \cdot E$

$G = F / (FTotal)$

487

SATWG Technology

A-1b-2 Matrix
"What" vs "How" Correlation

Substitute Quality Characteristics ("How's"):

Customer Demands ("What's"):	DEMANDED WT	3.1 Adaptive Guidance/Landing	3.2 Auto Nav, Rendezvous & Capture	3.3 Advanced Sensors/Effectors	3.4 Advanced Algorithms, S/W	3.5 (blank)	4.1 Adv'ed Antenna Systems	4.2 Adv'ed Transmit/Receive Systems	4.3 Adv'ed Digital Signal Processing	4.4 BW Efficient Modulation/Coding	4.5 Adv'ed RF/Op Tracking Sensors	5.1 Power Source	5.2 Power Distr/Cond
1.1 Be Insensitive to Schedule Chgs	3.7				1	3							
1.2 Accommodate P/L Variability	1.2												
1.3 Improved Maintainability	4.9												
1.4 Robust System Performance	6.1												
1.5 Launch & Perf Mission on Demand	6.1												
2.1 Minimize Cost of P/L Increases	1.2												
2.2 Meet Recur Cost Goal	1.2												
2.3 Meet Non-Recur Cost Goal	3.7												
2.4 Minimize Cost of Failure	4.9												
2.5 Improve Design Process (PDT's)	1.2												
2.6 Minimize Production Cost Risk	1.2												
2.7 Minimize Ops Cost Risk	4.9												
3.1 Be Reliable	6.1												
3.2 Be Safe	6.1												
3.3 Improve Product Quality	6.1												
3.4 Achieve Program Dependability	6.1												
4.1 Improve System Modernization Cap	3.7												
4.2 Improve Visibility Into System Health	6.1												
4.3 Optimize P/L Accommodation	1.2												
4.4 Optimize Mission Accommodation	2.4				3	3	9						
4.5 Man-Rateable	3.7												
4.6 Provide Benign Environment	1.2												

Study Objectives
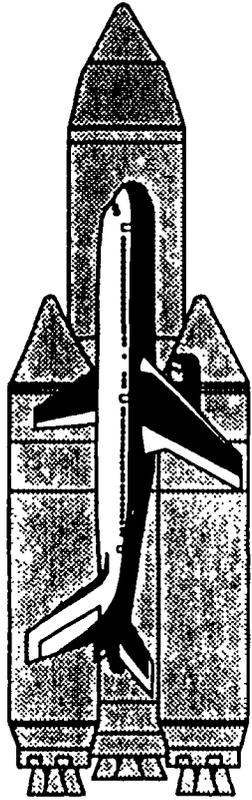
Boeing 777
ARINC 651

**Will commercial
avionics do the job?**

**Improvements
needed?**

N92-33340

Systems and Research Center

Honeywell

C910929 33

Program Objectives

COTS
FOR SPACE

Advanced Avionics
Johnson Space Center

- Greatly improve national launch capability
- Reduce operating costs
- Improve reliability, responsiveness, and mission performance

Lessons Learned and Study Reviews

Requirements

Mission Definition
Functional Requirements Performance Requirements Environmental Requirements Operational and Maintenance   Requirements Cost

Honeywell COTS Avionics

Architecture and Implementation Strawman

Boeing 777
AEEC/ARINC 651

Technology Shortfalls/Needs

Requirement X

Product B Product A

Requirement Y

Development Directions and Recommendations

Develop Tradeoffs
Systems Simplification Concepts
System Packaging Approaches
Life-Cycle Cost Validation

Architecture Requirements Comparison

Requirements	Advanced Avoinics System Requirements	Nondevelopmental Item Design Requirements	
		Military	Commercial
Cost	Low	High	Low
Reliability	High	High	High
Maintainability	Space	Ground/war	Ground
Useful Life	90 days/20 years	20 years	20 years
Safety	High	High	High
Quality Assurance	High	High	High
Transportability	High	High	High
Health Monitoring	High	Medium	Medium
Commonality	High	High	High
Environmental			
• Acceleration	ETO	Maneuvers	Low
• Vibration	ETO	High	Low
• Shock	Docking/landing/ETO	Hits	Landing
• Electromagnetic	Space	Electronics	Electronics
• Thermal	$2800 W/m^2$	Medium	Medium
• Contamination (dust)	High	Medium	Low
• Meteorite	Yes	No	No
• Radiation	GCR/SPE	No	No
• Partial vacuum	Yes	Some	No
Performance	High	High	High
Physical			
• Size	Small	Small	Small
• Mass	Light	Light	Light
• Power	Low	Low	Low
Materials	Outgassing	Minimum	Minimum
Autonomy	High	Medium	Low
Reusable/Expendable	Both	Reusable	Reusable
Manned/Man-Rated/Unmanned	Man-rated	Manned	Manned
Growth/Flexibility	High	Medium	Medium

Shaded areas indicate significant differences between commercial requirements and space requirements of this study

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Architectural Approach

- **Flexible integration—from transitional COTS+ integration to COTS+ framework with a minority of space-qualified products**

- **Universal applications**
  - **All future manned/unmanned space missions/vehicles**
  - **Backward integration**
  - **Surface applications**

- **Universal architecture**
  - **Standard interfaces (multistandard)**
  - **Open architecture**
  - **Modular avionics**
  - **ARINC 651***
  - **LRU subsystems compatible**

- **Cost-effective**

* Special acknowledgment and appreciation is given to the AEEC for permission to incorporate ARINC 651 concepts into this study.

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C910920-43

Architectural Framework
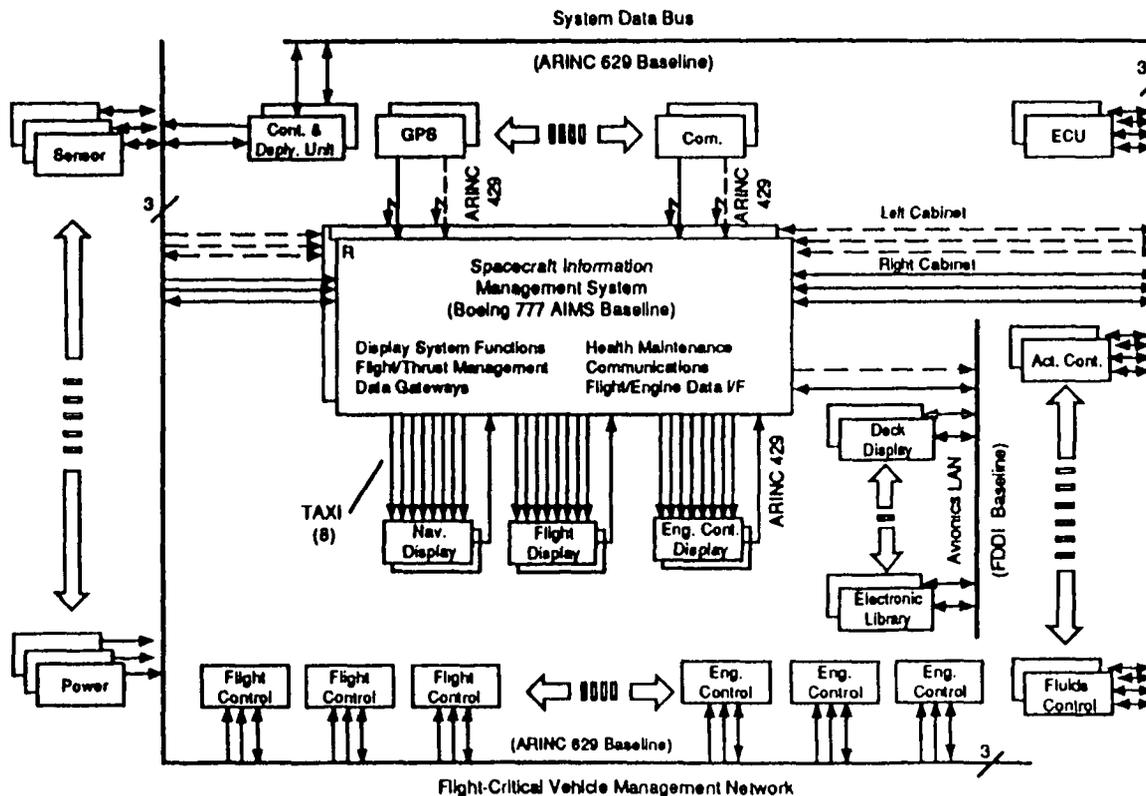
- **Scalable multinetwork/ bus architecture adapts to different missions/ vehicles and accommodates COTS interfaces**

- **Point-of-departure design is Boeing 777 architecture**
  - **Federated hierarchy**
  - **Distributed processing**
  - **Flight-critical, fault-tolerant ARINC 629 bus, ARINC 629 system data bus, high-throughput FDDI optical avionics LAN network, TAXI and ARINC 429 bus LRM interfaces**

Ultra-Reliability Requirement Implementation

COTS FOR SPACE

Advanced Avionics
Johnson Space Center

Problems

- Ultra-reliable avionics are desirable for extended and deep-space missions. These missions will degrade avionic reliability due to increased galactic and solar radiation explosure.
- Ultra-reliability (fail-op, 10 failure probability) may not be practical to meet. This normally brings up fault tolerance, fault detection, fault isolation, and recovery (FDIR) design issues.
- Parallel redundancy and/or pooled sparing redundancy using multipurpose electronics modules will be required for longer duration missions. The cost to orbit and posit powered redundant avionics is undesirable.

Solution

- Define architecture to allow cold sparing and hot/cold insertions
- Define new deferred maintenance architecture (e.g., quad channel) and operations (e.g., <10-day deferral) for cold sparing. In this solution, maintenance is used for cold spare insertions. This eliminates the cold spare technology gap identified in other studies.
- Specify cold storage in areas other than user chassis (may be protected storage areas)

Benefit

- Ultra-reliable reliability
- No technology gap

Pooled Hot Spare
Channel A
Channel n
Failed Module
Power

Channel A
Cold Spare Storage
10-Day Maximum
Time to Replace
Channel n
Failed Module

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C910929 93
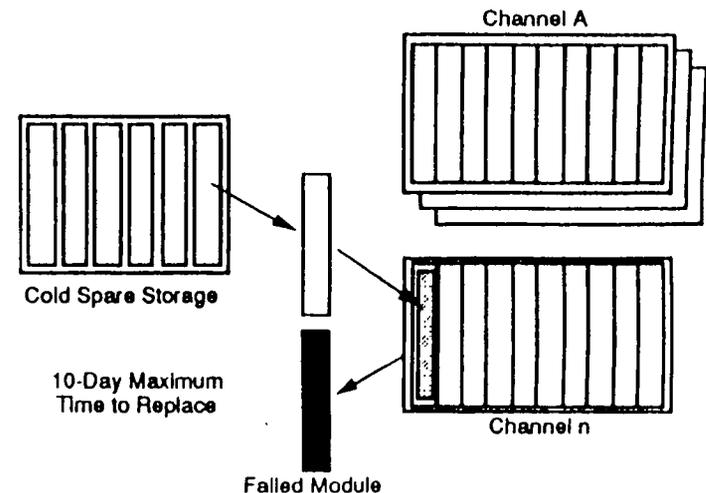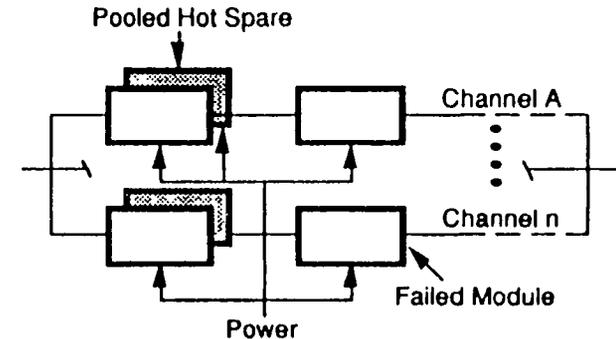
Summary of Results

- A COTS+ avionic system concept was defined and implemented within space avionic architectures

- System specifications, avionic architecture, and COTS+ qualification were modified as required to accommodate COTS+ integration; a relatively small set of COTS+ unique "needs" (or "changes" requiring further development) were identified
  - Distributed, physically partitioned avionics
  - Acceleration qualification as required
  - SEU recovery
  - Humidity and salt spray qualification
  - Acoustic qualification as required
  - Storm cell and safe for cold spares on extended-duration missions
  - Qualification by analysis for outgassing effects
  - Partial vacuum testing as required
  - Configuration and parts control

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Summary of Results
(continued)

- One concern (a possibly unaccommodating need) was identified—radiation tolerance of commercial parts
  - Possibility of substituting radiation-tolerant parts
  - Using radiation-protected equipment bays or spot shielding

- No technology gaps (development requiring significant involvement, funding, time lapse and/or risk) were identified

- COTS⁺ reduces space avionic needs and technology gaps
  - COTS⁺ is available by definition
  - Space avionics requirement study
    - --Red (technology gap) 13 →7
    - --Yellow (development required) 101 →41
    - --Green (current/near-term technology) 29 →95

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Conclusions and Recommendations

- COTS provides the space program many attractive benefits
  - Reducing technology gaps
  - Cost-effectiveness
  - Dependability
  - Delivery timelines

- The COTS+ concept for space avionic architectures appears workable in part or in totality. Radiation tolerance of COTS+ products is a concern requiring further assessment. It is recommended that the development needs identified within the study be addressed in ongoing studies. Sponsorship of standards, initiation of technical maturation programs, and further studies are recommended.
  - AHP utility assessments are suggested
  - Deferred maintenance concept should be validated
  - COTS demonstrations are encouraged
  - Transitional integration is suggested
  - A comprehensive development plan is suggested as a next step

Common Manned Module
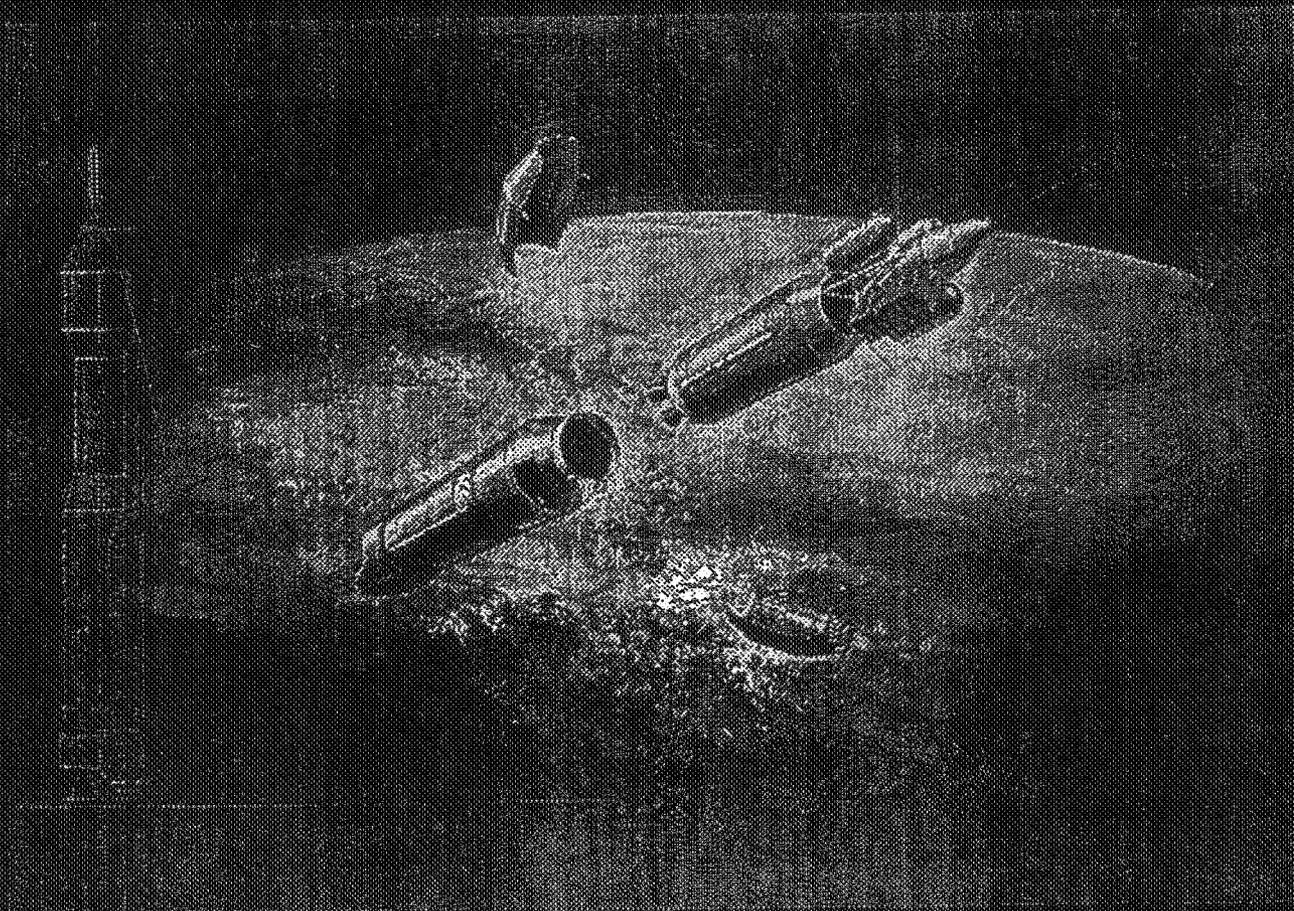
Space Exploration Initiative
Third Technical Interchange Meeting
University of Houston - Clear Lake
May 6, 1992

Combined Lunar Crew Module Studies

Economic analyses say reusable single crew module reduces costs of Moon to stay exploration program by $5B

- What does that module look like?
    - Aerodynamics?
    - Weights?
- Runway landing or prepared site recovery?
- Reusable or replaceable TPS?
- What technologies should be supported now?

Can a common module satisfy all manned reentry requirements?

Moon

Midcourse Correction
$\Delta V = 10$ m/sec

TransEarth Injection Burn
$\Delta V = 1000$ m/sec for 3 day transit
$\Delta V = 1160$ m/sec for 2 day transit

Earth-Moon Line

Earth

Latitude for KSC

Locus of potential Landing sites

Perigee Point
Latitude varies with Epoch Date
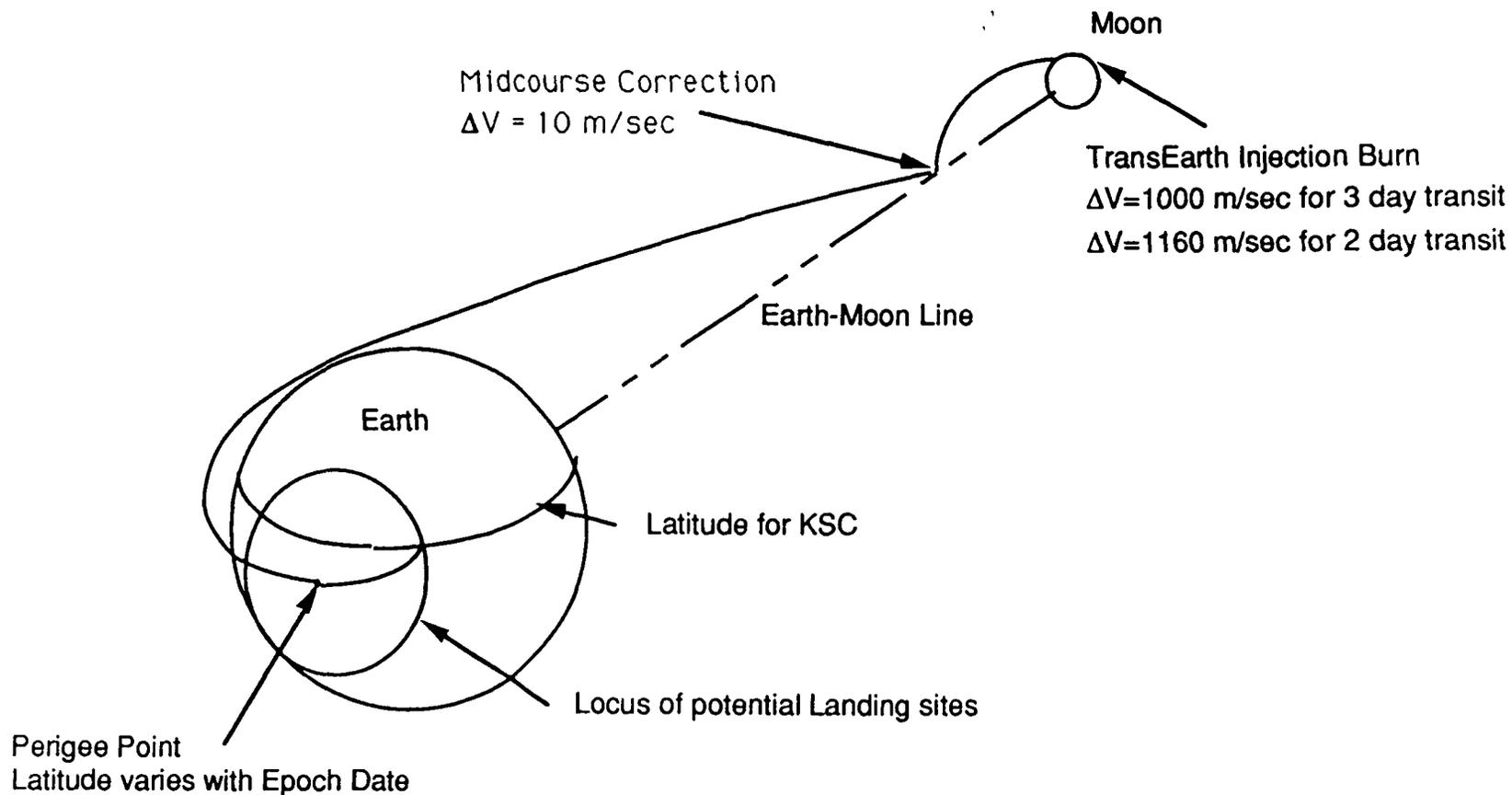
Return to KSC can be initiated on any lunar orbit

STV

MSFC - BOEING

Ballistic Parachute Earth Landing System
Reference LTS Crew Module Concept

Parachute compartment with retro-rocket or other terminal decelerator system

Reusable Crew Module containing Avionics, EPS, ECLSS & RCS

Tiled TPS outside edge

Land on spherical heat shield structure with omni-directional crushable honeycomb shock attenuation system

Nonreusable ablator TPS segment discarded for landing

Windowed "dormer" containing pilot's station and lunar access hatch

0.5 L/D AFE style disposable heat shield

1.12 m

34.00°

0.39 m R

1.42 m R

50.00°

26.50° RAKE ANGLE

3.84 m
3.53 m

4.42 m

L/D = 0.5 LCRV Nominal Trajectory

altitude versus time
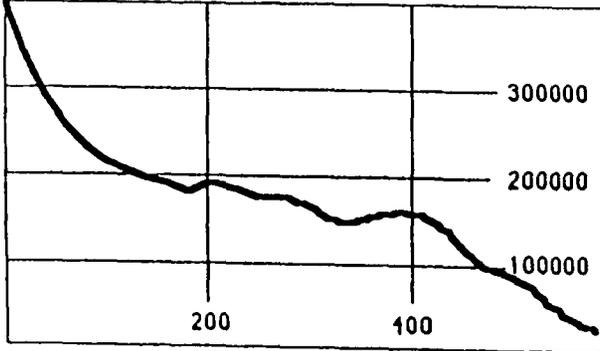
angle of attack versus time
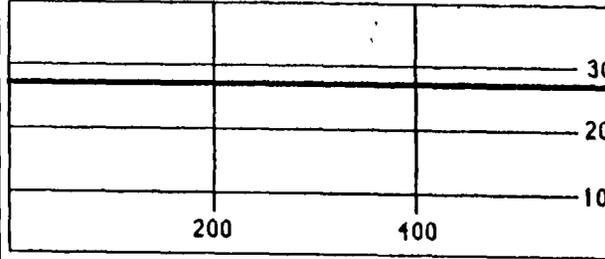
Rev B Trajectory

atmospheric velocity versus time
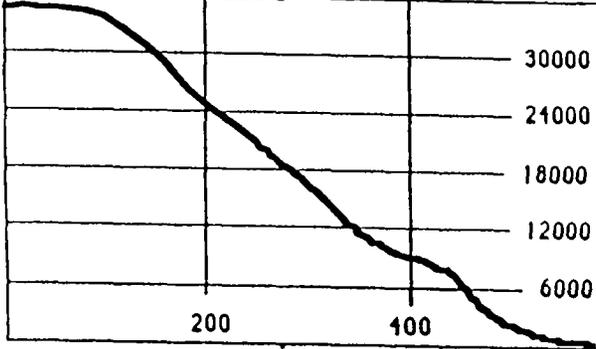
bank versus time
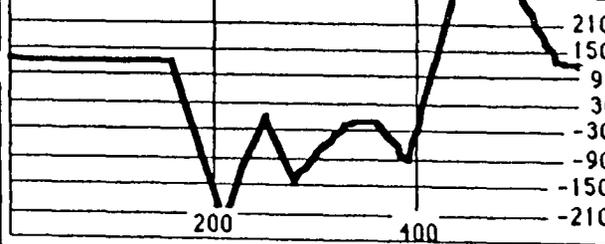
acceleration G's versus time
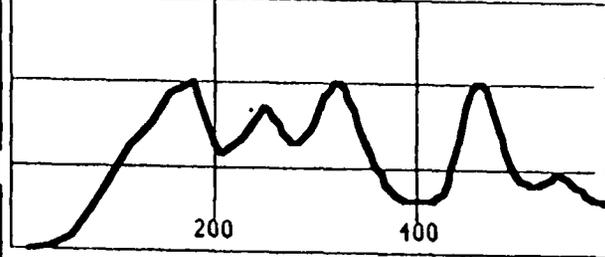
(4 G Nominal Limit)

geodetic latitude versus longitude
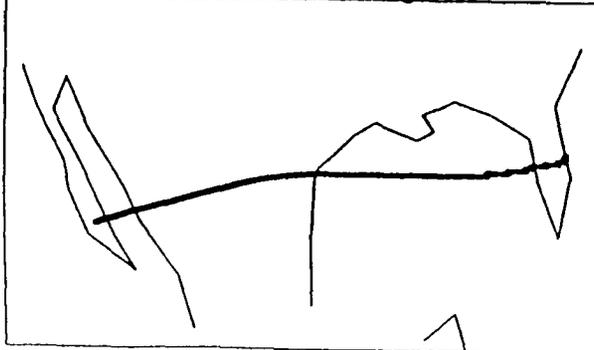
dynamic pressure versus time
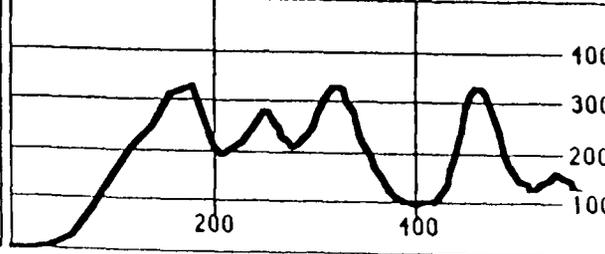
Equilibrium Temperatures for Common Manned Module During Lunar Return Reentry

Emissivity = 0.8
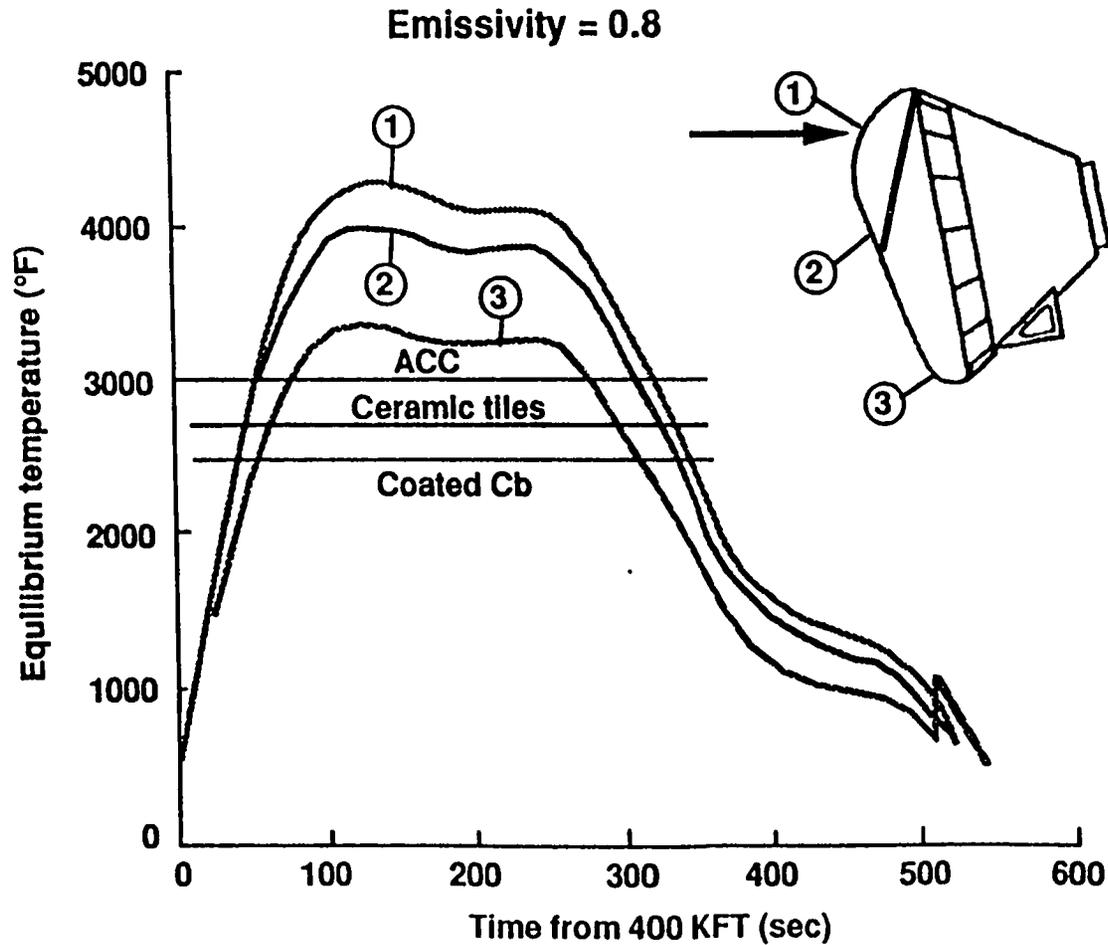
L/D = 0.5 LCRV Reentry Accuracy

3 DOF Guided results
100 Random Gram atmospheres

North

Final Conditions:
Altitude   = 25000 ft
Longitude = 80.582 W
Latitude   = 28.5 N

Rev B Trajectory
(4.0 G Nom Limit)

East

0.5 NM

1.0 NM

1.5 NM
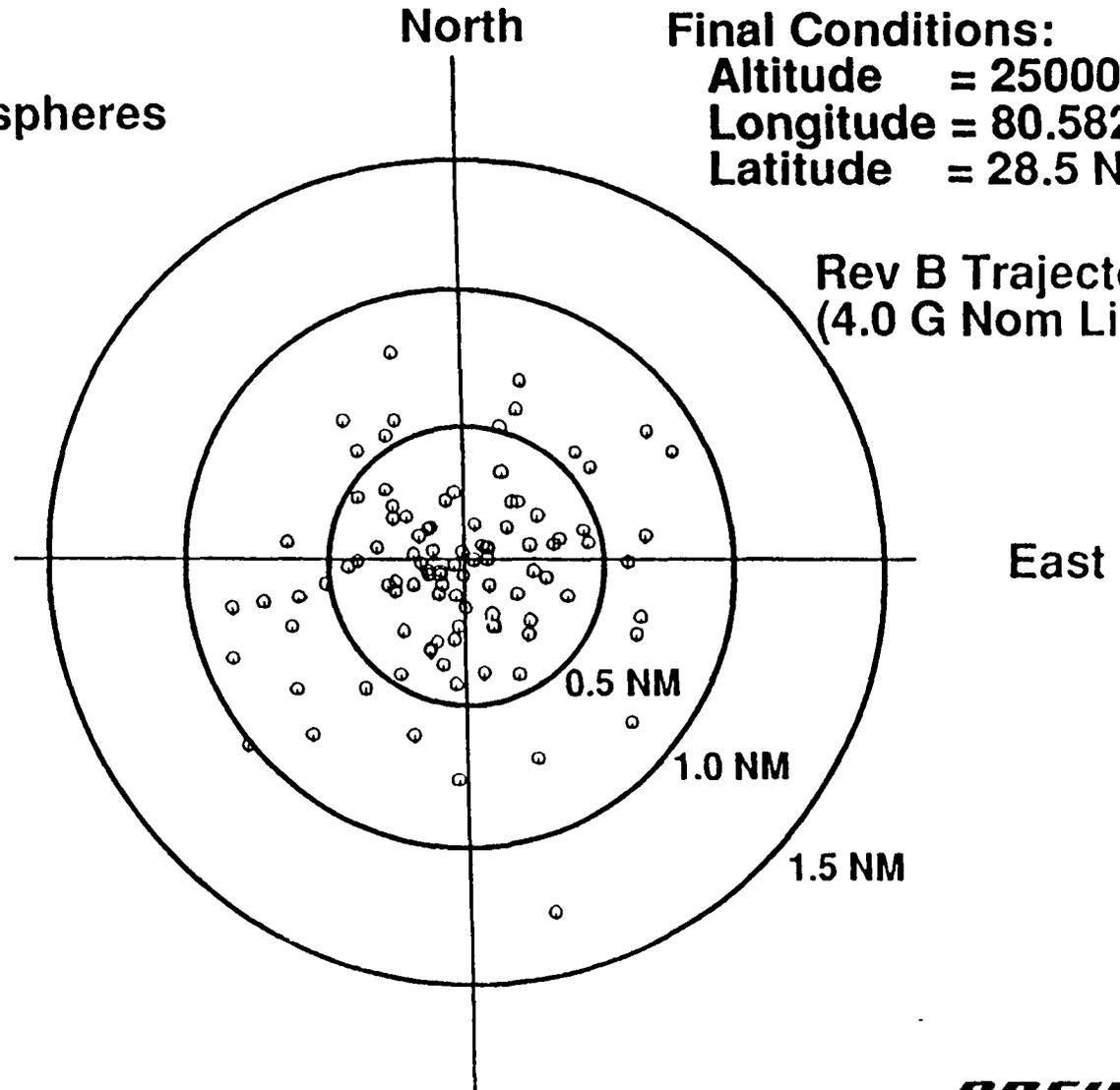
L/D = 0.5

Midcourse Correction Accuracy

**Last Midcourse Correction
66,300 km radius**

**Errors at Midcourse Correction
affect entry conditions:**

**Entry flight path
angle effect:**

**Entry
400,000 ft
Altitude**

Velocity magnitude   0.17 deg entry fpa / mps

Flight path angle velocity error   0.59 deg entry fpa / mps
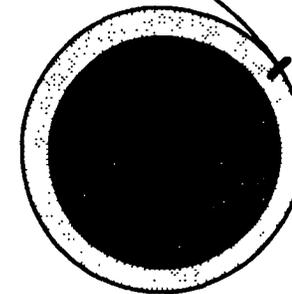
**Expect eventual reentry targeting to support
entry flight path angle errors of 0.25 to 0.5 degrees**

**(6DOF flown with 0.5 degrees, but not rigorously
tested in atmospheric dispersions over a wide
range of lunar reentry points)**

**Suggest midcourse correction
accuracy be on the order of:**

0.5 to 1 m/s

Three Future Manned Re-entry Systems Under Study

Personnel Launch System
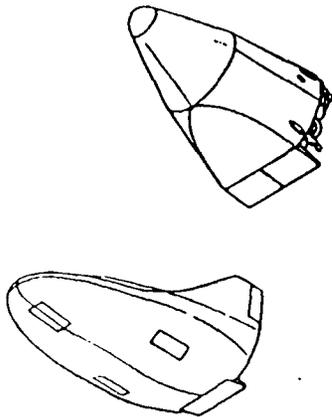
ACRV
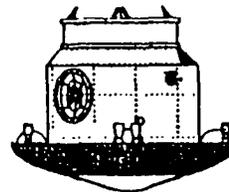
Common Lunar Lander
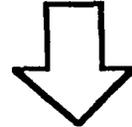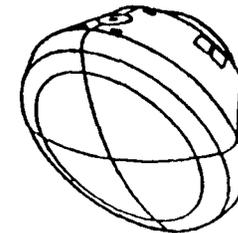
Common Manned Re-entry Module

Key design features

Avionics modules
- Plug-in cold spares option
- Optional leaves of redundancy
- Adaptive guidance algorithms

Common pressure shell
- Modular avionics suite
- Modular ECLSS
- Common RCS
- Add-on OMS
- Crushable honeycomb energy absorber
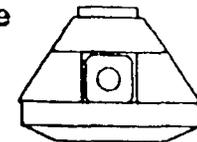
Replaceable heat shield
- LEO-return: High-temp, low-cost ceramic tile
- Lunar return: Low density hybrid ablator

BOEING
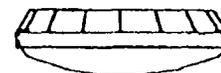
Space Transfer Vehicle

**Space Transfer Vehicle Concepts
and Requirements Study
(NAS8-37856)**

Third
Space Exploration Initiative

Multi Stage Lunar Lander
Storable vs Cryo Propulsion

University of Houston
06 May 1992
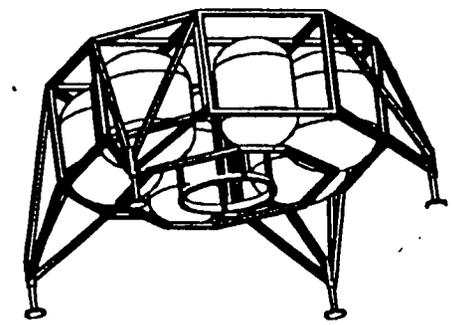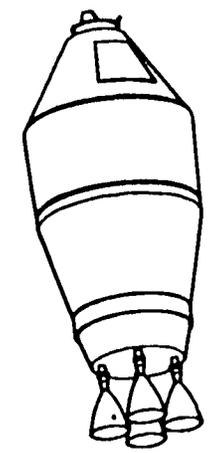
John R. Hodge (303) 977 - 2792

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Groundrules and Assumptions

TLI Stage Interface

- **Post TLI Payload Capability Is 76 t**
  - Current Baseline Is 93 t

Element Design

- **The Return Stage Will Have the Capability of Bringing 200 kg of Cargo Back to Earth**
- **The Lander Will Have the Capability of Delivering at Least 27.5 t of Cargo or 5.0 t of Cargo and Crew of Four to the Lunar Surface**
  - Current Cargo Baseline Is 31 t
- **Crew Module Mass Is 9.2 t (including radiation shielding & consumables)**
  - Current Baseline Is 6 t

440 Descent & 444 Ascent System Isps

Mission

- **4 Day Trans-Lunar/Trans-Earth Transfer Time**
- **45 Day Lunar Stay Time**
- **4 RL10-A3 Engines (Thrust = 16,300 lbs)**
- **15% (of Total) Trapped and Residual Propellants**

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Performance Parametrics

MSFC

Configuration	TLI Mass	Comments
• EXPO Reference	91 t	• Cryo Descent/Storable Ascent
• Cryo 2 Stage Lander	78 t (23% Savings)	• All Cryo • Cost Benefits - Propulsion Commonality

• Cryo 1-1/2 Stage Lander	75 t (26% Savings)	• All Cryo
		• Cost Benefits
		  - Propulsion Commonality
		  - Fewer Program Elements

Piloted Cargo vs. TLI & Crew Cab Mass

- 1.5 Stage All Cryogenic Lander
- Direct/Direct Mission Scenario
- 4 x RL10A-3 Engines
- Ballistic Earth Return
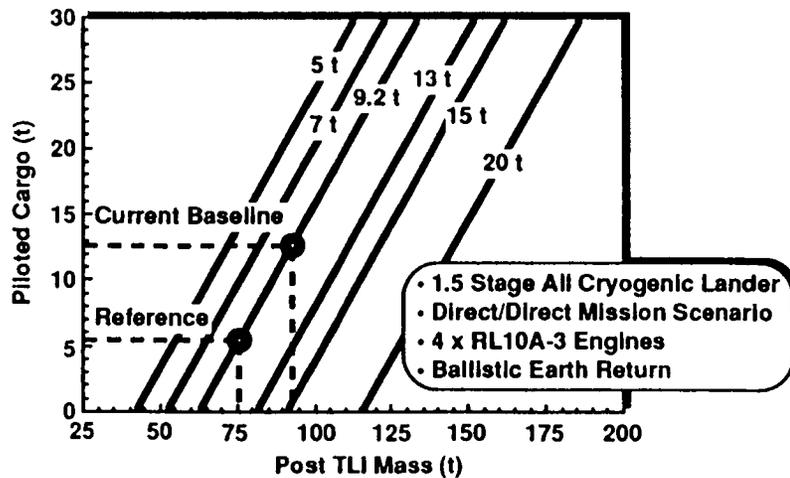
Lunar Surface Cargo Only vs. Post TLI Mass

- 1.5 Stage All Cryogenic Lander
- Direct/Direct Mission Scenario
- 4 x RL10A-3 Engines

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003
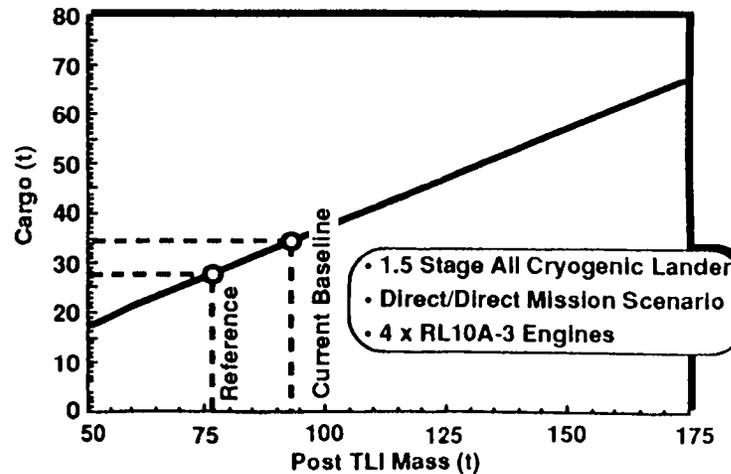
Comparison of the Lander Configurations

MSFC

Option	Advantages	Disadvantages	Mass Reduc.	Fraction
#1	• Close To Surface P/L Platform • Packages in 33 ft Dia. • Few Strct Mod's For Cargo Mission • Conventional Tank Mounting	• Large # of Tanks • Large Surface Area / Volume Ratio	0.0%	0.848
#2	• Close To Surface P/L Platform • Packages in 33 ft Dia. • Few Strct Mod's For Cargo Mission • Fewer # of Tanks	• Additional Baffles & Acquisition Device Work for Tanks • Non Conventional Tank Mounting	7.7%	0.855
#3	• Lower Structural Mass • Packages in 33 ft Dia. • Fewer # of Tanks	• Increased Thermal Leak From Tank Attach Structure • No Infinite Plane Cargo Deck • Complex Mechanisms	1.6%	0.849
#4	• Moderately Lower Structural Mass • Packages In 33 ft Dia. • Fewer # of Tanks	• High Thermal Leak From Tank Attach Structure • Complex Ascent Adaptor • Non-Conventional Tank Mounting	6.9%	0.855
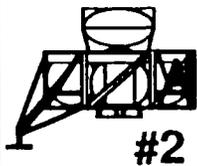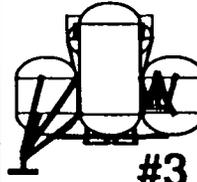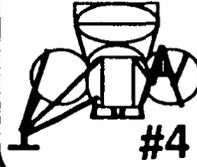
Recommendation: Option #1 as Baseline & Option #2 as Alternate

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004

SE920317-05A

510

Recommended Configuration

Piloted & Cargo Vehicles with a Common Lander

Piloted
Earth Capture
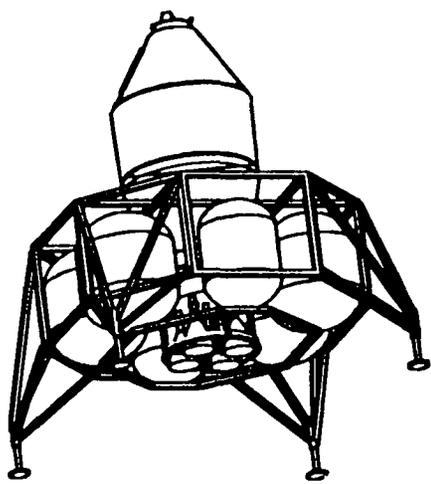
Piloted
(Descent)

Piloted
(Ascent)

Cargo
(Descent)

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005
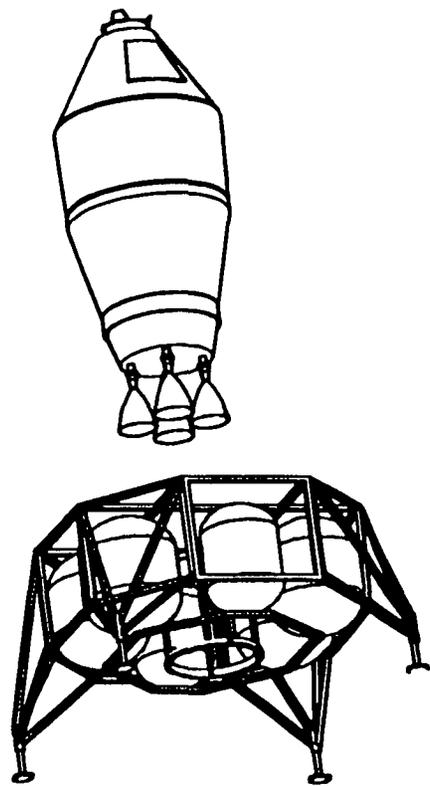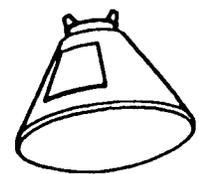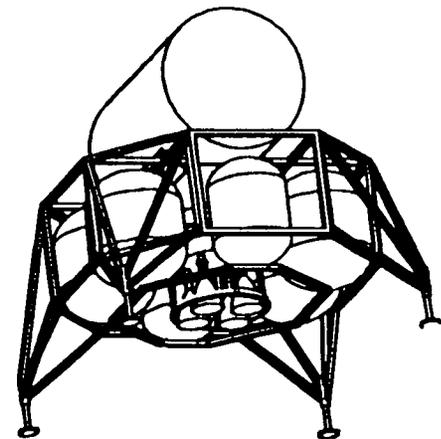
Configuration Analysis

Piloted Mission Payload Capabilities

9.4 m

3.6 m

53°

4.0 m

2.0 m

>1.5 m
Clearance

Ascent Guide Rails

Launch
Fairing

Payload Volume
≈ 70 m^3

Payload Volume
Subset (hxw = 2.4
m x 1.0 m) ≈ 32
m^3

Piloted Mission

Payload Fairing

6.4 m
(Landing)

4.9 m
(Post-TLI)

>1.5 m
Clearance

Base of Lander
Leg (Surface)

Cargo Mission

Hab Module

6.9 m
(Landing)

5.2 m
(Post-TLI)

P/A Module From
Ascent Stage
(some Geometry
Removed For Clarity)
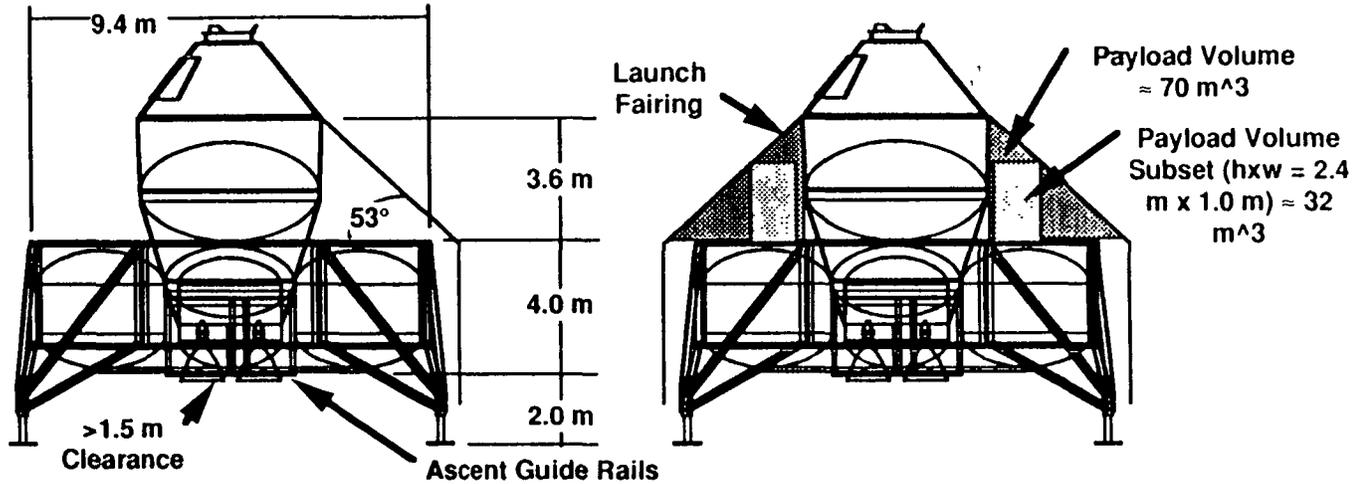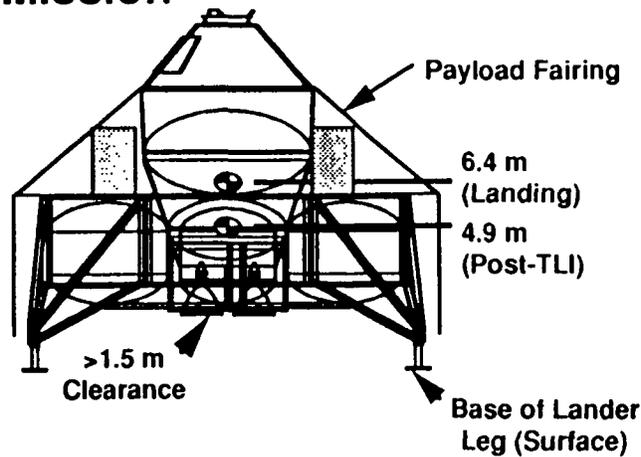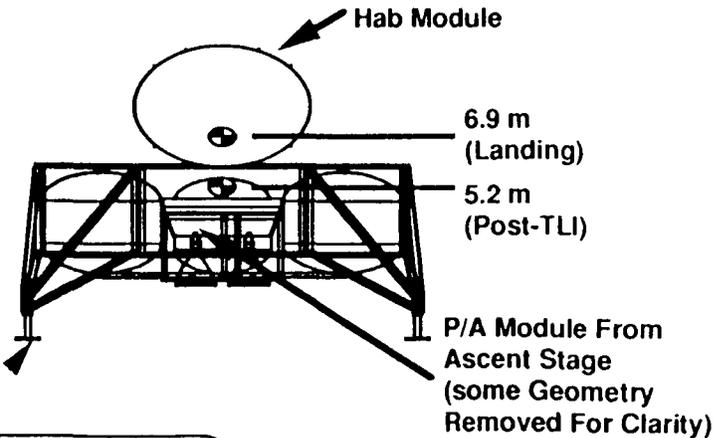
- All CG Locations Are From Base of Landing Leg (Surface)
- Landing Leg Pad Diagonal Diameter 13.95 meters

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006

RS920302-REV

512

Operations - Landing Debris

Concern -

Undesirable Effects
from Debris Generated
by Engine Exhaust
Impingement
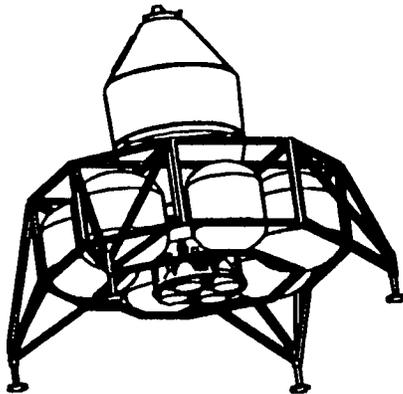
Obscuring of Landing Site, Making Hazard
* Avoidance and Navigation More Difficult

Throwing of Large Size Debris Significant
* Distances (Hundreds or Even Thousands of Feet).
Concern over Major Damage from Debris, or
Problems from Dust Contamination of Optics,
Windows, or Mechanisms on Existing Lunar
Installations.

Causing Damage to Critical Portions of a Lander,
* which May Interfere with a Safe Landing, or
Preclude Reuse as an Ascent Vehicle.

Mitigation -

Prepare a Hard Surfaced Landing Site as Soon as
* Possible
   - Eventually, Routine Flight Operations Should
    Use Prepared Sites

Place Engines in Close Proximity
*   - This Will Help Minimize the "Fountain" Effect,
    Which May Blast Lunar Debris at the Lower
    Side of the Lander

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007

MSFC

Concern

Separation Impingement Due to Launch Anomalies
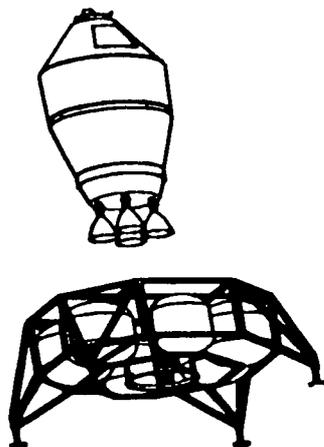
Mitigation:

- Simultaneous Start-Up and Desired Thrust Level Achieved in All Engines Prior to Separation.

- Guide Rails Attached to Lander For Controlled Separation of Ascent Stage Eliminating Impingement of Engine Bells With Surrounding Structure

- Additional Shielding May Be Added To Act As a Plume Deflector Forcing The Lander Onto The Surface, During Separation

Follow-on:

- Additional Analysis is Needed To Determined Effects of Timing Sequence During Engine Start-Up and Motion Realized By Lander

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008

JH920429-01A

Operations Analysis - Abort

- Abort Scenarios and Options Were Developed for Each Phase of the Mission:
  Pre-TLI to Lunar Landing to Earth Reentry

- Abort During Lunar Descent Is a Major Discriminator Between the 1.5 Stage
  and 2 Stage Systems in Regards to a Main Propulsion Failure
    - The 1.5 Stage Vehicle Has No Abort Option Available
    - The 2 Stage Vehicle Can Abort to LLO with the Ascent Stage
    - This Can Be Mitigated with Single Engine Out Capability

- Both Lander Options Cannot Tolerate an Engine Failure during the Lunar
  Ascent Phase of the Mission without Incorporating an Engine Out Capability
    - Also Would Give the 1.5 Stage System Engine Out Capability During
      Descent on the Piloted and Cargo Missions

The Single Propulsion Approach, Consisting Of 3 - 4 Engines And Single
Engine-out, Provides Both a Highly Reliable and Efficient System.

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009

Propulsion System Reliability

No Engine Out

Probability of Catastrophic Failure   0.05
Probability of Successful Switchoff   0.999

Single Engine Out

Probability of Catastrophic Failure   0.05
Probability of Successful Switchoff   0.999
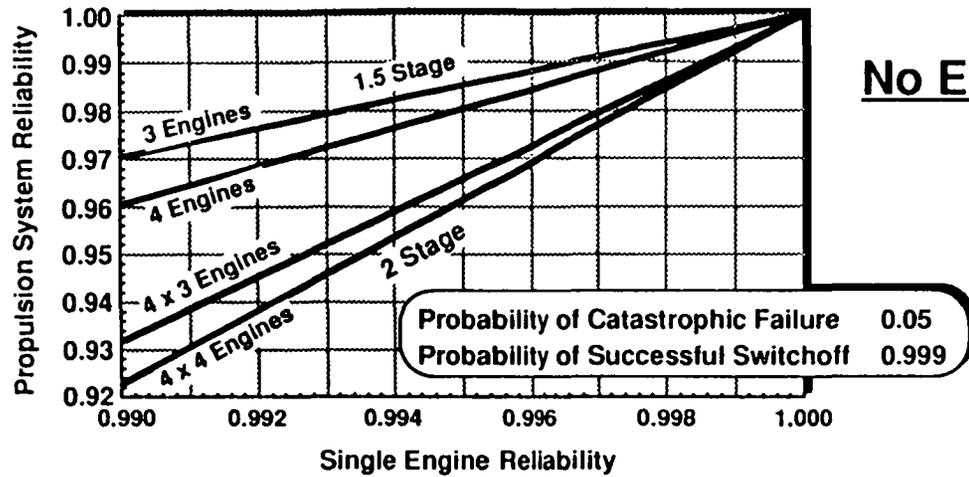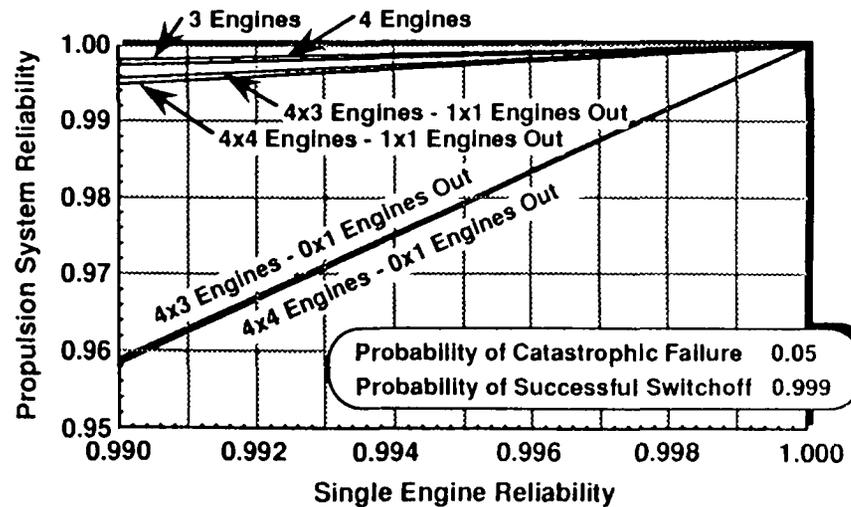
The 1.5 Stage System Will Have a Lower Probability of Propulsion Failure than the 2 Stage System

MARTIN MARIETTA

010

JH920428-05A

516

System Risk Assessment

• **Cryogenic Propellant Management (Ascent)**

Risk	Mitigation
• **Boiloff Of Critical Return Propellant** - Lunar Day/Night/Day (7% H2, 1.5% O2 per month)	• **Reduce Tanks' View To Direct Radiation** - Center of Vehicle - Combination Vapor Cooled and Debris Shield • **Separate Ascent and Descent Tanks** - Heavier Insulation Possible on Ascent Tanks
• **Pressure Build Up In Tanks** - Frozen Vents - Large Temperature Increases On Tank Surfaces	• **Backup Cryo Management Systems** - Redundant Pressure Relief - Redundant Vapor Cooled Shield Tubing • **Tankage Configuration Reduces Visibility to Heating Source**
• **Liquid Acquisition** - Problem Similar With Storable - LH2 & LO2 Difficult To Handle	• **Acquisition Devises Ensure Vapor-Free Liquid.** - Tank Head Idle - Paramagnetic

MARTIN MARIETTA

System Risk Assessment

• Single Propulsion System

Advantages

• Reduction In Engine Quantities

• Performance Gains with Higher Cryo Isp

• Reduction In Overall System Height

• Operational Confidence Due to LOI and Descent Burns.

Risks

• Disconnects With Lander Tankage Vulnerable to Leakage

• Increased Plumbing Complexities

• Higher Potential For Engine Damage During Initial Ascent

• Potential Damage Or Contamination During Final Descent

• Lander/Ascent Vehicle Clearance
  - Engine Gimbaling
  - Lander Deformation At Landing
  - Non-Vertical Ascent

• Release Failure of:
  - "Hold-Downs"
  - Fluid Disconnects
  - Electrical Disconnects

• System Restart Following Extended Surface Stay

MARTIN MARIETTA

012

JH920305-02A

518

System Risk Assessment

- ## RL-10A3 Cryogenic Engine

Risk	Concerns/Mitigation

• **System Restart Following Extended Surface Stay**
 	
  - **Lunar Day/Night/Day**
  - **Up to Three Starts Prior to Ascent**
  - **Temperature Differential Across Engines**
	• **Longest Period Between RL-10 Burns Has Been 24 Hours (In-Space)**
	  - **Titan/Centaur Operations:**
	    - **Ten Minutes Between First and Second Burns**
	    - **Several Hours Between Second and Third Burns**
	  - **Tested To 290°R (-170°F) With Successful Restart**
	
	• **Temperature Differentials Create Start Lags**
	  - **Centaur Specification = 700 ms Δ**
	  - **Colder Engine Slower To Start**
	  - **Controllability Impacted**
	  - **Thermal Control Proven (Passive/Active)**
	    - **Thermal Control Systems**
	    - **Centaur Roll Providing Uniform Heating (In-Flight Option Only)**

Single System Risks Are Mitigated By Design, Operation or Margin
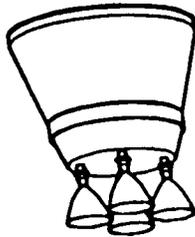
MARTIN MARIETTA

013

Subsystem Analysis - Multi Engine Start

Concern

**Startup Lag Times
Due To Multi Engine
Propulsion
Configuration**

Applicable Data: Saturn IVA

- **Engines = RL10A-3**
- **Quantities = 6**
- **Launch Date - Jul 10, 1965**
- **Verified Mission Event Sequence**
  - **Engine Chilldown = + 112.5 sec**
  - **# 3 Engine Ignition = + 151.49 sec**
  - **# 4 Engine Ignition = + 151.51 sec**
  - **# 6 Engine Ignition = + 151.52 sec**
  - **# 2 Engine Ignition = + 151.69 sec**
  - **# 1 Engine Ignition = No Time Available**
  - **# 5 Engine Ignition = No Time Available**

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520

014

JH920429-02A

Subsystem Analysis - Multi Engine Start
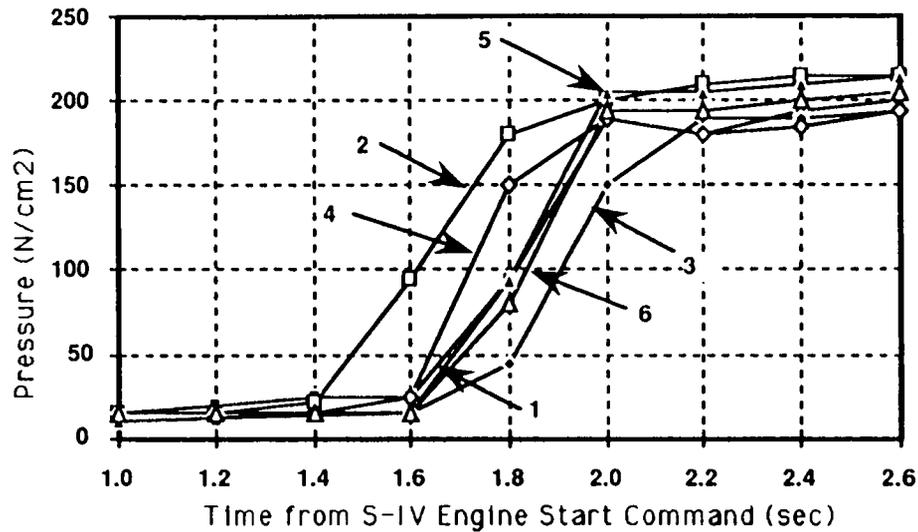
Individual Engine Start Transients

Engine Thrust Over Time
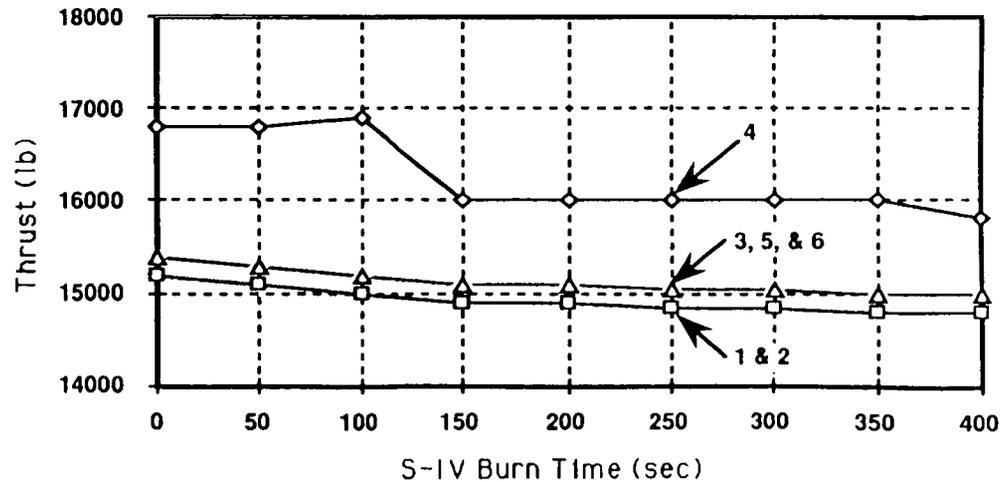
NASA LeRC ASAO STUDY

MAY 6, 1992

"BENEFITS AND TECHNOLOGY READINESS FOR USING CRYOGENIC INSTEAD OF STORABLE PROPELLANTS FOR RETURN MISSION FROM MOON"

DAVID W. PLACHTA
PHONE (216)433-6366

N92-33343

522

B8

The FLO Mission is Different Than Previous SEI Mission, and Has Different Cryo Requirements

Launch Strategy: Single launch, no LEO assembly, 4 day trip to moon.

- Zero-G pressure control and thermal control cryo issues are greatly minimized

Lunar Stay Time: 45-day maximum stay, and longer stays for future missions.

- Analysis, using actual test data, shows 34 layers of MLI on O_2 and H_2 tanks is adequate.
- Cryo insulation systems are evolvable to longer missions.

Lunar Direct Strategy: No rendezvous with orbiting vehicle, like Apollo.

- More ascent stage propulsion is required.

FLO cryo requirements are less demanding than previous lunar missions, e.g., 90-day study.

Cryogenic vs. Storable Propellant Ascent Stage
for Varying Stay Times on Lunar Surface

Analysis for FLO Mission
Weights from ExPO 4/2/92
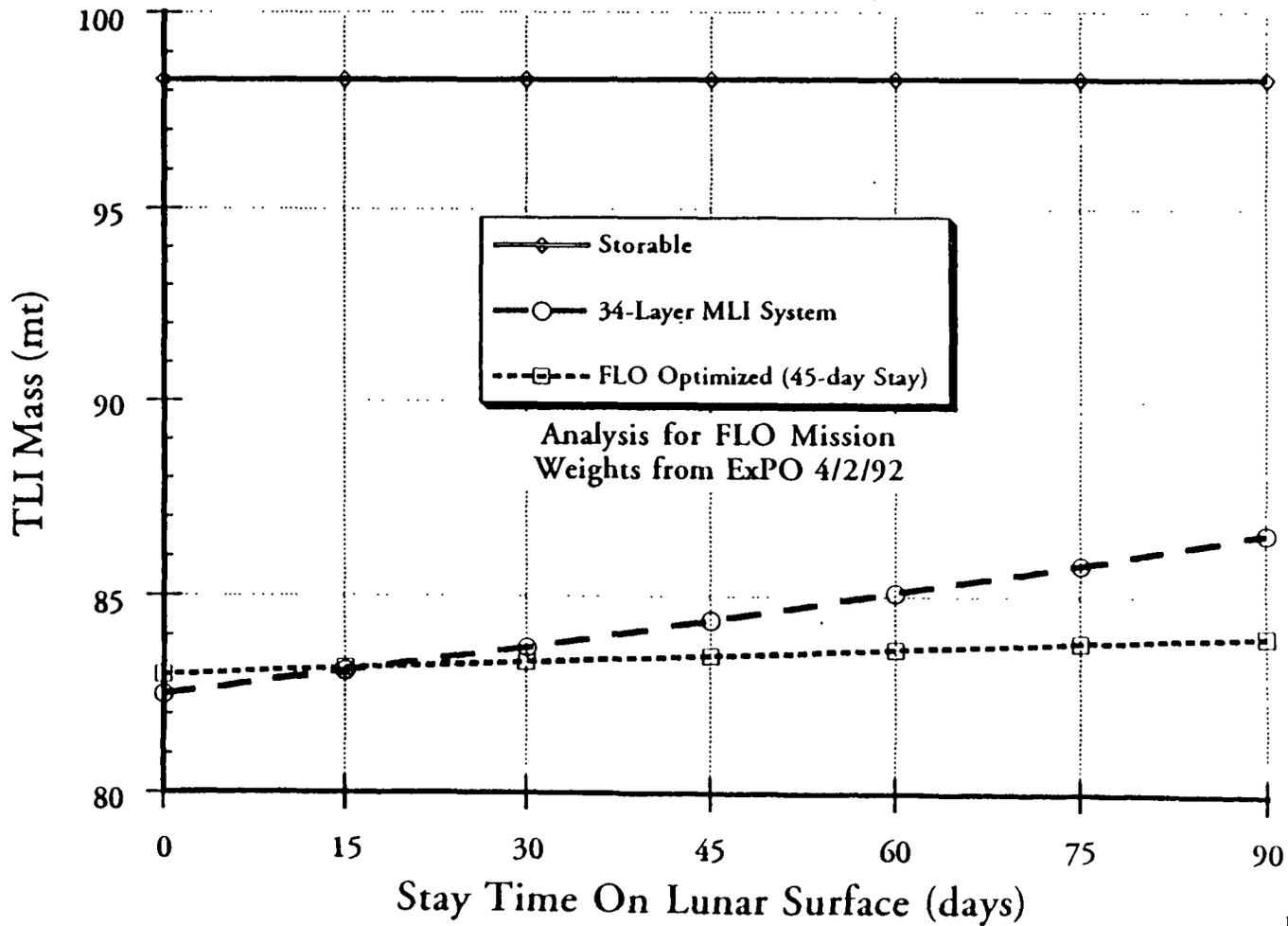
DWP 5/6/92

TWO CRYO THERMAL CONTROL SYSTEMS STUDIED FOR FLO

INSULATION SYSTEM	HEAT FLUX	DENSITY	WEIGHTED BOIL-OFF RATES (%/mo)
34 LAYER MLI SYSTEM	0.36 Btu/hr/ft^2, FROM K-SITE TEST DATA DURING SIMULATED LUNAR SUNRISE, 175 ft^3 TANK	0.3 lb/ft^2	4.8%
OPTIMIZED (TOTAL INS. WT. AND BOIL-OFF WT. MINIMIZED) 2" MLI ON H$_2$ TANK 3" MLI ON O$_2$ TANK	0.12 Btu/hr/ft^2, FROM EXTRAPOLATED TEST DATA AND LOCKHEED EQN. FOR H$_2$ TANK 0.08 Btu/hr/ft^2, FROM SAME EXTRAPOLATED DATA, FOR O$_2$ TANK	0.6 lb/ft^2, H$_2$ TANK 0.9 lb/ft^2, O$_2$ TANK	1.3%

HEAT FLUX ESTIMATES ARE BASED ON UNIFORM HEAT FLUX. PACKAGING OF TANKS WILL RESULT IN LOWER AVERAGE HEAT FLUX.

ANALYSIS OF TANK BOIL-OFF RATES

- ALL CRYO MISSION ASCENT STAGE PROPELLANT LOAD IS 12.7 MT, OR 28,000 LBS. ASSUMING A 5.3:1 MIXTURE RATIO, THEN

    - 4435 LBS. OF HYDROGEN, 1025 FT^3, AND 491 FT^2 SURFACE AREA
    - 23500 LBS. OF OXYGEN, 330 FT^3, AND 230 FT^2 SURFACE AREA

    BOIL-OFF RATE IS = q x A / h

    q= HEAT FLUX
    A= SURFACE AREA OF TANK
    h= HEAT OF VAPORIZATION

HEAT FLUX, $Btu/hr/ft^2$	H_2, LB/MO	%/MONTH	O_2, LB/MO	O_2, %/MONTH	WEIGHT AVERAGED, %/MONTH
0.36	680	15.4%	660	2.8	4.8%
H_2 0.12 O_2 0.08	227	5.1%	141	0.6%	1.3%

MISSION ANALYSIS GROUND RULES

PHASE	STAGE	Δ V (M/S)	INERT MASS (MT)
TLI	I	3200	0
LOI	II	892	12.2
LD	II	1830	12.2
LA	III	1810	5.22*
TEI	III	848	5.22*

- ISP 444
- NO LEO ASSEMBLY. 1 DAY LEO, 3 DAY COAST, 1 DAY IN LUNAR ORBIT
- 5 MT PAYLOAD DELIVERED TO LUNAR SURFACE
- 7.2 MT CREW CAB
- 2% PROPELLANT RESERVE ON EACH STAGE- LeRC INPUT
- 2% PER MONTH BOIL-OFF RATE FOR PHASES OTHER THAN LUNAR STAY- LeRC INPUT
- STORABLE ASCENT STAGE COMPARISON WAS BASED ON THE FOLLOWING:

    - ISP = 320
    - LD INERT MASS = 12.2 MT
    - LA INERT MASS = 5.22 MT

- ALL MASSES, DELTA V'S, AND Isp'S FROM JSC.

* MASS OF OPTIMIZED MLI SYSTEM ASSUMED TO BE INCLUDED

MLI FOR CRYOGENIC STORAGE

- THERMAL CONTROL OF CRYOGENIC TANKAGE FOR FLO WILL REQUIRE USE OF MLI

- MLI TECHNOLOGY IS IN PLACE FOR RELATIVELY SHORT MISSIONS ANTICIPATED FOR FLO (< FEW MONTHS). NO SHOW STOPPERS.

  - MEASURED PERFORMANCE IS ADEQUATE

  - EXTENSIVE EXPERIENCE EXISTS ON INSTALLING AND HANDLING MLI

  - METHODS OF APPLYING MLI TO WET-LAUNCHED LH_2 TANKS HAVE BEEN INVESTIGATED AND MLI CAN SURVIVE LAUNCH LOADS AND RAPID DECOMPRESSION DURING ASCENT

  - MLI SYSTEM SELECTED WILL ONLY REQUIRE SOME MINOR ENGINEERING DESIGN TO OPTIMIZE FOR GIVEN MISSION

- THERMAL CONTROL FOR LONGER MISSIONS (MONTHS TO YEARS) WILL REQUIRE:

  - EXTENSION OF MLI DATA BASE TO THICKER SYSTEMS FOR A RANGE OF BOUNDARY TEMPERATURES

  - THE ADDITIONAL USE OF OTHER THERMAL CONTROL TECHNIQUES SUCH AS VAPOR-COOLED SHIELDS, REFRIGERATION SYSTEMS, SHADOW SHIELDING, ETC.

THERMAL PERFORMANCE OF MLI HAS BEEN DEMONSTRATED FOR LH$_2$ TANKAGE

MOST MLI DATA IS FOR A HOTSIDE
BOUNDARY TEMPERATURE OF 530 °R

LH$_2$ BOIL-OFF AS A
FUNCTION OF HEAT TRANSFER
RATE & SEI TANK SIZES

CALORIMETER DATA
○ ◿ ◺ △ ◨ ◻ ◇ □

TANK APPLIED DATA
● ◗ ▶ ◀ ▼ ● ◢

ANALYTICAL - RADIATION
ONLY (ε = .05)

LOCKHEED EQ (ref. 5)

HEAT TRANSFER RATE, BTU/HR FT²

COMBINED H$_2$/O$_2$ BOIL-OFF = 6%/MON.

65,120 FT³ (WTR)

350 FT³ (TWO FOR CRYO-RETURN)

175 FT³

MLI HANDLING/INSTALLATION DEMONSTRATED FOR LH$_2$ TANKAGE
(LIST NOT INCLUSIVE)

- **LERC**

  - 175 FT^3 TANK   34 LAYERS (DAM & SILK NET SPACERS)   REPLACEABLE SYSTEM
  - 180 FT^3 TANK   30, 40, 50, 60 LAYERS (DAM & DEXIGLASS SPACERS)   "   "
  - 50 FT^3 TANK   34 LAYERS (DAM & SILK NET SPACERS)   "   "
  - 19 FT^3 TANK   18 LAYERS (SEMI-SYSTEM CO$_2$ FILLED)

- **MSFC**

  - 34 FT^3 TANK   15 LAYERS (DAM & DACRON NET) OVER FOAM SUBSTRATE
  - 450 FT^3 CALORIMETER   70 LAYERS (DAM & DACRON NET) - MACDAC SYSTEM

- **GDC**

  - 7.2 FT^3 TANK   128 LAYERS (~4" SUPERFLOC)
  - 175 FT^3 TANK   44 LAYERS (SUPERFLOC)
  - 1266 FT^3 D1-T FLIGHT TANK   3 TO 46 LAYERS (HE PURGED)
  - 1892 FT^3 SHUTTLE/G-PRIME TANK   3 LAYERS OVER 1.5" FOAM (HE PURGED)

- **LOCKHEED**

  - 180 F^3 TANK   30, 40, 50, 60 LAYERS (DAM & DEXIGLASS) FOR TESTS AT LERC

- **OTHERS**

  MOST AEROSPACE COMPANIES HAVE DIRECT MLI EXPERIENCE THAT HAS BEEN OR COULD BE APPLIED TO LH$_2$ TANKAGE

> EXTENSIVE EXPERIENCE EXISTS IN HANDLING AND
> INSTALLING MLI SYSTEMS
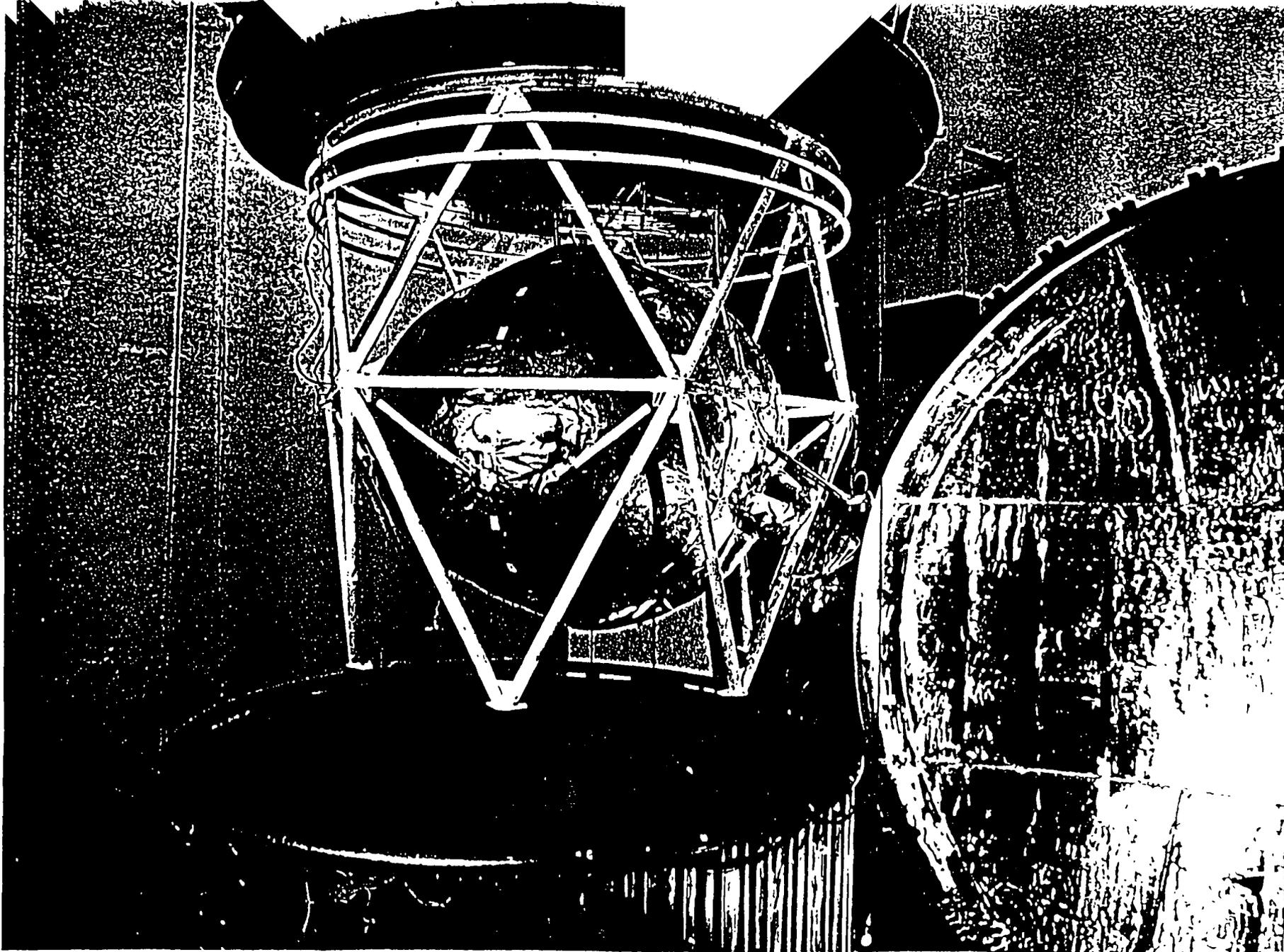
87 INCH DIAMETER MLI INSULATED LH2 TANK TESTED AT NASA LeRC K-SITE FOR BOTH NEAR-EARTH, LUNAR SURFACE AND DEEP SPACE CONDITIONS
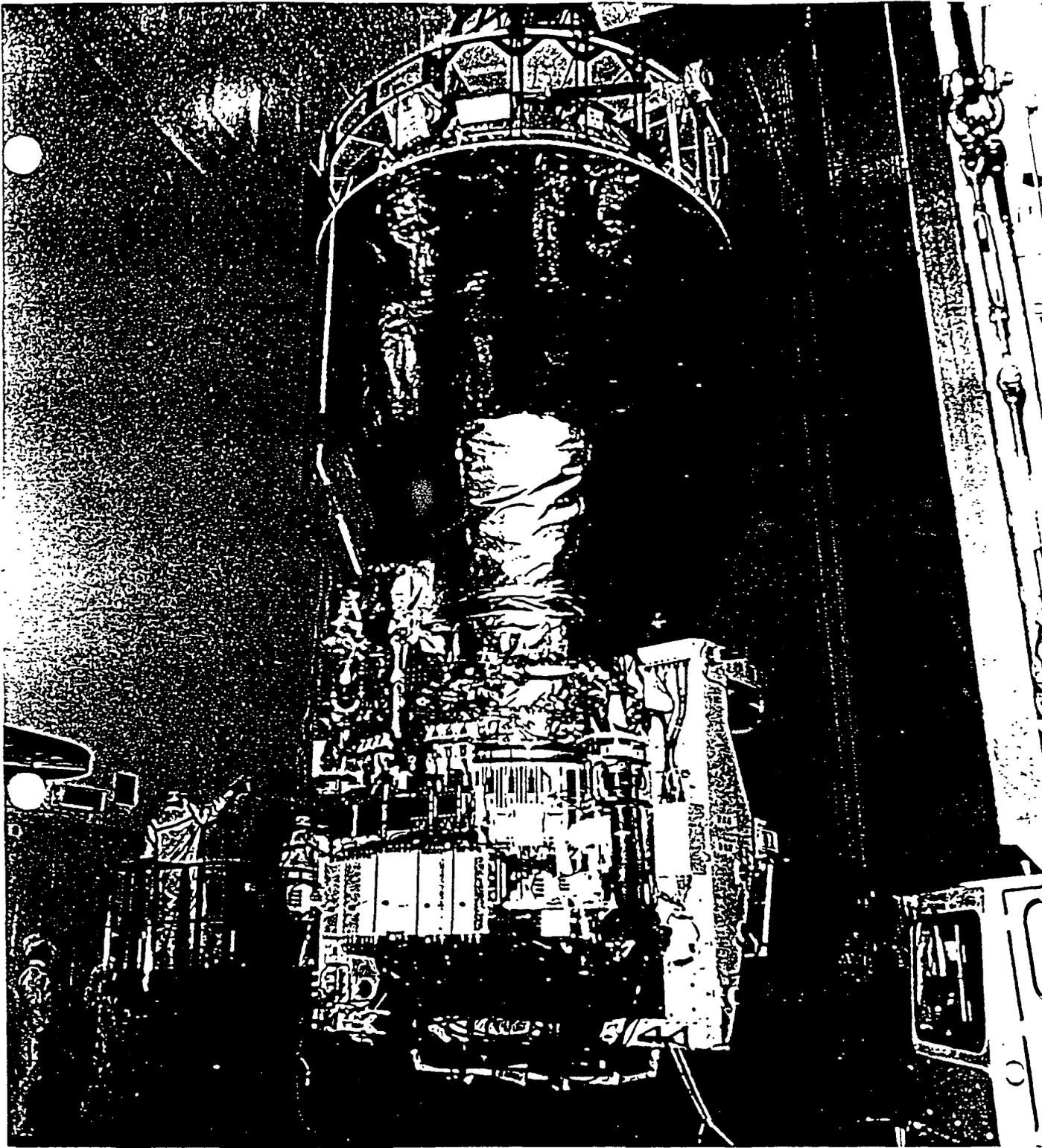
CENTAUR VEHICLES ARE ONLY FLIGHT APPLICATION OF MLI TO
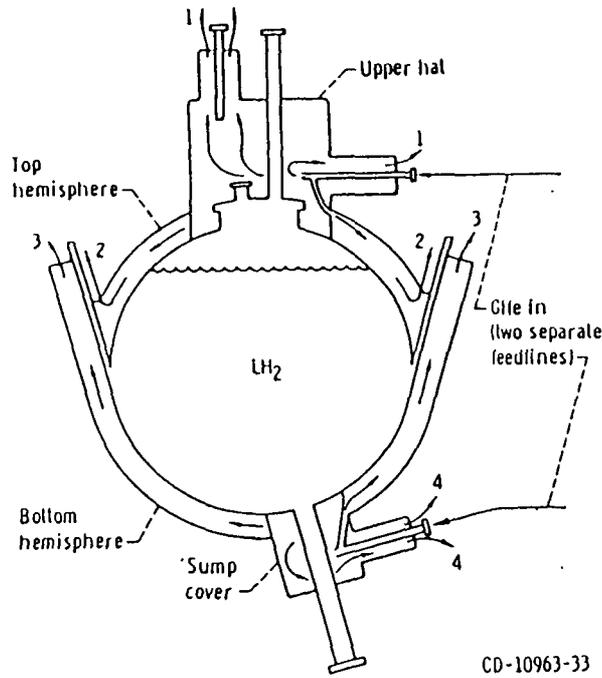LIGHT-WEIGHT CRYOGENIC TANKAGE

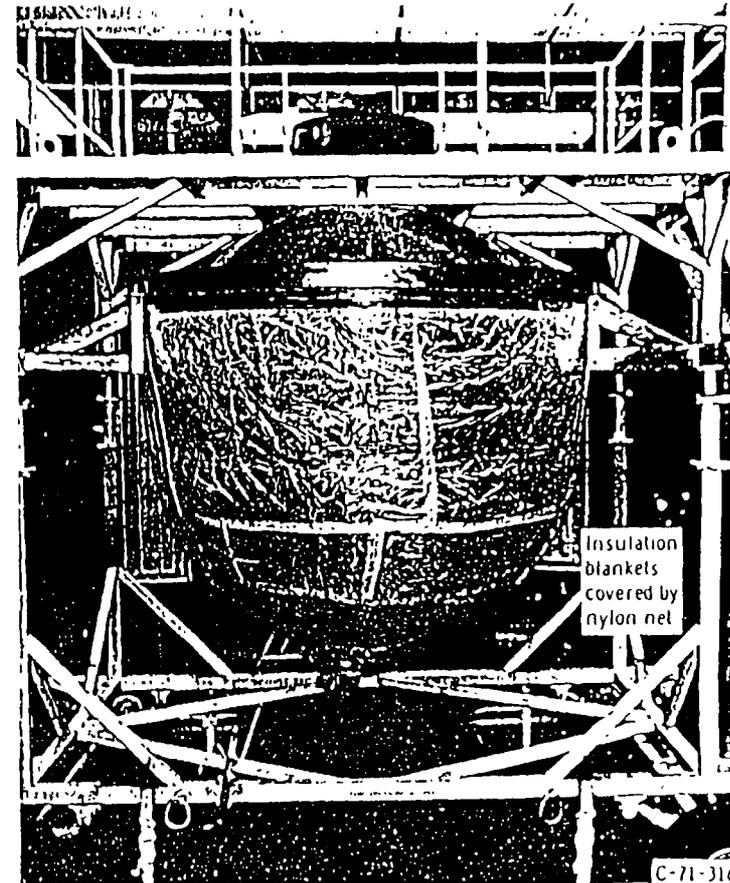Figure 3. - Ground-hold system and helium gas purge flow paths.

Figure 4. - Insulation blankets.

180 FT^3 LH$_2$ TANK TESTED AT LeRC --
30, 40, 50 AND 60 LAYER MLI SYSTEMS (DAM AND DEXIGLASS SPACERS)

GROUND PERFORMANCE AND ASCENT DEPRESSURIZATION EFFECTS
ON MLI HAVE BEEN DEMONSTRATED

LeRC

- $50 \text{ FT}^3 \text{ LH}_2$ TANK WITH 34 LAYERS MLI (FLIGHTWEIGHT SYSTEM)

  - SPACE TUG LAUNCH PRESSURE PROFILE
  - NO SIGNIFICANT THERMAL OR MECHANICAL DEGRATION AFTER 19 RAPID DEPRESSURIZATION CYCLES (12% DEGRADATION IN ON-ORBIT PERFORMANCE)
  - GROUND HOLD WITH HE PURGE $< 200 \text{ BTU/HR FT}^2$

- 30 INCH LH_2 CYLINDRICAL CALORIMETER

  - SIMULATED SATURN V PRESSURE PROFILE
  - TESTED 3 MLI CONCEPTS WITH NO DAMAGE NOTED
  - GROUND HOLD WITH HE PURGE GAVE:
    - 55 BTU/HR FT^2 WITH 30 LAYERS MLI AND 0.75" OF FOAM
    - 115 BTU/HR FT^2 WITH 30 LAYERS MLI AND 0.75" FIBERGLASS SUBSTRATE
    - 170 BTU/HR FT^2 WITH 30 LAYERS MLI AND 0.5" FIBERGLASS SUBSTRATE
    - NO MEASURABLE DEGRADATION

MSFC

- $450 \text{ FT}^3 \text{ LH}_2$ CALORIMETER WITH 70 LAYERS (MLI)

  - SIMULATED SATURN V PRESSURE PROFILE
  - SOME THERMAL DEGRADATION SUBSEQUENT TO DEPRESSURIZATION (.13 TO .195 BTU/HR FT^2 SPACE PERFORMANCE)

GROUND PERFORMANCE AND ASCENT DEPRESSURIZATION EFFECTS
ON MLI HAVE BEEN DEMONSTRATED

GDC

- 7.2 FT^3 LH_2 TANK (25" DIA.) WITH 128 LAYERS OF MLI (~4")

    - INSULATION SYSTEM PUMPED TO 1 TORR IN 70 SECONDS (MORE SEVERE THAN SATURN V)
    - NO DISCERNABLE DAMAGE MECHANICALLY OR THERMALLY
    - GROUND HOLD WITH HE PURGE GAVE 50 BTU/HR FT^2

- DEVELOPMENTAL TESTS FOR CENTAUR USING 9.5" DIA. LH_2 CYL. CALORIMETER

    - TESTED ALL MLI CONCEPTS USED UNDER SIMULATED LAUNCH-ASCENT

- FULL SCALE GROUND HOLD TESTS OF FLIGHT QUALIFIED SHUTTLE/CENTAUR G-PRIME

    - THREE LAYEERS MLI OVER 1.5 INCH FOAM
    - GROUND HOLD WITH HE PURGE < 100 BTU/HR FT^2

- 175 FT^3 LH_2 TANK WITH 44 LAYERS MLI (FLIGHTWEIGHT)

    - 100 FLIGHT CYCLES (RAPID DEPRESSURIZATION)
    - 26% DEGRADATION AFTER 50 CYCLES

GROUND PERFORMANCE AND ASCENT DEPRESSURIZATION EFFECTS ON MLI HAVE BEEN DEMONSTRATED

BALL AEROSPACE

- 150 LAYERS MLI (~3") MOUNTED ON 52 INCH DIA. TEST PLATE

  - SIMULATED STS CARGO BAY PRESSURE PROFILE
  - INSULATION SEALED ON 52" CIRCUMFERENCE TO FORCE EVACUATION THROUGH ONE LINEAR SEAM
  - NO DISCERNABLE DAMAGE TO INSULATION OR SEAM AREA
  - NO THERMAL VERIFICATION
  - CONCLUSION: TESTING DEMONSTRATED EDGE PUMPING ADEQUATE

LOCKHEED

- RAPID EVACUATION TESTS ON 27." DIA. MLI SAMPLES TO VERIFY ANALYSIS

  - SATURN V PRESSURIZATION PROFILE
  - 20 TO 80 LAYER SAMPLES
  - BOTH HE AND N_2 PURGES
  - GOOD CORRELATION WITH ANALYSIS

CONSIDERABLE EXPERIENCE EXISTS IN DESIGNING
WET-LAUNCHED MLI SYSTEMS FOR LH2 TANKS
EXPOSED TO LAUNCH PRESSURE PROFILES

Reasons for Changing Ascent Stage
Propulsion Baseline

- Using a cryo ascent stage to perform FLO yields-

    - At least a 15% reduction in TLI mass--using identical weights for the storable return case and the cryogenic return study.

    - The IMLEO savings would be 25 to 30%, enabling a 200 mt class launch vehicle.

- We know how to do this - how to insulate the tank and what the performance is.

    - The technical issues for FLO have been resolved by ground based testing.

Reasons for Changing Ascent Stage
Propulsion Baseline (cont.)

- An MLI system for the baselined cryo descent stage is needed for the 3-4 day trip to the moon-

  - Development work is needed for the descent stage MLI system, of perhaps 15 to 20 MLI layers.

  - The ascent stage MLI system will have more layers, but the design and manufacturing utilizes same technology.

- These MLI systems are evolvable to longer or more ambitious lunar missions and for Mars missions, with some technology development.

- LeRC and MSFC have planned test programs to test and study MLI systems for FLO and other SEI missions.

Is there any reason not to do this?

NASA/UNIVERSITY OF ARIZONA
SPACE ENGINEERING RESEARCH CENTER
FOR UTILIZATION OF LOCAL PLANETARY RESOURCES

PROPOSED LUNAR MISSIONS

- 65 kg Artemis payload: ISMU Pilot Plant

- 65 kg Artemis payload: Lunar Resource Prospector

- 200 kg Artemis payload: Integrated Oxygen Production Plant

N92-33034

ISMU PILOT PLANT FOR OXYGEN PRODUCTION

NASA/UNIVERSITY OF ARIZONA SPACE ENGINEERING RESEARCH CENTER FOR UTILIZATION OF LOCAL PLANETARY RESOURCES

(Contact: Ramohalli, 602-322-2304)

PRESENTATION ON FEBRUARY 4-6, 1992 TO:
Workshop on Early Robotic Missions to the Moon
Lunar and Planetary Institute, Houston, Texas

Mirrors Deploying

Secondary Mirror

Primary Mirror
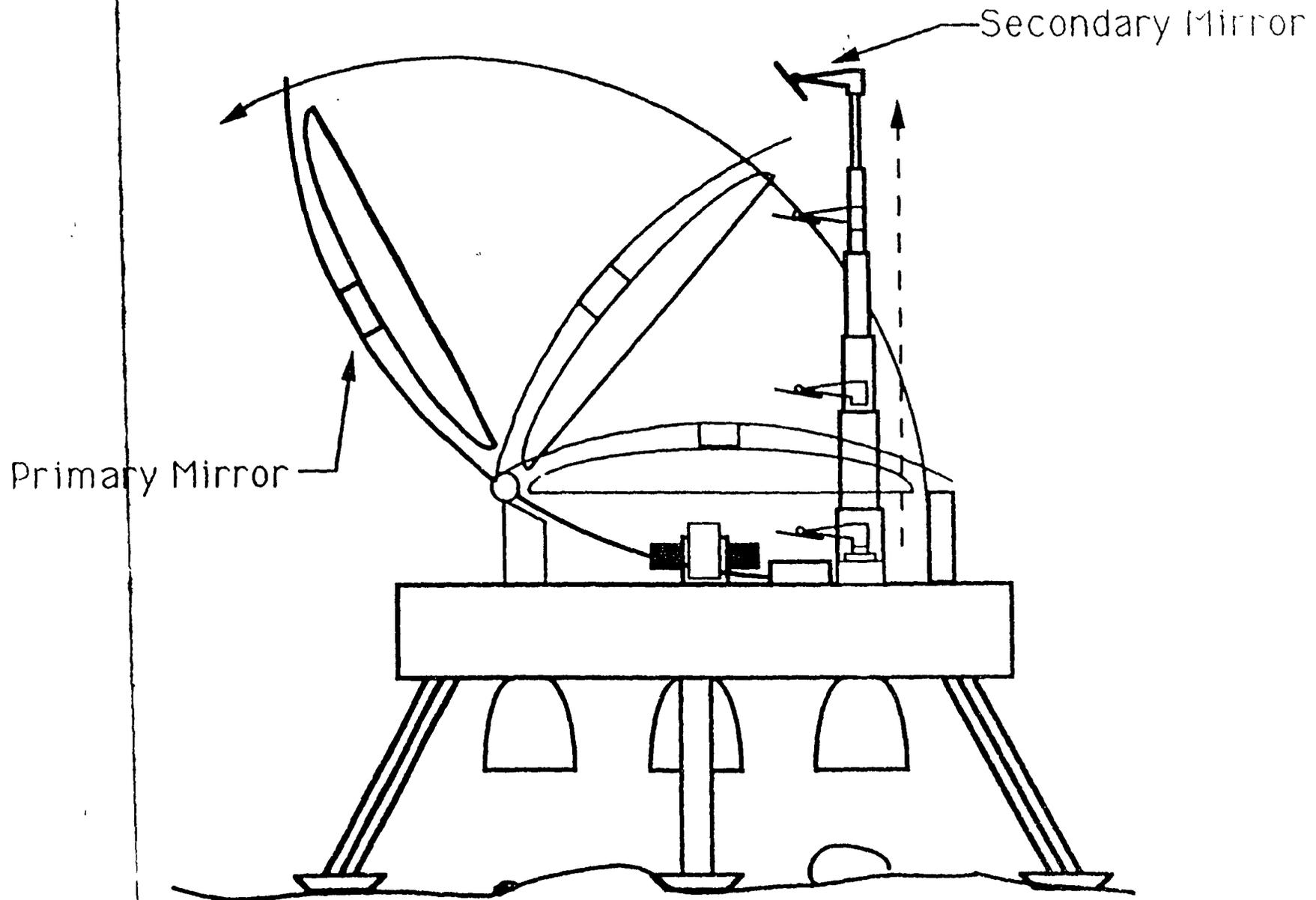
TASK DECOMPOSITION
INTEGRATED OXYGEN PRODUCTION

- **SOIL SAMPLE ACQUISITION**
  - move arm and gather soil
  - deposit in crucible through sieve

- **REACTOR OPERATION**
  - mix solid carbon powder with soil
  - insert crucible at the focus
  - control heating (mirror adjustment)
  - measure/identify gases
  - remove and store residue (tiles from slag)

- **DATA MANAGEMENT**
  - obtain measurements and store data

- **TELEMETRY AND UPLOAD**
  - adjust antenna/transmit data
  - upload code and data

UA SERC: KNRR

ACCURATE SCALE MODEL
COMMON LUNAR LANDER
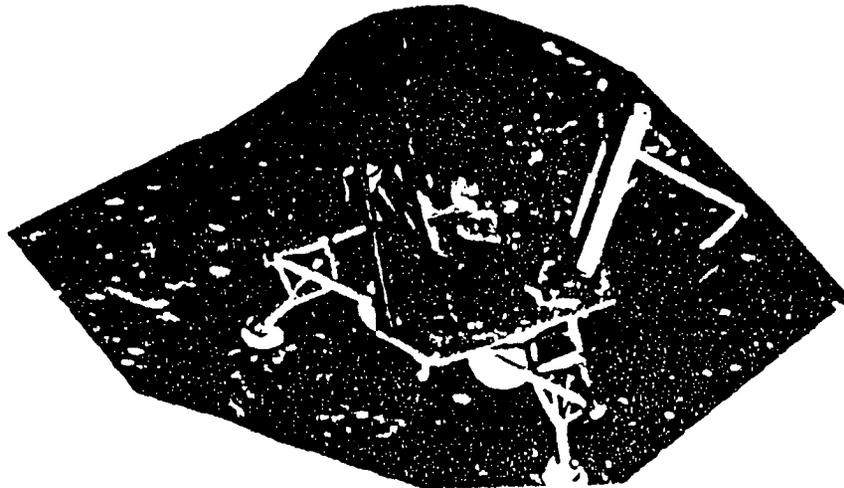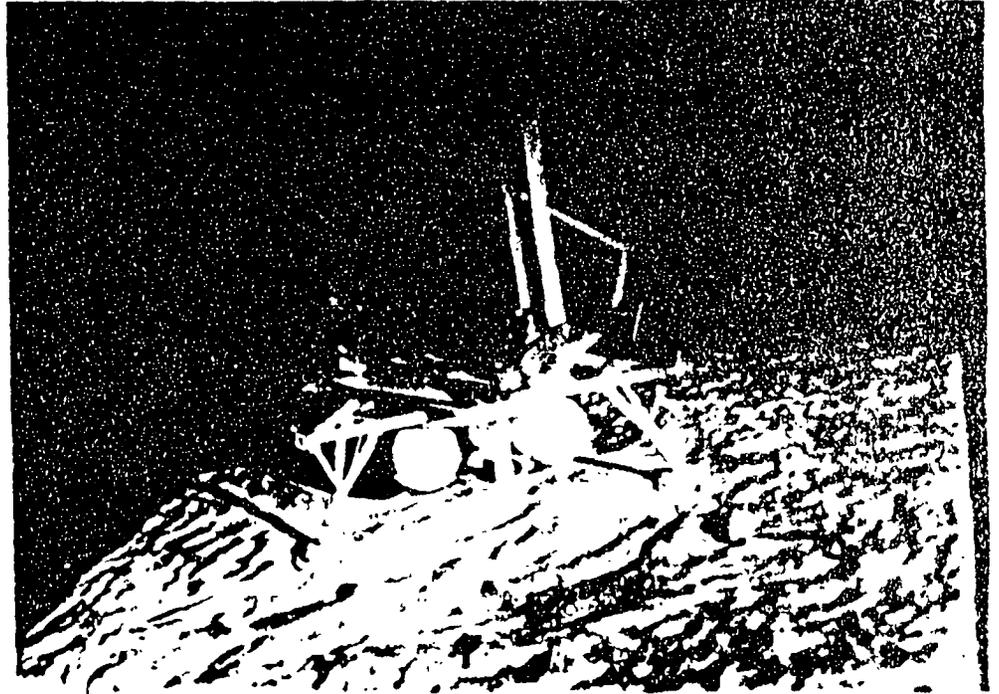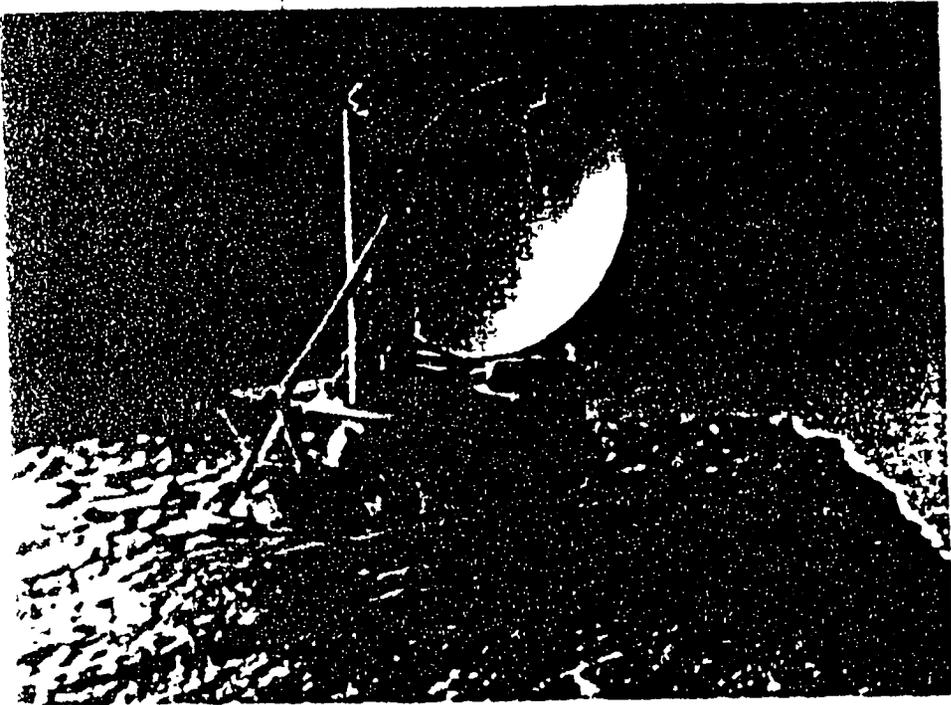
A GENERAL-PURPOSE INTELLIGENT VEHICLE SYSTEM AS LUNAR RESOURCE PROSPECTOR

NASA/UNIVERSITY OF ARIZONA SPACE ENGINEERING RESEARCH CENTER FOR UTILIZATION OF LOCAL PLANETARY RESOURCES
(Contact: Schooley, 602-621-2352)

NATIONAL INSTITUTE OF STANDARDS AND TECHNOLOGY UNITED STATES DEPARTMENT OF COMMERCE

PRESENTATION ON APRIL 29-30, 1992 TO:
Lunar Rover/Mobility Workshop
*Lunar and Planetary Institute, Houston, Texas*

Goal:

To develop an Intelligent Robotic Vehicle System (IRVS) based on the NIST SPIDER Robot for general-purpose services of Lunar and/or Martian missions.

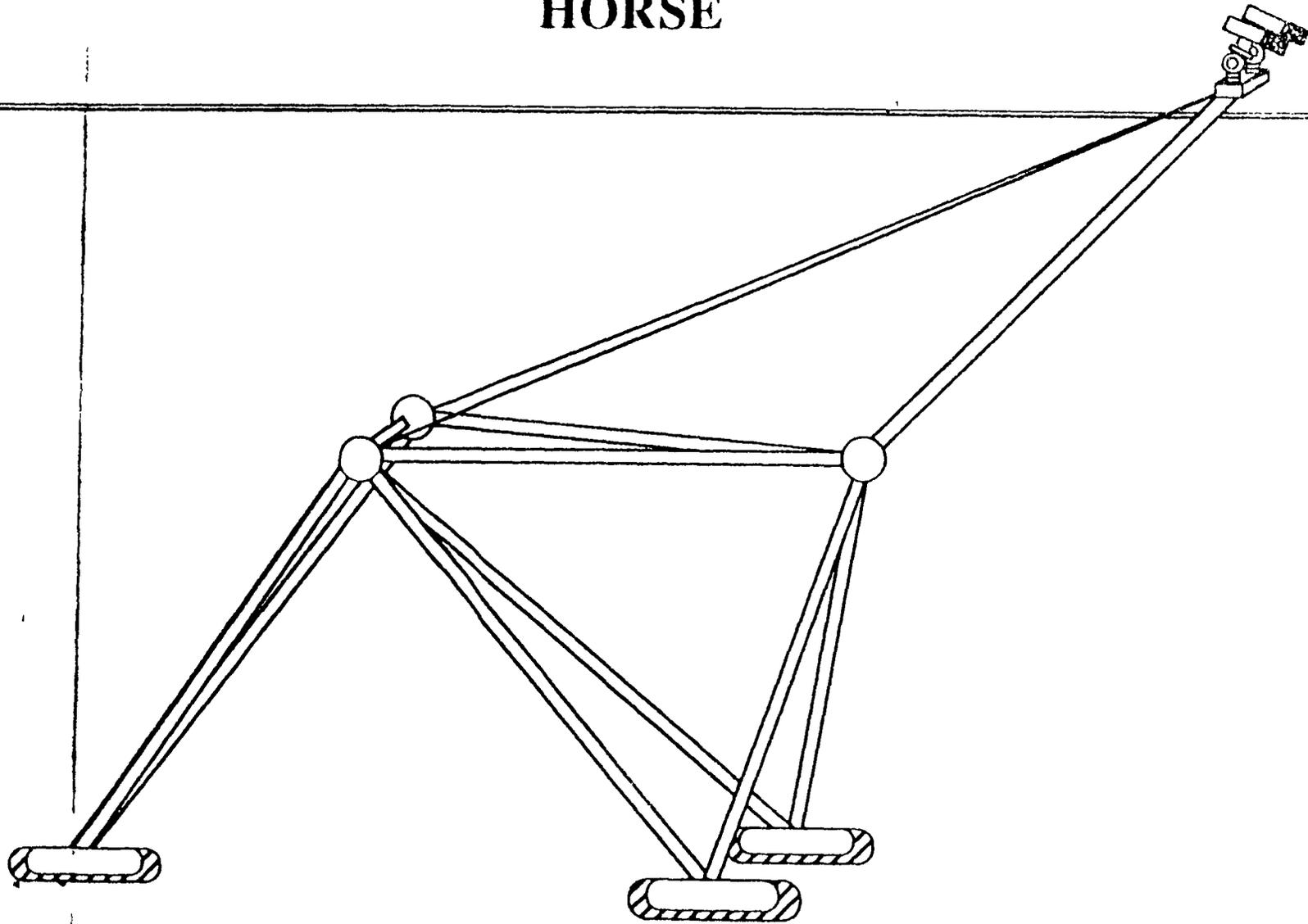## Advantages of the SPIDER Structure:

- A powerful platform with large working volume to accommodate the execution of diversified tasks

- Exceptionally high force to mass ratio

- Compact packaging for transport

- Mobility, dexterity, versatility, flexibility of scale, and ease of reconfiguration

SPIDER
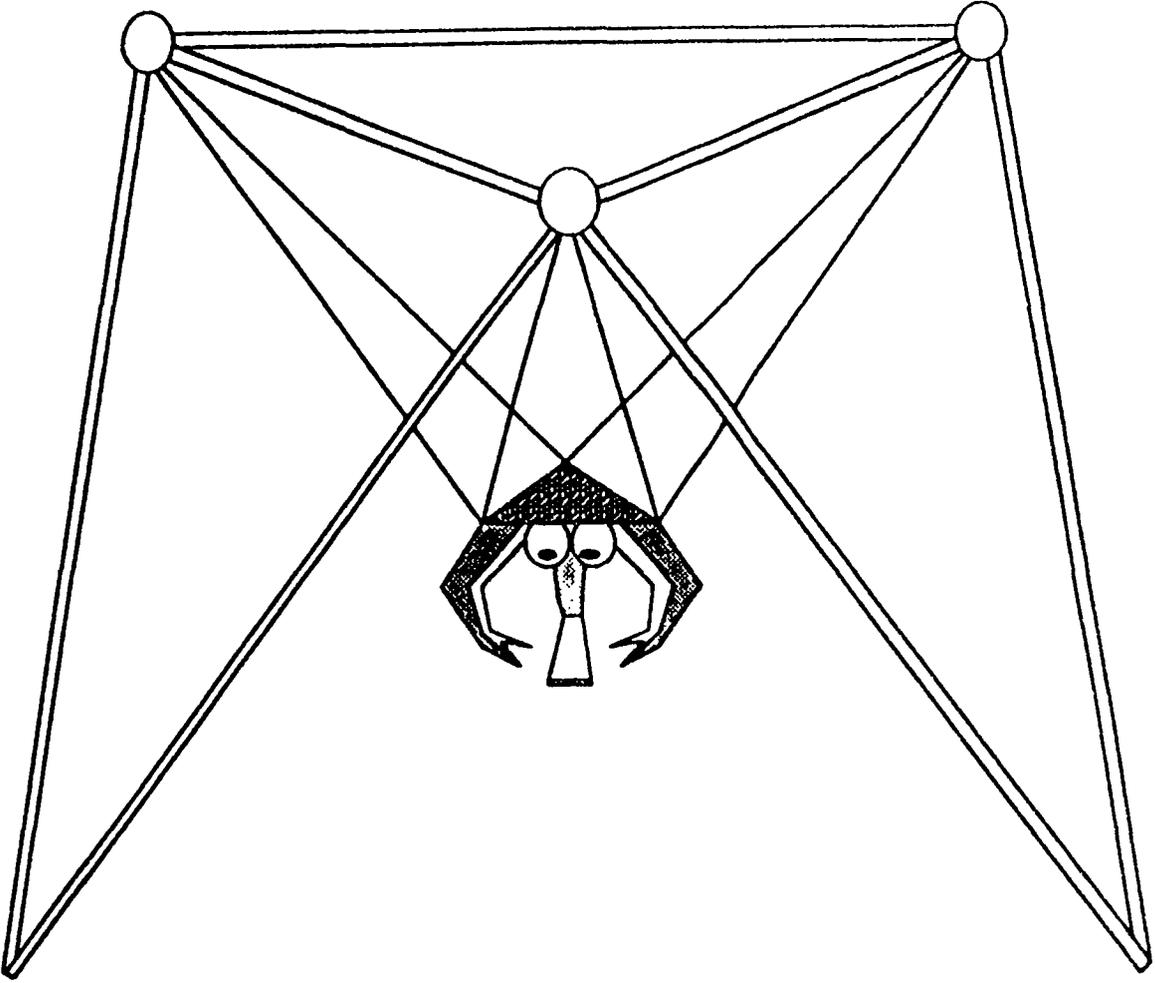
HORSE

SUMMARY

SPIDER AND HORSE represent a new approach to manipulation and locomotion

Structure is lightest and strongest possible

Configuration is ideal for mobility, soil sampling, & instrument deployment

Concept is simple and control technology is ready

Phase I IRVS Task

I Geophysical Instrument Deployment

- **Collect visual information**
- **Navigate to the inspection site**
- **Arm deploys instrument (e.g., seismometer) on the ground**

II Sample Collection and Analysis

- **Arm attaches a sample collection tool (e.g., bucket)**
- **Arm collects soil sample**
- **Arm places the sample into the assay instrument (e.g., gamma ray spectrometer)**

III Site Evaluation

- **Statistical analysis**
- **Information fusion**
- **Mineralogical site classification database and geological map generation**

INTEGRATED OXYGEN PRODUCTION PLANT

NASA/UNIVERSITY OF ARIZONA SPACE ENGINEERING RESEARCH CENTER FOR UTILIZATION OF LOCAL PLANETARY RESOURCES
(Contact: Ramohalli, 602-322-2304)

NATIONAL INSTITUTE OF STANDARDS AND TECHNOLOGY
UNITED STATES DEPARTMENT OF COMMERCE

PRESENTATION ON APRIL 29-30, 1992 TO:
Lunar Rover/Mobility Workshop
Lunar and Planetary Institute, Houston, Texas

INTEGRATED OXYGEN PRODUCTION PLANT

## A mobile Stewart platform carrying an on-board ISMU plant with associated tools and instruments.

TASKS:
- *Prospecting and Resource Assessment*
- *Mining and Beneficiation*
- *Plant Services and Oxygen Production*
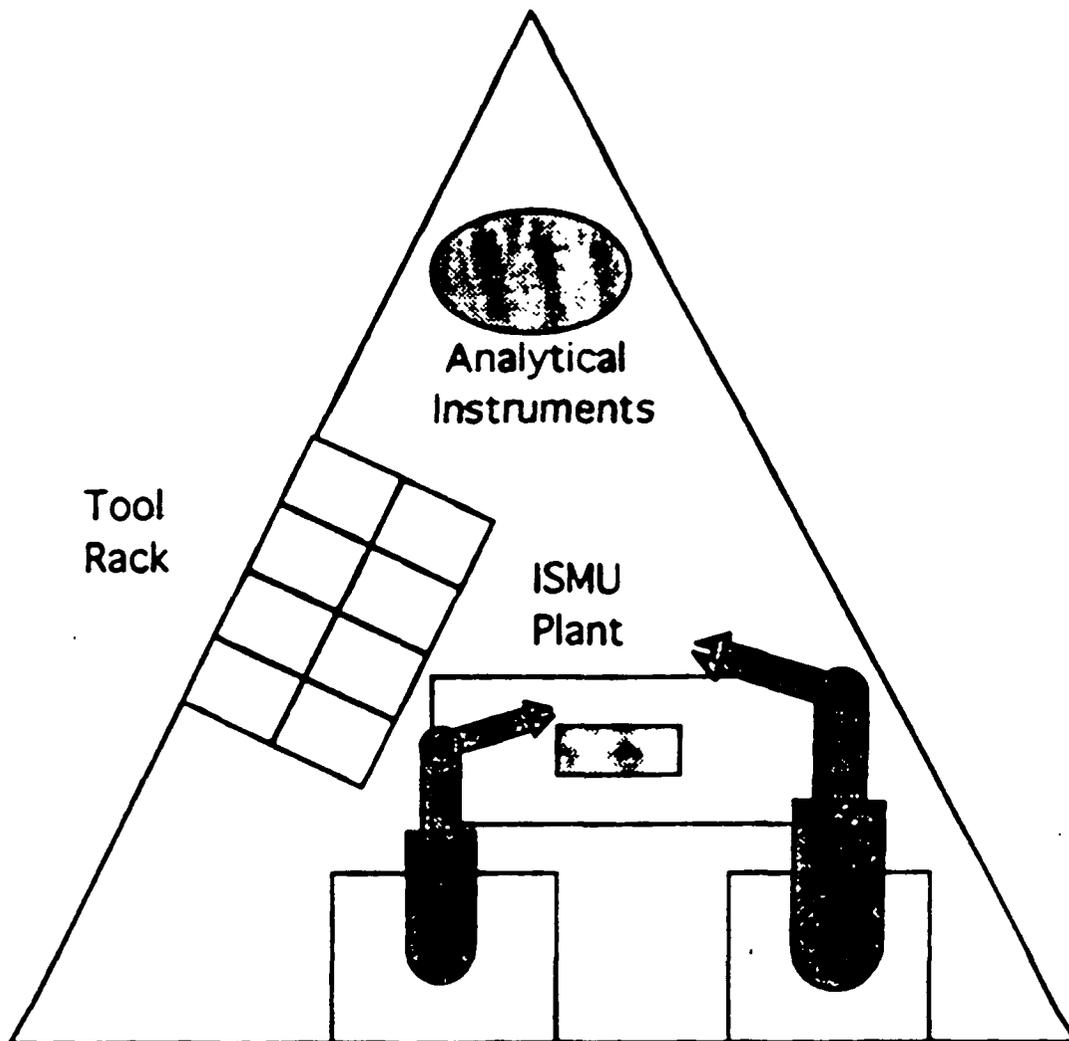- *Identification of Useful Byproducts*

EQUIPMENT:
- *Analytical Instruments*
- *Assorted Tools*
- *Beneficiation Machines*
- *Production Plant*
- *Vehicle Infrastructure*

Phase II : Mining and Beneficiation

Analytical Instrument

Tool Rack

Beneficiation
Machines

Arm

Excavation
Bucket
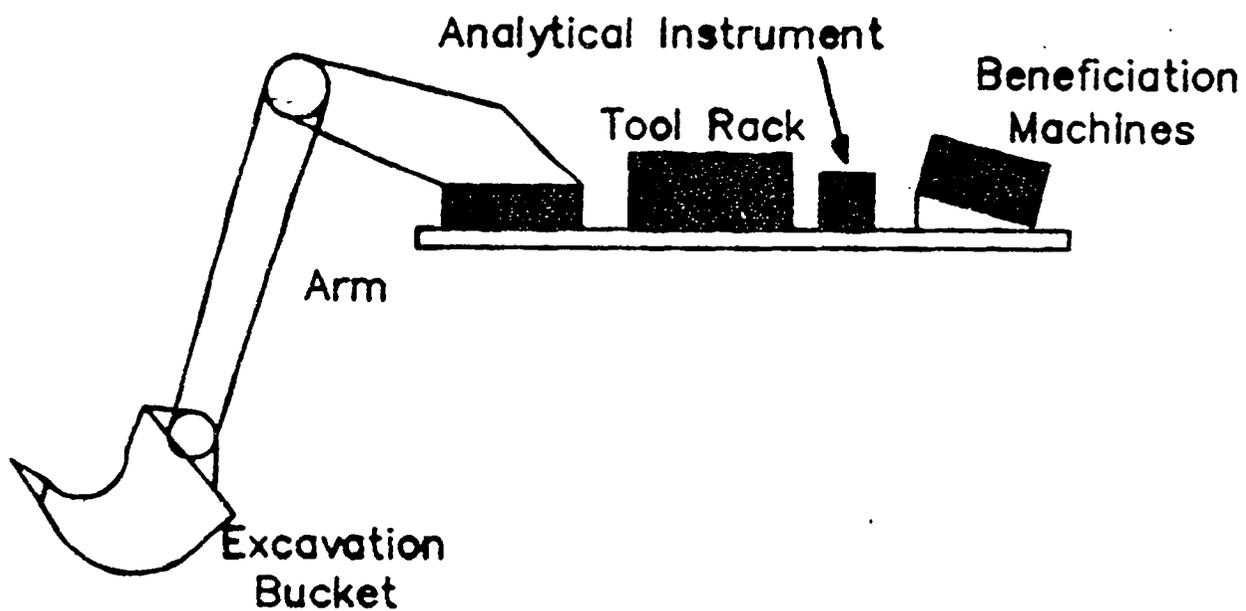
Side View of NIST Task Platform

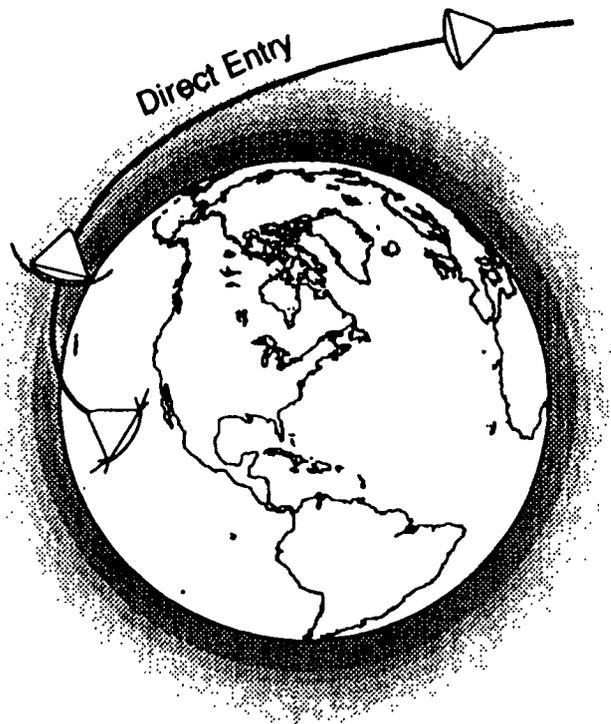# Phase III: General Services to ISMU Plants

Top View of the NIST Work Platform

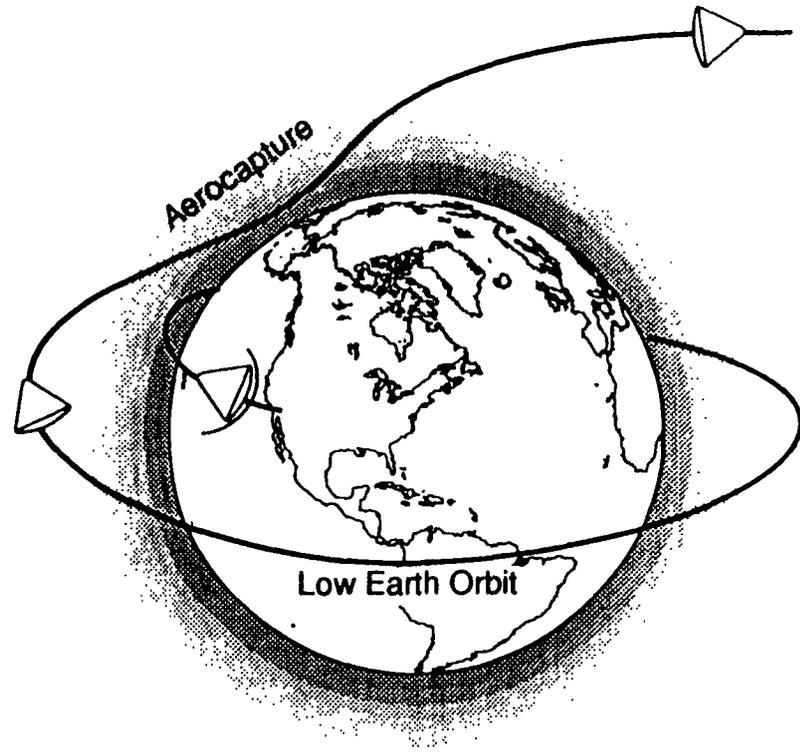# FIRST LUNAR OUTPOST CREW MODULE THERMAL PROTECTION DESIGN SENSITIVITY

Stan Bouslog, Bill Rochelle, Stan Williams
Lockheed Engineering & Sciences Co., Houston, TX

Joe Caram, Don Curry, Steve Derry, Matt Ondler
NASA Johnson Space Center

Direct Entry

Aerocapture

Low Earth Orbit

**DIRECT-ENTRY
LUNAR-RETURN MISSION**

**AEROCAPTURE AND LEO ENTRY
LUNAR-RETURN MISSION**

SEI 3rd TECHNICAL INTERCHANGE MEETING
MAY 5-6, 1992
HOUSTON, TEXAS

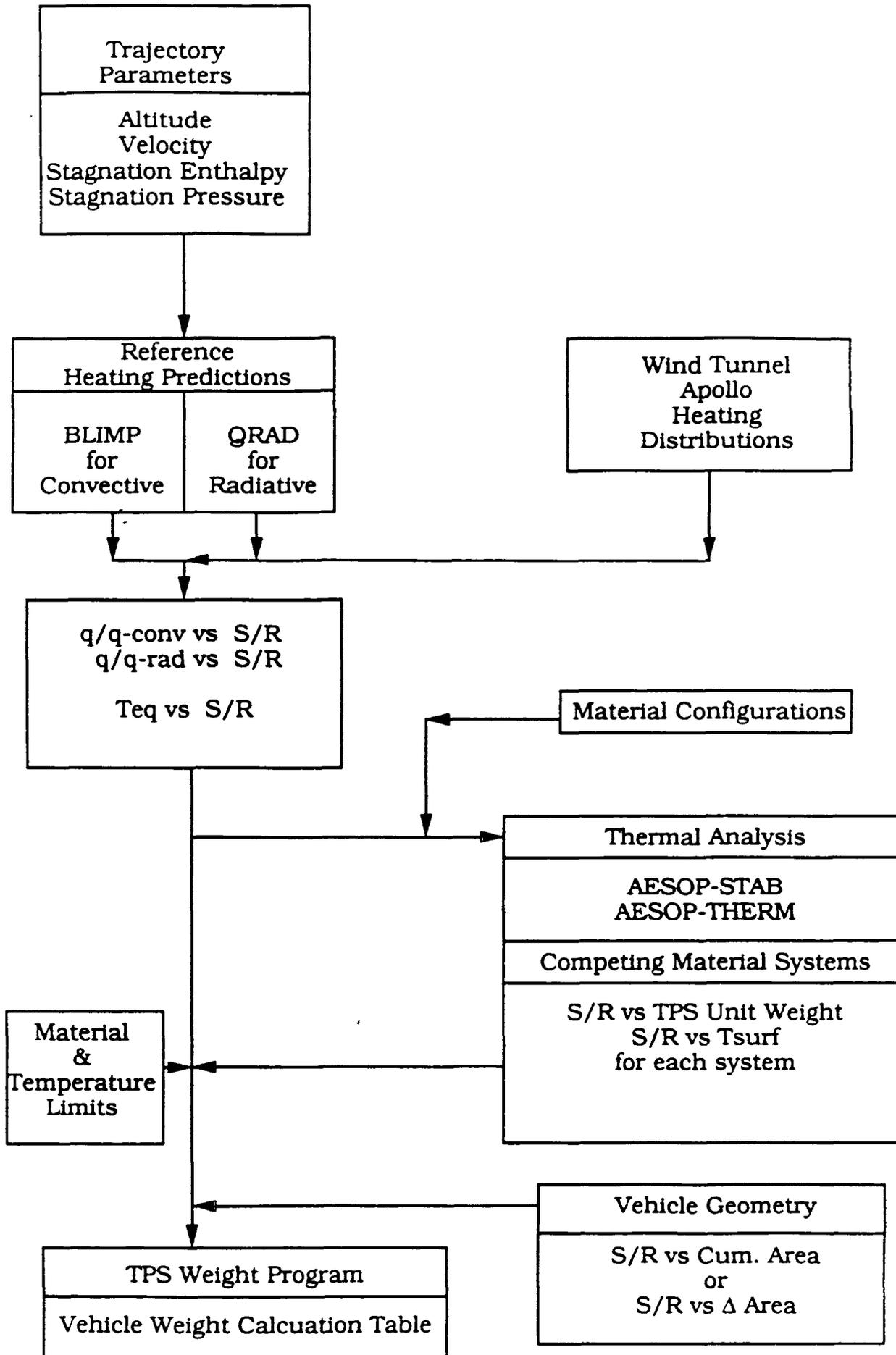# FIRST LUNAR OUTPOST (FLO) CREW MODULE THERMAL PROTECTION DESIGN SENSITIVITY

STUDY OBJECTIVES

- ASSESS EFFECTS OF VEHICLE SIZE ON AEROTHERMODYNAMIC ENVIRONMENT AND THERMAL PROTECTION SYSTEM (TPS) WEIGHT

- ASSESS EFFECTS OF LUNAR-RETURN STRATEGIES ON AEROTHERMODYNAMIC ENVIRONMENT AND TPS WEIGHT

- ASSESS WEIGHT PENALTY FOR COMMON TPS DESIGN FOR ALL LUNAR-RETURN STRATEGIES

STUDY GUIDELINES

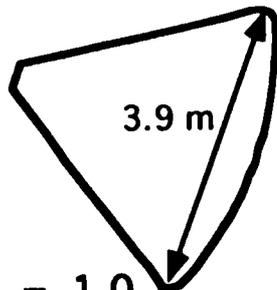- APOLLO COMMAND MODULE VEHICLE CONFIGURATION FOR SCALES OF 1.0, 1.5, 2.5

- LUNAR-RETURN STRATEGIES
  - DIRECT-ENTRY (MODIFIED APOLLO GUIDANCE)
  - AERO-CAPTURE (AFE GUIDANCE) AND LOW-EARTH ORBIT ENTRY (SHUTTLE GUIDANCE)

- AEROTHERMODYNAMICS
  - APOLLO RADIATIVE AND CONVECTIVE (LAMINAR) HEATING DISTRIBUTIONS

- TPS MATERIALS
  - FLIGHT CERTIFIED MATERIALS
  - MODIFIED APOLLO TPS SIZING METHOD

Sketch of TPS Weight Calculation Process

```
┌─────────────────────────┐
│   Trajectory            │
│   Parameters            │
├─────────────────────────┤
│   Altitude              │
│   Velocity              │
│   Stagnation Enthalpy   │
│   Stagnation Pressure   │
└─────────────────────────┘
           │
           ▼
┌─────────────────────────┐         ┌──────────────────┐
│     Reference           │         │   Wind Tunnel    │
│  Heating Predictions    │         │   Apollo         │
├────────────┬────────────┤         │   Heating        │
│  BLIMP     │  QRAD      │         │   Distributions  │
│  for       │  for       │         └──────────────────┘
│  Convective│  Radiative │
└────────────┴────────────┘
```

Reference Heating Predictions

| BLIMP for Convective | QRAD for Radiative |

Wind Tunnel Apollo Heating Distributions

q/q-conv vs S/R
q/q-rad vs S/R

Teq vs S/R

Material Configurations

Thermal Analysis

AESOP-STAB
AESOP-THERM

Competing Material Systems

S/R vs TPS Unit Weight
S/R vs Tsurf
for each system

Material
&
Temperature
Limits

Vehicle Geometry

S/R vs Cum. Area
or
S/R vs Δ Area

TPS Weight Program

Vehicle Weight Calcuation Table

VEHICLE AND TRAJECTORY DESIGN ASSUMPTIONS

SCALE = 1.0
3.9 m

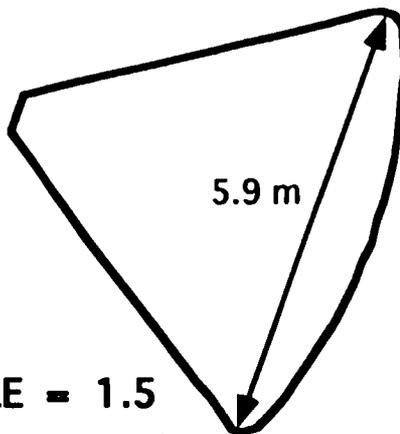WEIGHT = 6,466 kg

$W/C_D A$ = 432 kg/m^2

REF. AREA = 11.6 m^2
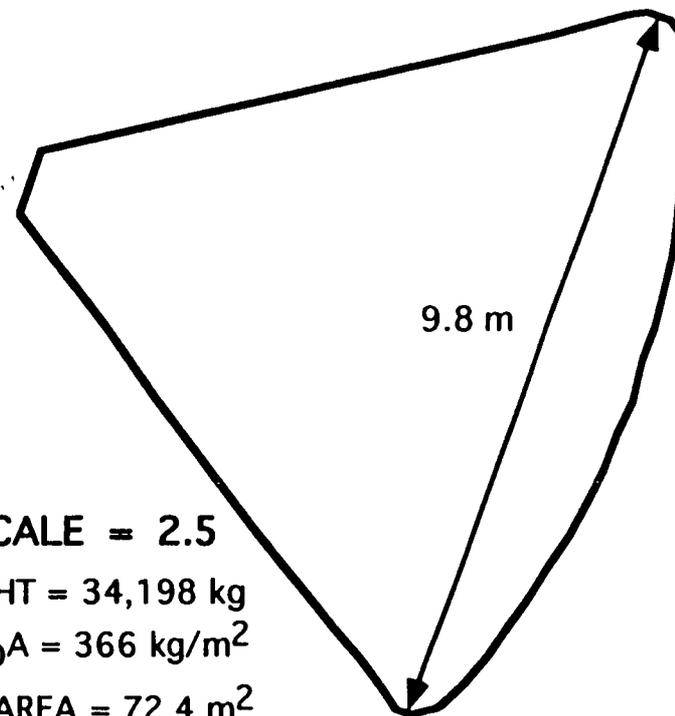
ORIGINAL APOLLO

WEIGHT = 5,668 kg

$W/C_D A$ = 379 kg/m^2

SCALE = 1.5
5.9 m

WEIGHT = 11,724 kg

$W/C_D A$ = 348 kg/m^2

REF. AREA = 26.0 m^2

SCALE = 2.5
9.8 m

WEIGHT = 34,198 kg

$W/C_D A$ = 366 kg/m^2

REF. AREA = 72.4 m^2

TRAJECTORY DESIGN

MODE OF ENTRY	INERTIAL VELOCITY (KM/SEC)	FLIGHT PATH ANGLE (DEGS.)
DIRECT-ENTRY	11.0	-6.6
AERO-CAPTURE	11.0	-5.2
LOW-EARTH ORBIT ENTRY	7.9	-1.5
APOLLO 4 (Alpha=24°)	11.2	-6.95

AERODYNAMIC CRITERIA

ANGLE-OF-ATTACK = 20 DEGS.

C_D = 1.2906

C_L = 0.3867
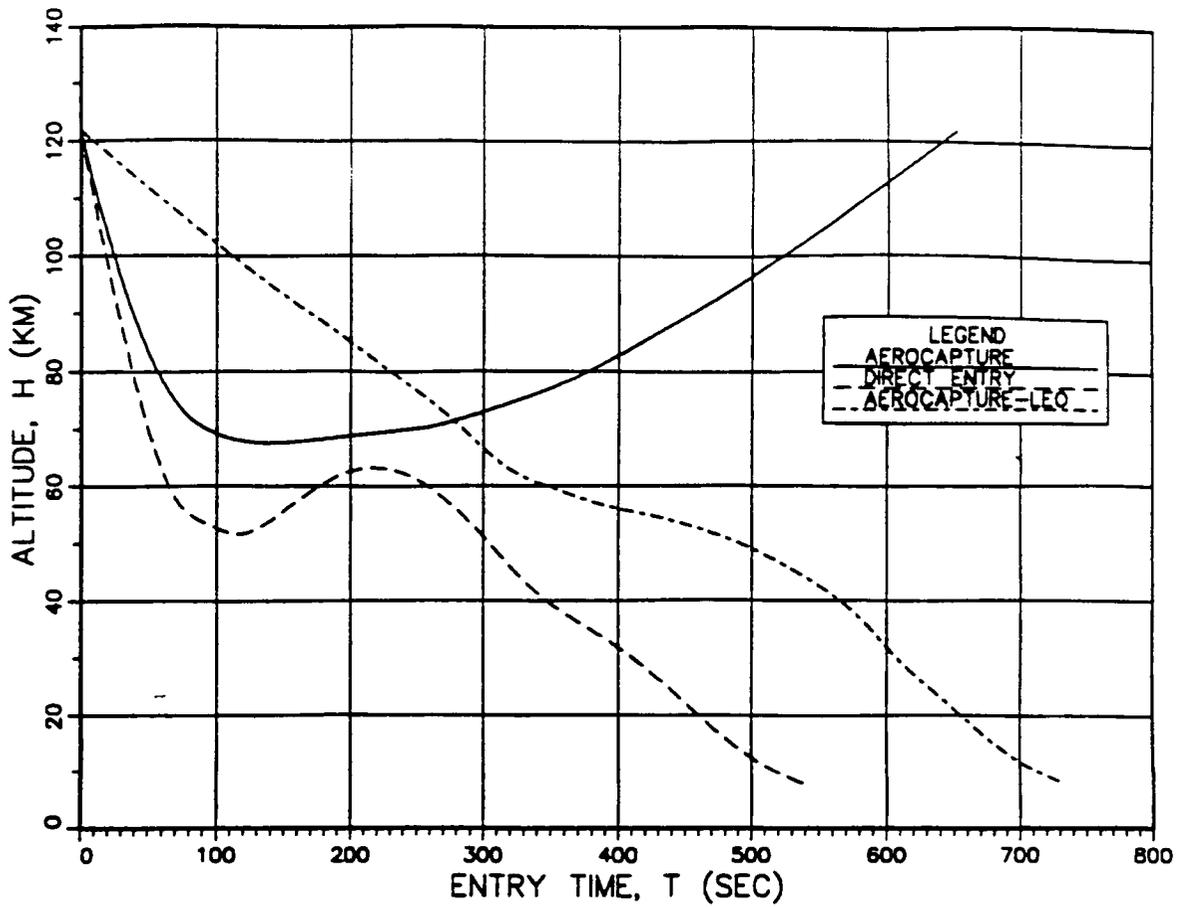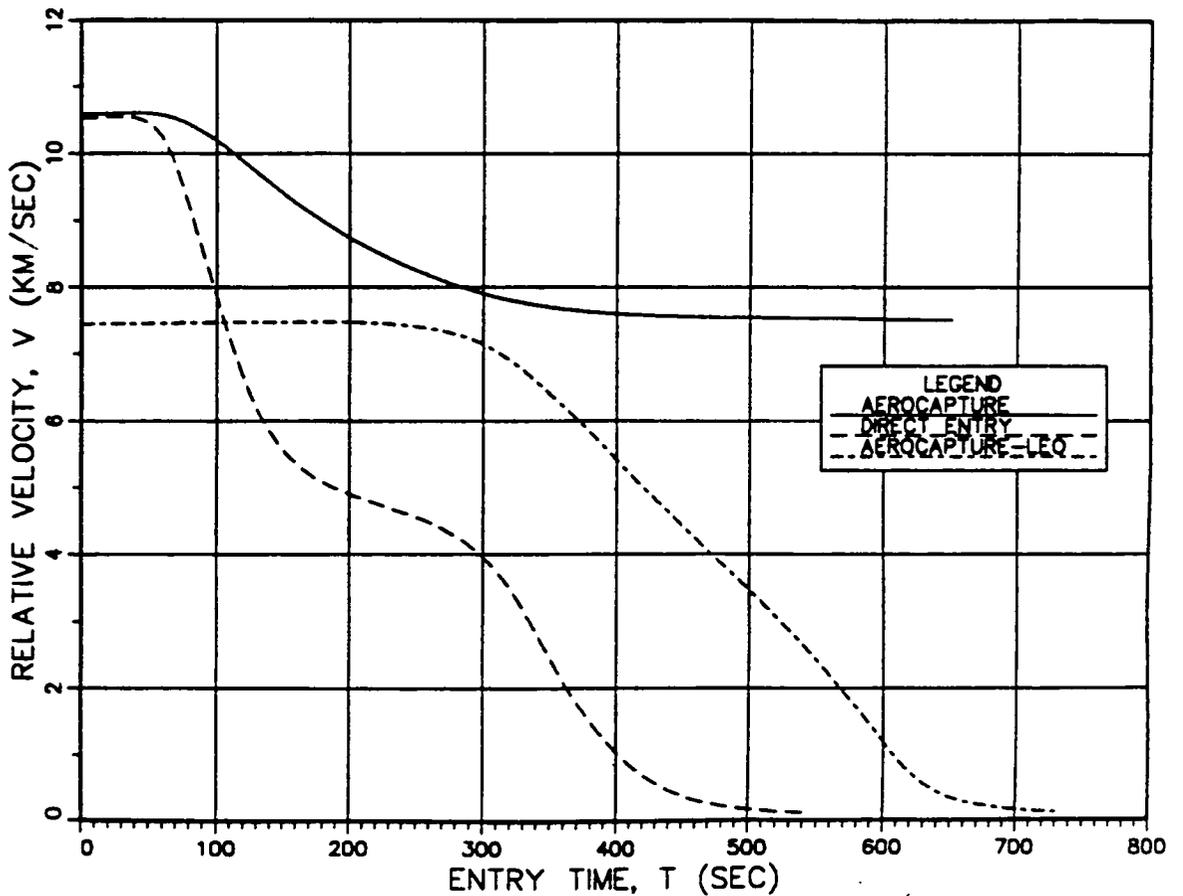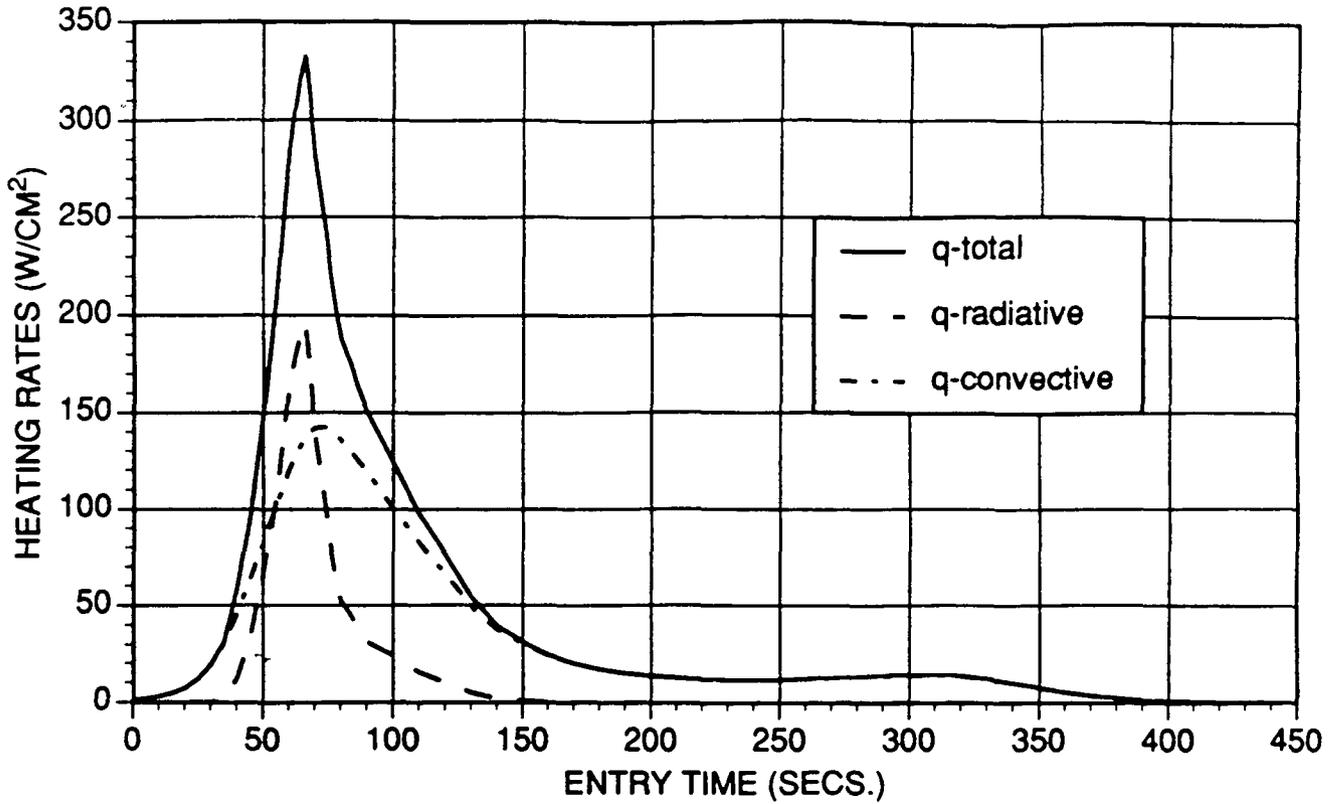
L/D = 0.3

ALTITUDE AS A FUNCTION OF TIME FOR 1.5-SCALE APOLLO VEHICLE
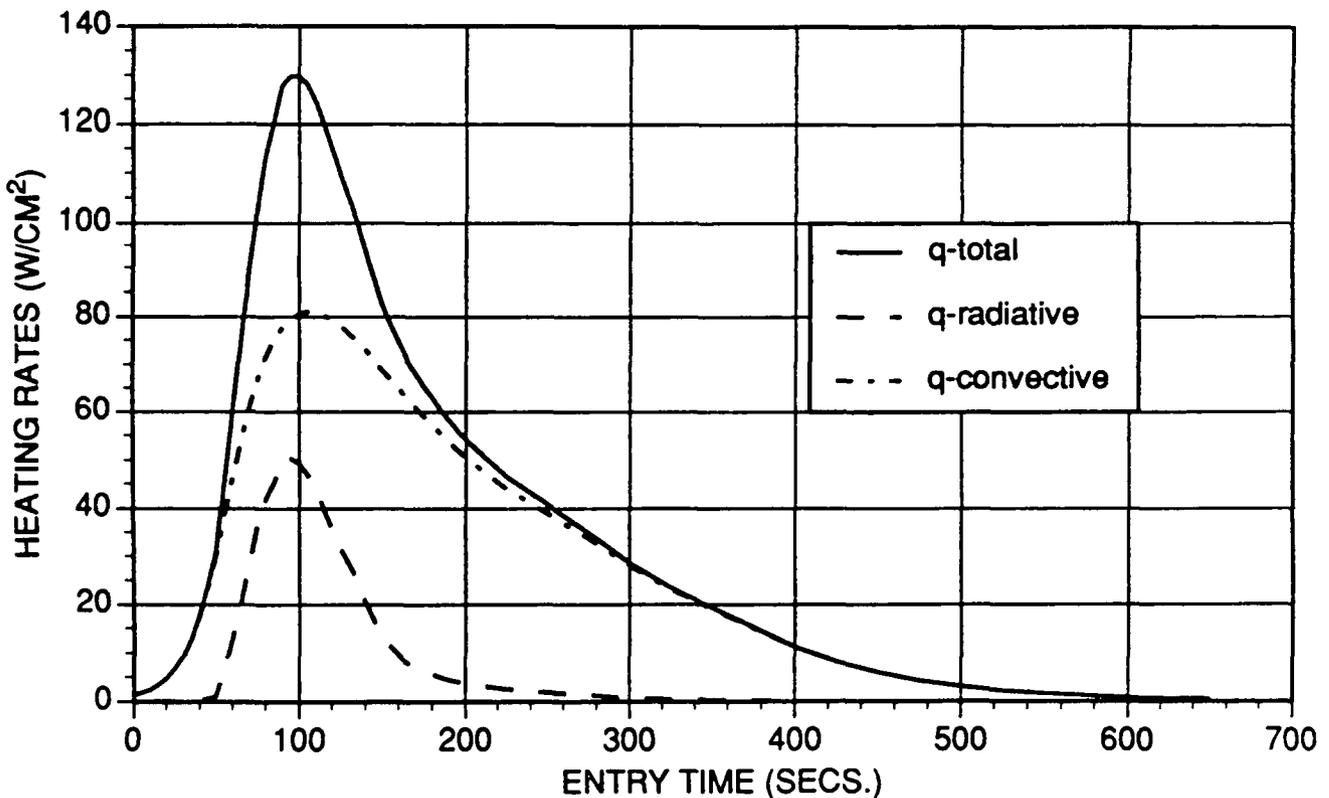FOR AEROCAPTURE, DIRECT ENTRY, AND A/C-LEO ENTRY LUNAR MISSIONS

VELOCITY AS A FUNCTION OF TIME FOR 1.5-SCALE APOLLO VEHICLE
FOR AEROCAPTURE, DIRECT ENTRY, AND A/C-LEO ENTRY LUNAR MISSIONS

55

PEAK HEATING RATES TO 1.5 SCALE APOLLO VEHICLE

DIRECT-ENTRY TRAJECTORY

AERO-CAPTURE TRAJECTORY

560

PEAK HEATING RATES TO SCALE APOLLO VEHICLES
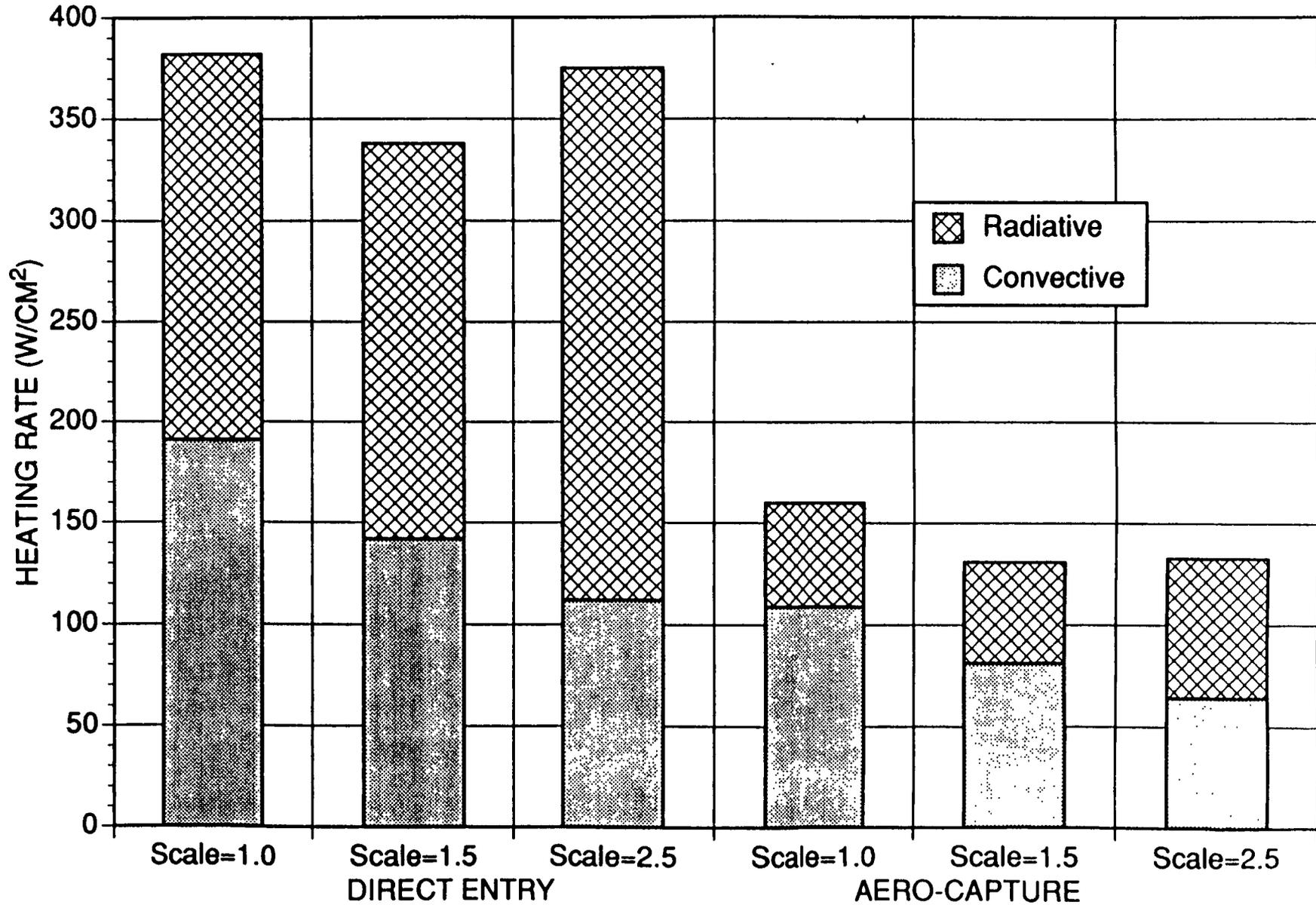
HEAT LOADS FOR SCALE APOLLO VEHICLE AT PEAK HEATING LOCATION

PEAK HEATING RATES AS A FUNCTION OF VEHICLE DIAMETER
FOR LUNAR RETURN DIRECT ENTRY

PEAK HEATING RATES AS A FUNCTION OF VEHICLE DIAMETER
FOR LUNAR RETURN – AEROCAPTURE
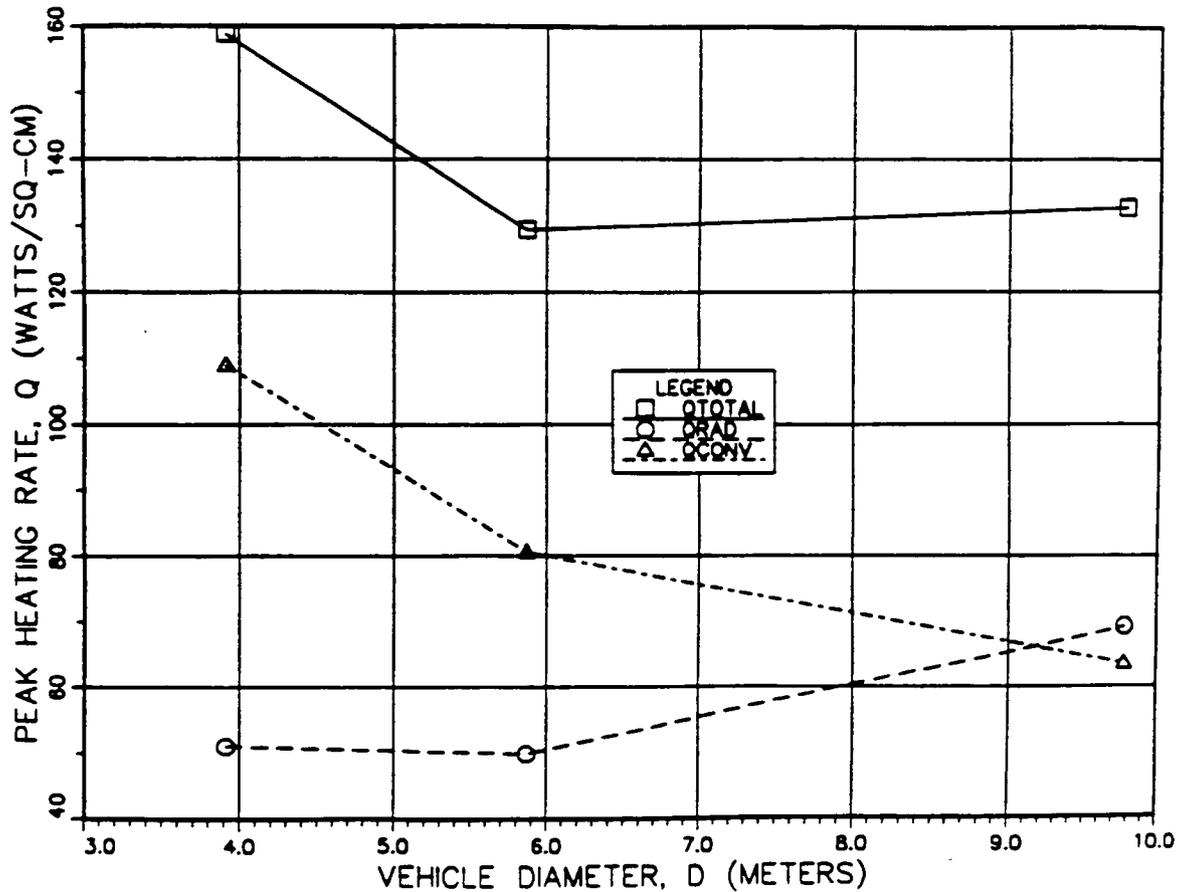
56

Thermal Protection System

Typical RSI Thermal Model

Thermal Model of the Ablator
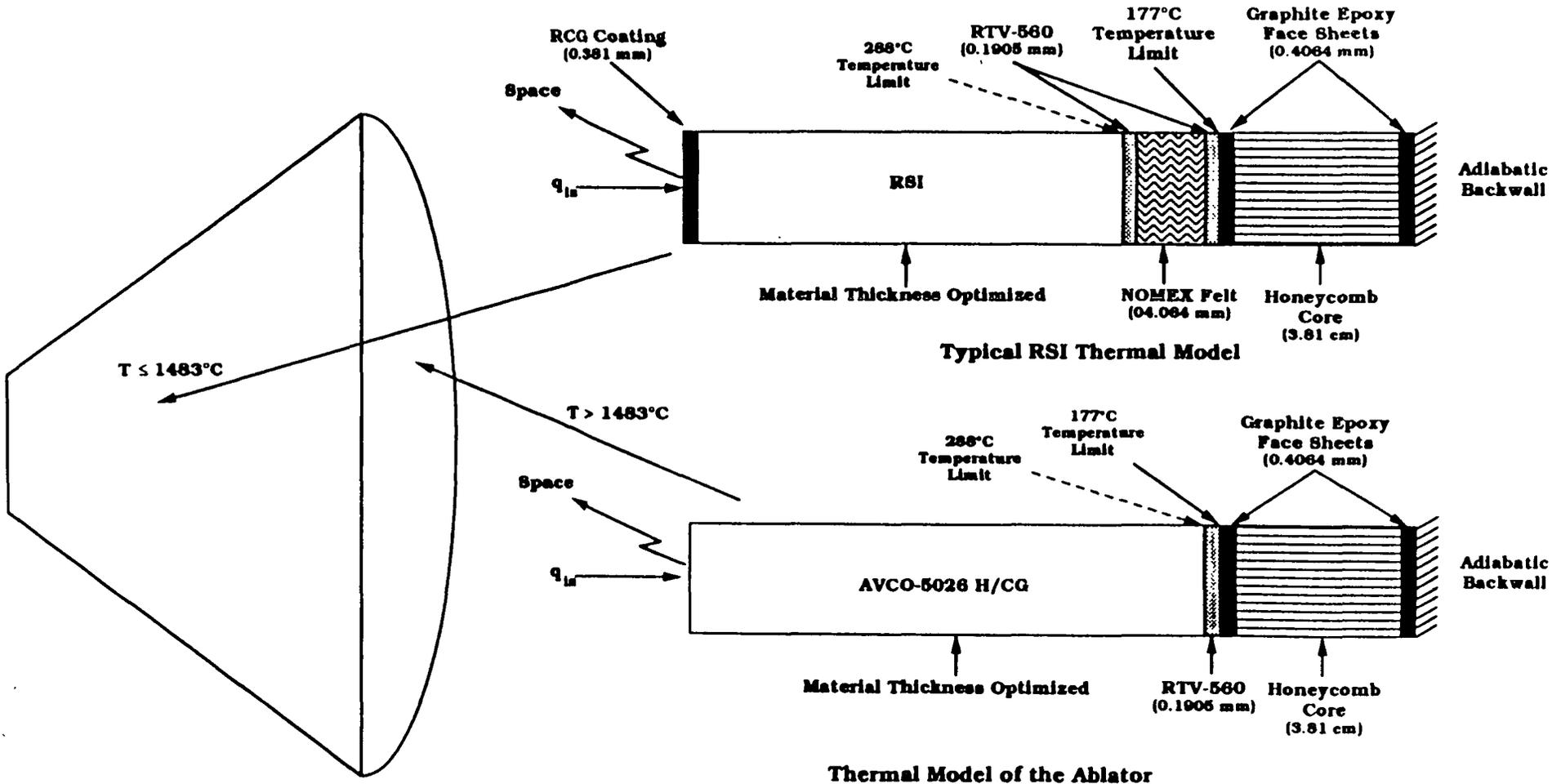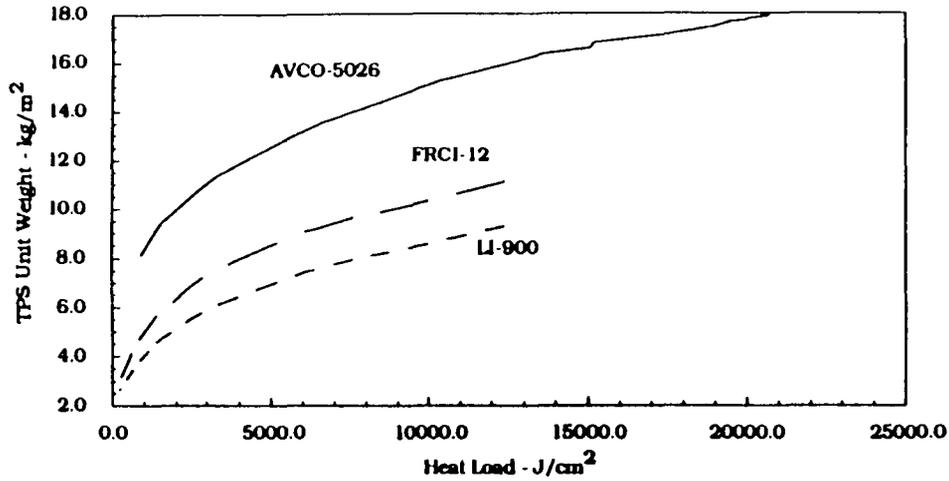
Comparison of the TPS Unit Weights vs. Heat Load
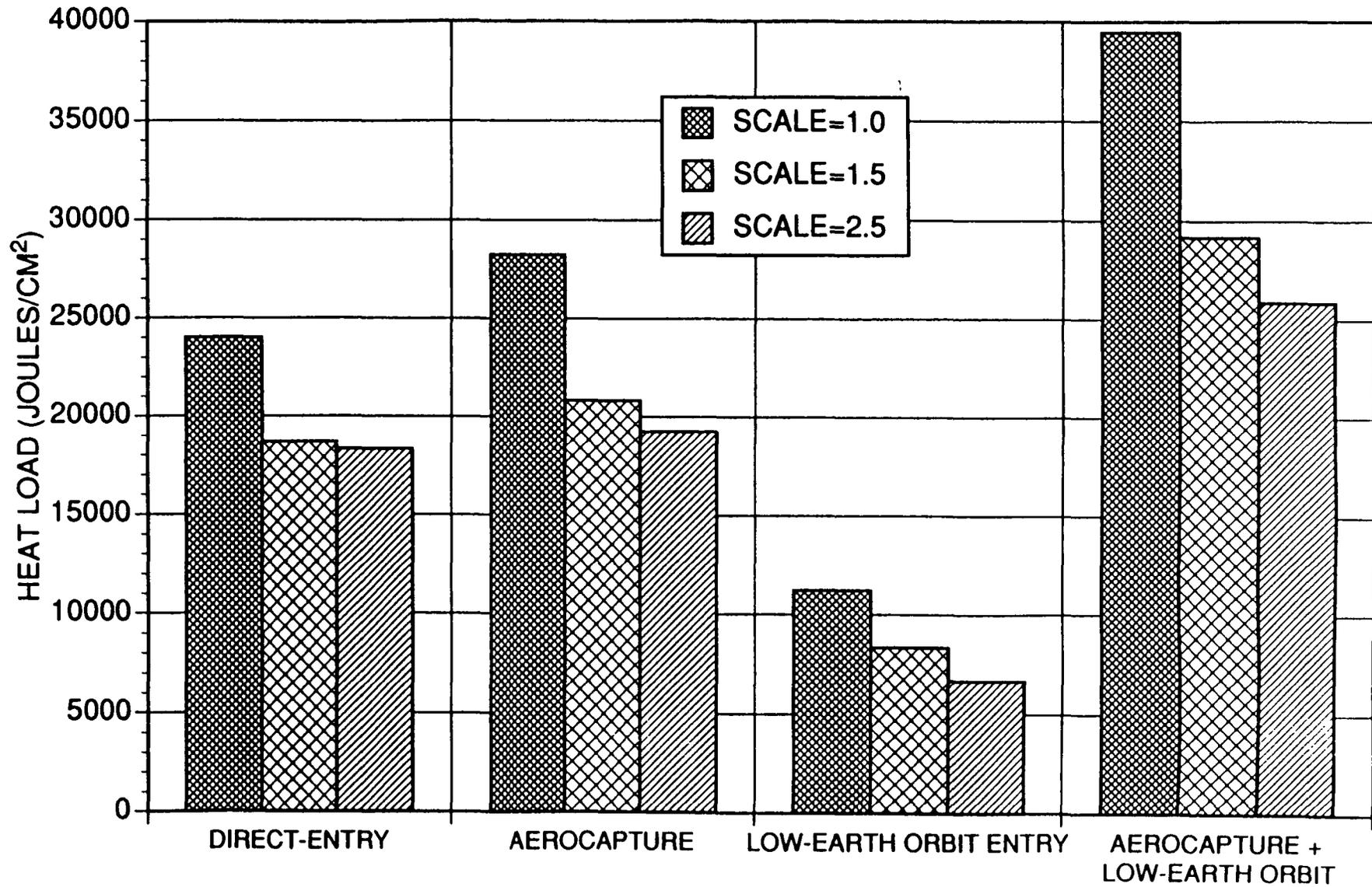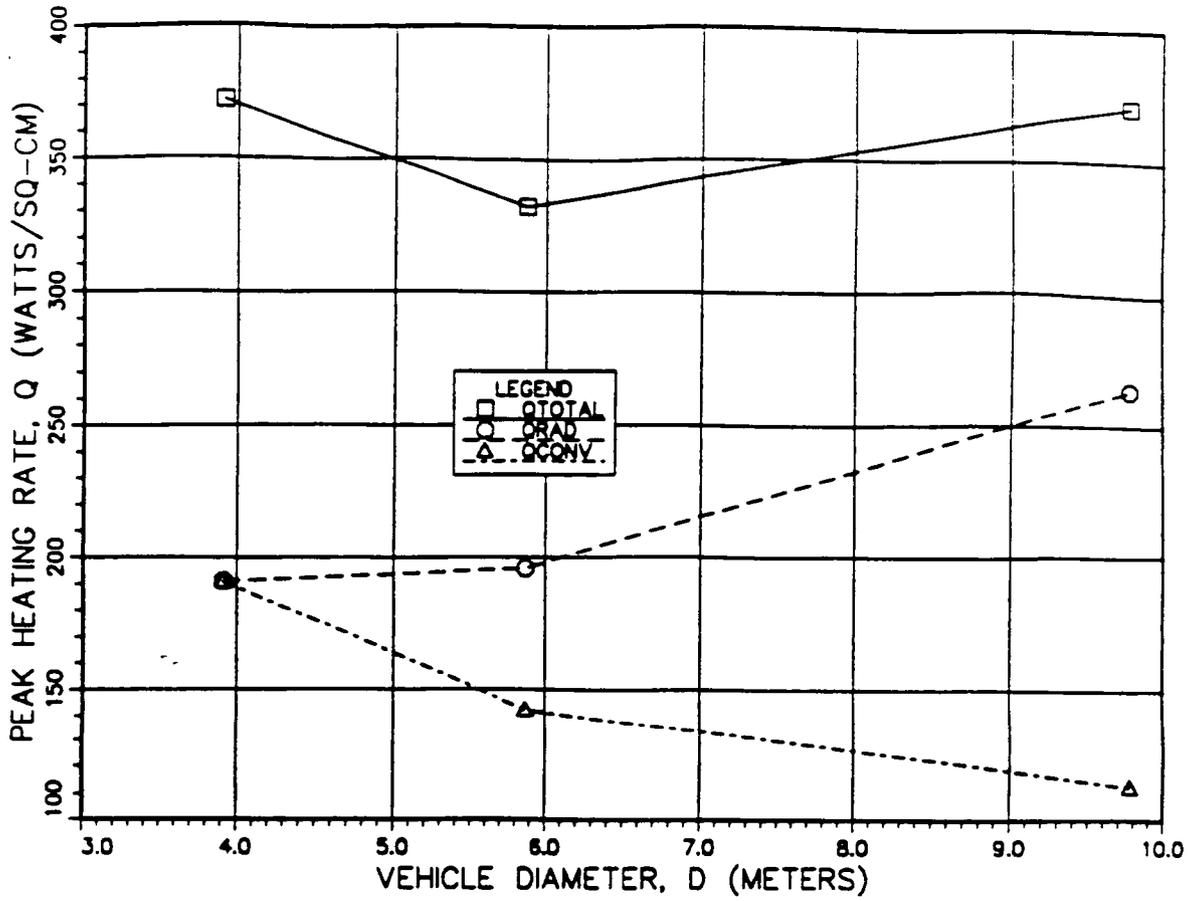for the 1.5 Scale Apollo CM for Direct Entry
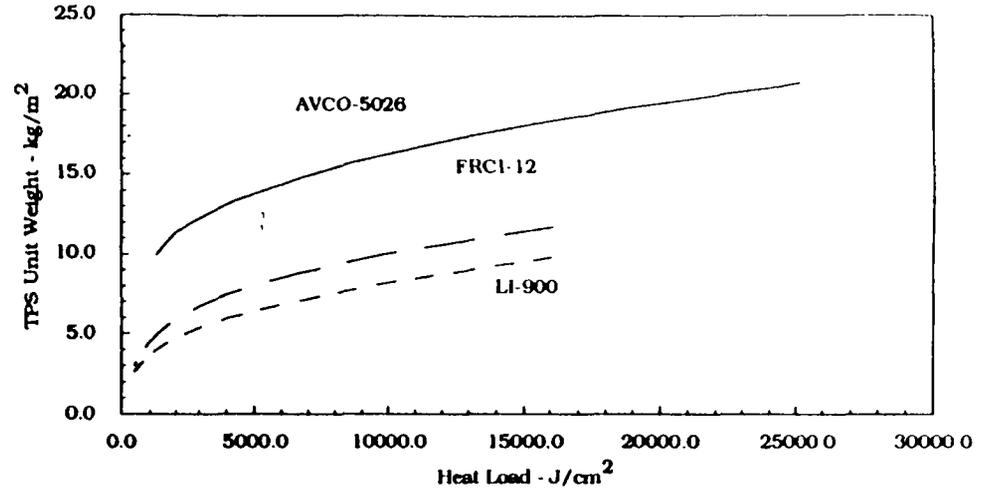
Comparison of the TPS Unit Weights vs Heat Load
for the 1.5 Scale Apollo CM for the Aerocapture
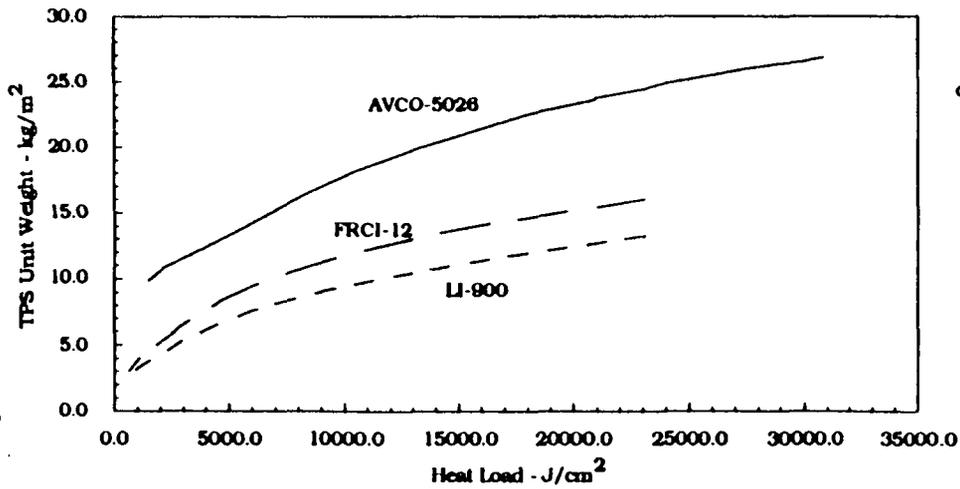
Comparison of the TPS Unit Weights vs. Heat Load
for the 1.5 Scale Apollo CM for the Aerocapture-1 Orbit-Entry
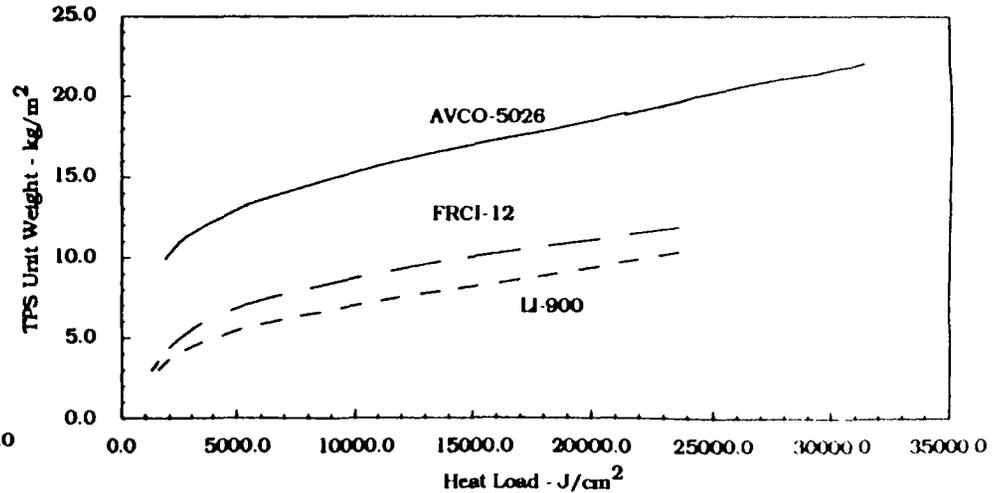
Comparison of the TPS Unit Weights vs. Heat Load
for the 1.5 Scale Apollo CM for the Aerocapture-5 Orbit-Entry
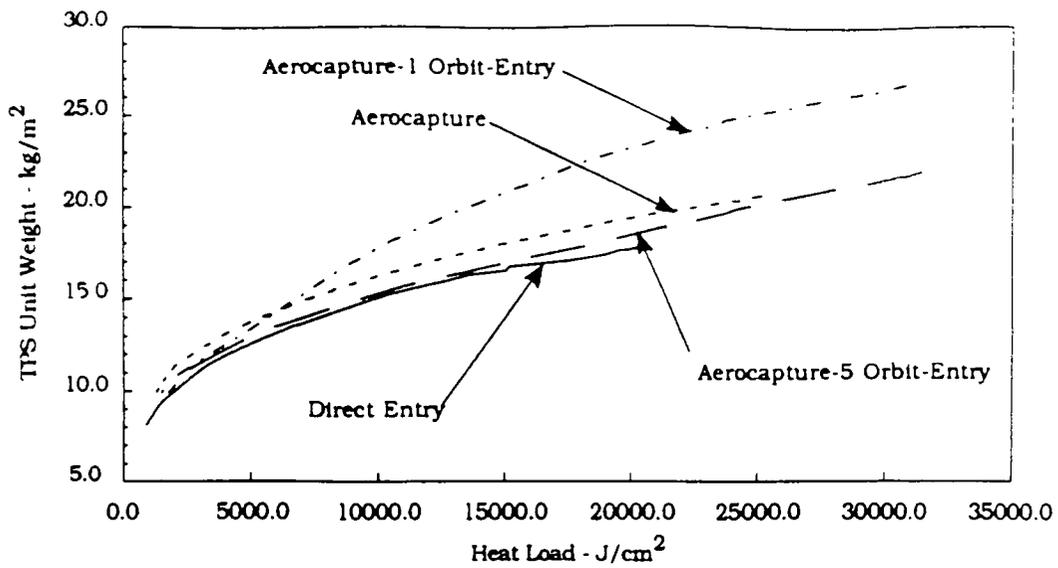
Comparison of the AVCO-5026 TPS Unit Weights vs. Heat Load for the 1.5 Scale Apollo CM
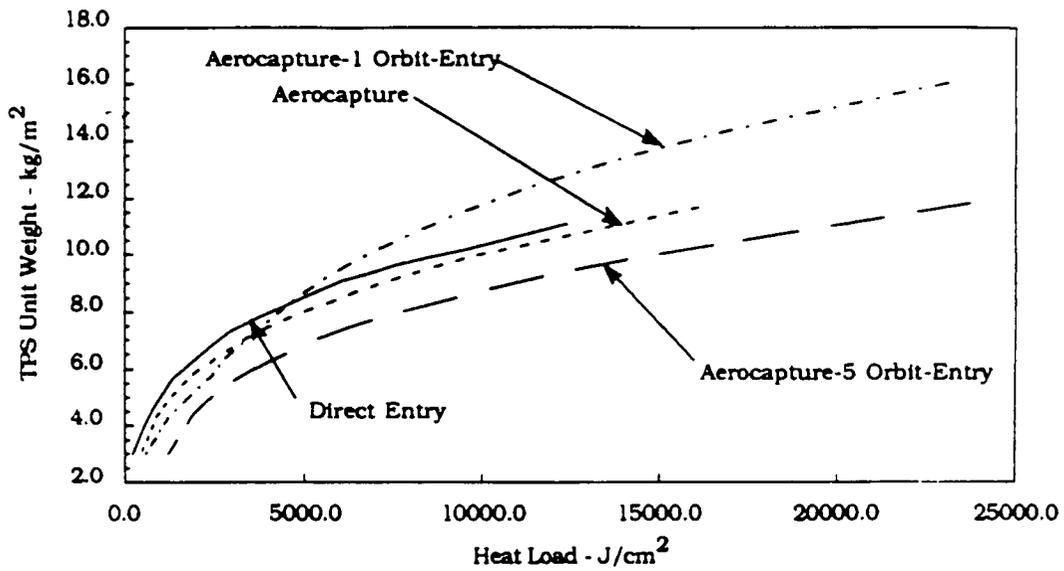
Comparison of the FRCI-12 TPS Unit Weights vs. Heat Load for the 1.5 Scale Apollo CM
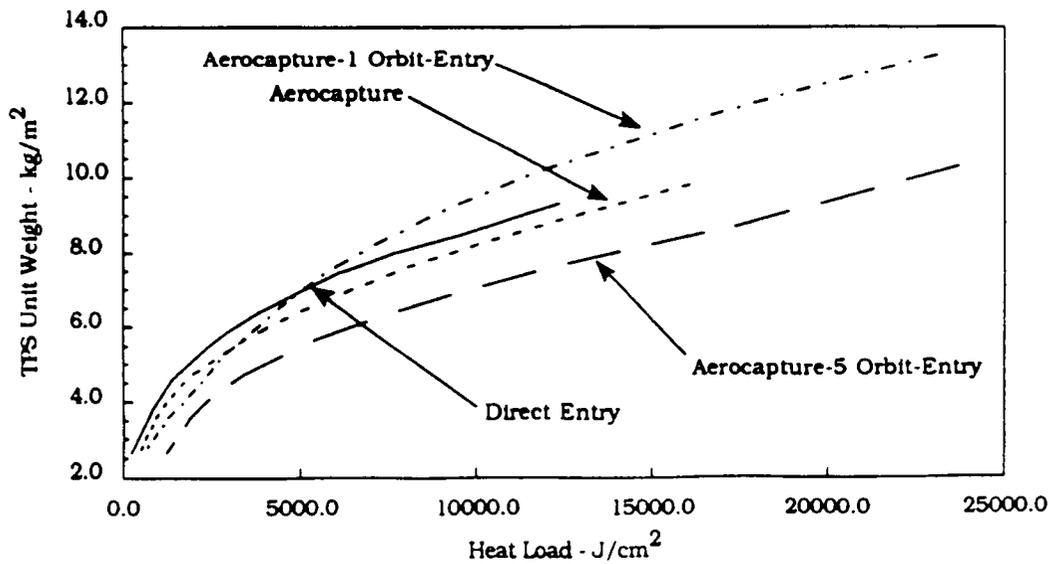
Comparison of the LI-900 TPS Unit Weights vs. Heat Load for the 1.5 Scale Apollo CM
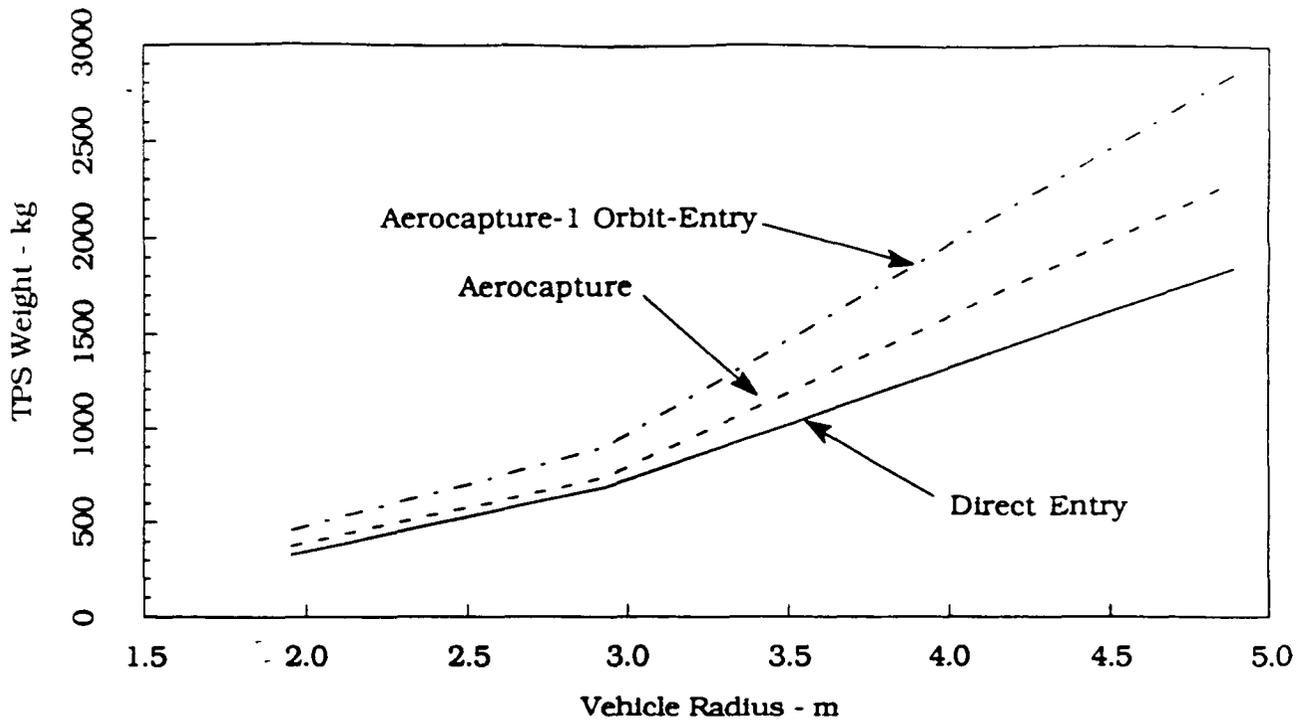
TPS Weight as a Function of the Vehicle Radius
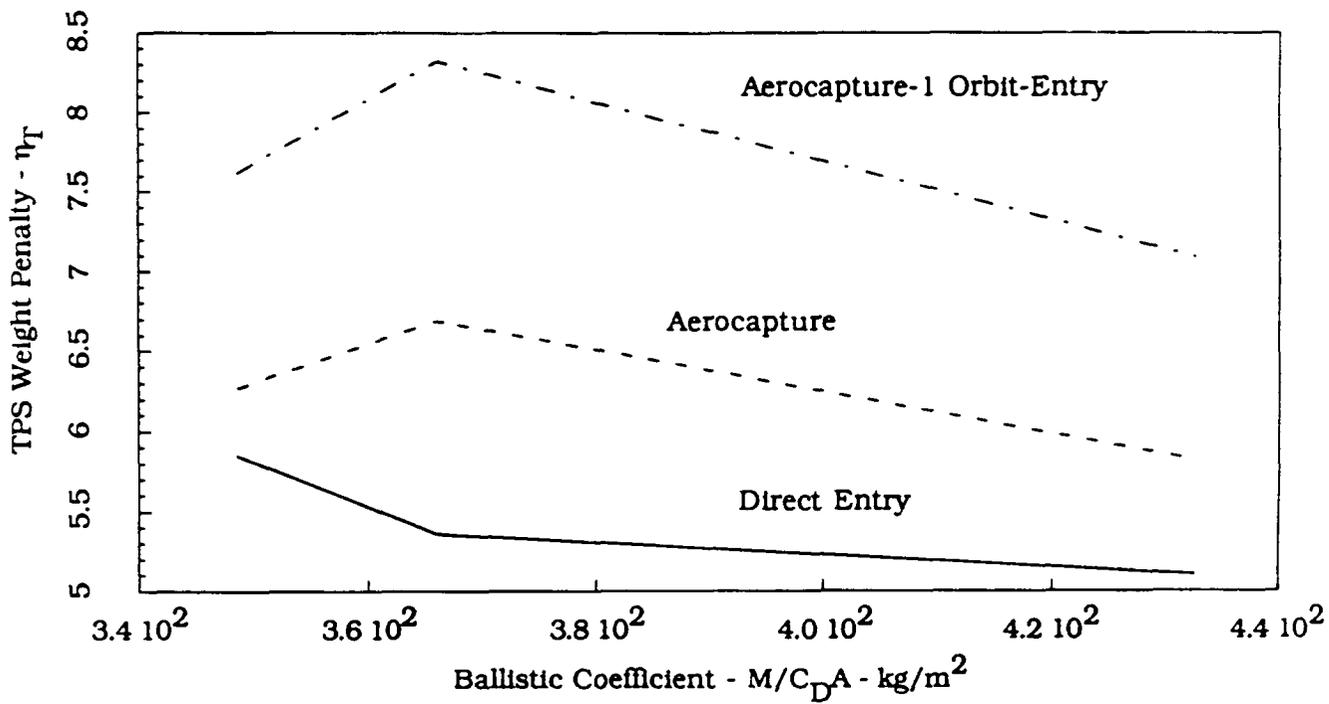
TPS Weight Penalty as a Function of Ballistic Coefficient
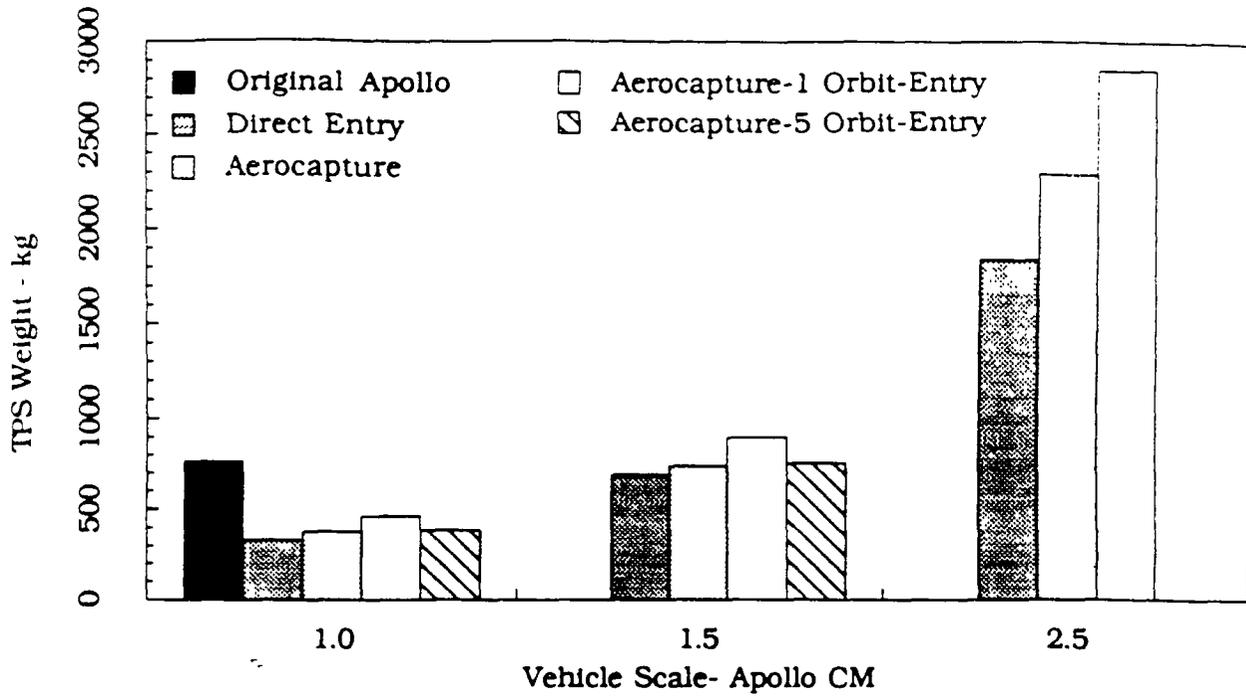
TPS Weight as a Function of Vehicle Size
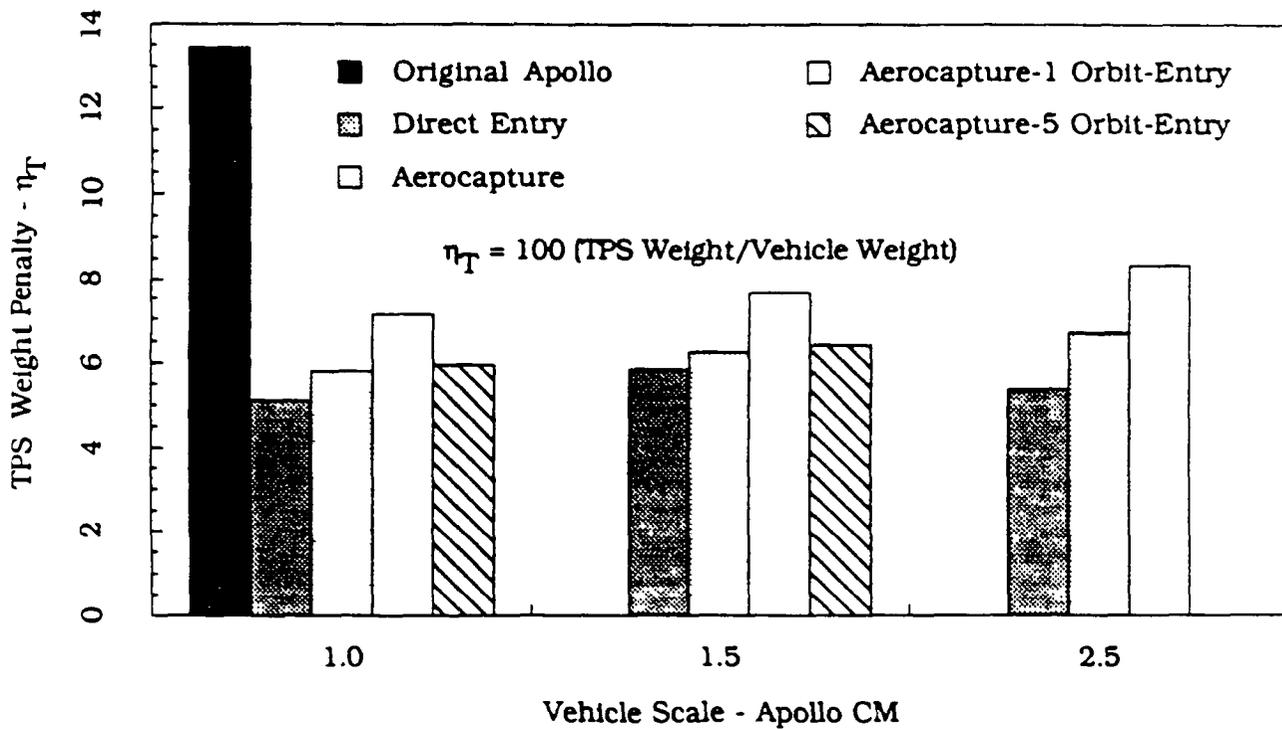
TPS Weight Penalty as a Function of Vehicle Size

CONCLUSIONS

- VEHICLE SIZE SENSITIVITY

  - PEAK HEATING RATES NOT STRONG FUNCTION OF VEHICLE SIZE BUT LARGER VEHICLES
    HAVE RELATIVELY LOWER HEAT LOADS.
 - TPS WEIGHT PENALTY NOT STRONG FUNCTION OF VEHICLE SIZE.

- LUNAR-RETURN STRATEGY SENSITIVITY

  - DIRECT-ENTRY HAS SIGNIFICANTLY HIGHER HEATING RATES BUT AERO-CAPTURE
    LUNAR-RETURN HAS HIGHER HEAT LOADS.
  - DIRECT-ENTRY HAS LEAST TPS WEIGHT PENALTY.
  - AEROCAPTURE PLUS LOW-EARTH ORBIT ENTRY RESULTS IN AN 10- 30% (65-208 KGS)
    INCREASE IN TPS WEIGHT OVER THE DIRECT-ENTRY CASE FOR THE 1.5 SCALE VEHICLE.

- TPS WEIGHT PENALTY FOR A COMMON TPS DESIGN FOR ALL LUNAR-RETURN STRATEGIES
  IS APPROXIMATELY 7.5% FOR 1.5 SCALE VEHICLE.

- PRESENT TPS WEIGHT (SCALE=1.0) 50% LESS THAN ORIGINAL APOLLO DESIGN
  DUE TO FLIGHT EXPERIENCE AND TO IMPROVED MATERIALS.

FUTURE EFFORTS

- IMPROVE TRAJECTORY GUIDANCE ALGORITHMS.

- UTILIZE IMPROVED METHODS FOR PREDICTING HEATING ENVIRONMENTS.

- EVALUATE BENEFITS OF ADVANCED MATERIALS.

Space Habitation and Operations Module (SHOM)

Presented to

Exploration Programs Office
Technical Interchange Meeting

by

Ralph Eberhardt

May 6, 1992

MARTIN MARIETTA

B-11

Space Habitation and Operations Module (SHOM)

"Group"

■ Meetings/teleconferences

■ Eating   Pre/post EVA
                  ■

                  ■ Meal clean-up
Small group
recreation   ■ Meal Prep   ■
   ■   ■ General   Prox Op's
                        Houcokeeping   ■
"Private"   ■ Excercise   IVA support of EVA
   ■   ■ Logistics/resupply   "Public"
Private   Hand/face
recreation   cleansing   ■ Clothing maint.
        Personal ■   Planning/schedule ■   ■
        ■ hygiene ■   ORU maint.   Subsys.
   ■   Dress/undress   monitoring
Full body ■
cleansing Urination/defecation
                              ■ Training
                         ■ Payload support
                      ■ Life sciences experiment
■ Sleep   ■
                     Medical care ■ Materials proc. experiment

"Individual"

Space Habitation and Operations Module (SHOM)

"Group"

Meetings/teleconferences

WARDROOM

Small group recreation

RADIATION SHELTER

"Private"

PERSONAL HYGIENE

Private recreation

Personal hygiene

Full body cleansing

Hand/face cleansing

Dress/undress

Urination/defecation

WASTE MGT

Sleep

QUARTERS

FITNESS

Excercise

Eating   Pre/post EVA

GALLEY

Meal clean-up

Meal Prep   C³

General

Prox Ops

Housekeeping   IVA support of EVA

Logistics/resupply   "Public"

Clothing maint.

Planning/schedule

LAUNDRY   ORU maint.

Subsys. monitoring

WORK AREAS

Training

Payload support

Life sciences experiment

Materials proc. experiment

Medical care

"Individual"

MARTIN MARIETTA

572

PLAN VIEW UPPER FLOOR SECTION B SECTION A

COMMUNICATIONS COMAND & CONTROL

Workbench

Moveable Particians

WORK AREA #1

GALLEY

Counter

Food Storage

Desk

Moveable Particians

WORK AREA #2

Table

Portable Table

Door

RAD SHELTER

Workbench

Portable Table

WARDROOM

Table

Instrumentation & Equipment

1 m scale

LOWER FLOOR

Shelves

Shelves

Bed

Closet Closet

Bed

Weight Machine

FITNESS CENTER

Bicycle

QUARTERS

QUARTERS

Closet

QUARTERS

Laundry

Shelves

Bed

Closet

Head

Cab

Sink

Shower

Bench

STORAGE AREA

PERSONAL HYGIENE

SECTION B

END VIEW   SECTION A

WARDROOM

50% M

50% M

RAD SHELTER

95% M   50% M   95% F

Cables

GALLEY

Air Ducts

Cables

PERSONAL HYGIENE

QUARTERS

Air Ducts

50% M   50% M Shower   95% F   95% F

Bed

Personel Stow

STORAGE

1 m scale

Floor Space

Perceived Volume

SIDE VIEW

CONTROL ROOM   WORK AREA   GALLEY   WARDROOM

95% F

50% M   50% M   50% M   Kitchen Table

50% M

QUARTERS   Shelves   Shelves   QUARTERS

Shelves

Closet   Closet

50% M   50% M

Bed   50% M   50% M

Personel Stow   Bed

50% M   50% M   Personel Stow

STORAGE

Command &
Communications
Stations

Food Storage\
Food Preperation

Desk \ Workstation

Work Area \ Dining \ Life Sciences

Wardroom

Radiation Shelter

Audio \Visual

Fold-Away PC

Main Storage Locker

Crew Quarters

Laundry
Unit

Head

Sink

Shower

Fitness Center

Storage

Storage

Space Habitation and Operations Module (SHOM)

Martin Marietta Contacts for SHOM:

Ben Clark	(303)	971-9007
Carolyn Cooley	(303)	971-9375
Bob Boyd	(303)	971-9069
Rohan Zaveri	(303)	971-9369

MARTIN MARIETTA

Earth Land Landing Alternatives
Lunar Transportation System

Presented to the 3rd Space Exploration Initiative Technical Interchange

6 May 1992

NASA JSC EG3
(713) 483-4629

What is the LSWG?	Navigation, Control & Aeronautics Division Aeroscience Branch	
	Robert Meyerson	May 1992

The Landing Systems Working Group is a multi-disciplinary team made up of Engineering Directorate personnel organized to plan landing systems acitivity for future NASA missions.

Objectives

- Develop the land landing option so that it is a viable trade option for future NASA missions.

- Provide NASA programs with solid technical support in the landing systems area.

- Develop the technical staff.

- Advance the state of landing systems technology to apply to future NASA missions.

	Navigation, Control & Aeronautics Division Aeroscience Branch	
Study Process	Robert Meyerson	May 1992

- Two Phase Process

System Identification and Evaluation →

Products

- SOA Reviews
- Develop Mission Scenario and Requirements
- What is a Landing System?
- Form Sub-Teams
  - Descent
  - Impact
  - Avionics
  - Systems Integ.
- Define Landing Concepts
- Evaluate and I.D. Critical Areas
- I.D. Ongoing Programs
- Team Formation
- State-of-the-Art Assessment
- Identify Customers & Interviewees
- Interview Customers
- Define Customer Needs
- Match ? — Yes / No
- Develop Action Plan
- JSC Activities Plan
- Sandia Plan

NASA Needs Assessment ⟶

- Needs Assessment Complete
- System Evaluation in progress
- **Study to be complete in June 1992**

Preliminary Needs Assessment	Navigation, Control & Aeronautics Division Aeroscience Branch	
	Robert Meyerson	May 1992

- Needs identified from interviews with Program, Engineering User representatives at NASA JSC.
  - Interviews performed in November 1991

Inhouse expertise and technology base
Technology assessment of outside efforts
Focal point for landing systems issues
Perform entry through landing flight tests

Simple, Affordable, Reliable, and Safe space vehicles
Automatic/Autonomous Landing Capability
Minimize crew input and training workload
Assurance of vehicle readiness
Reduce system cost

Focus on critical technical areas
Descent - Reduce footprint, gliding chutes
Avionics - Automatic landing, DMS, Sensors
Impact Attenuation - Retrorockets, no tumbling

Plan Facility Development
Take advantage of existing facilities

	Navigation, Control & Aeronautics Division Aeroscience Branch	
Program Requirements	Robert Meyerson	May 1992

GOAL - "Develop reliable manned land landing capability"

- Use 8-crew SCRAM concept as baseline vehicle for concept evaluation
- Land landing at a pre-selected, unprepared landing zone (with water backup)
  - Immediate post-landing access
  - Landing site size = 5 mile diameter nominal (10 mile dia emergency)
  - No ground preparation
  - No navigation aids at the site
  - Weather limits from ACRV SEDB (90th percentile nominal, 99th percentile for emergency)
  - Service module jettisoned during entry
- No tumbling
- Physiological constraints = +/- 15 Gx, +/- 10 Gy, +8/-8.5 Gz Nominal
  - +40/-30 Gx, +/-20 Gy, +25,-20 Gz Emergency
- Night landing capability
- Single fault tolerance
- Fully automatic operations
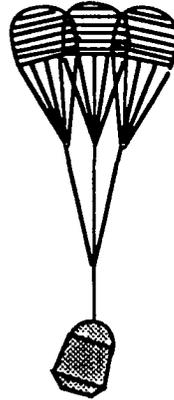- 15,000 lb landed weight
- Consider reusabilty in design

Option 1 - Apollo Land Lander

Drogue
Deploy
Mach 1.5

Conventional Parachutes deployed at
5,000 - 15,000 ft

Steady State Descent
Vv = 20 - 40 fps

Touchdown with Vx = Wind Speed
(no external impact attenuation)
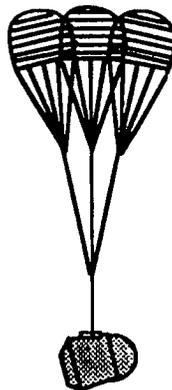Jettison Parachutes

Landing
Site

R. E. Meyerson/NASA JSC EG3

Option 2 - Advanced Apollo

Drogue
Deploy
Mach 1.5

Conventional Parachutes deployed
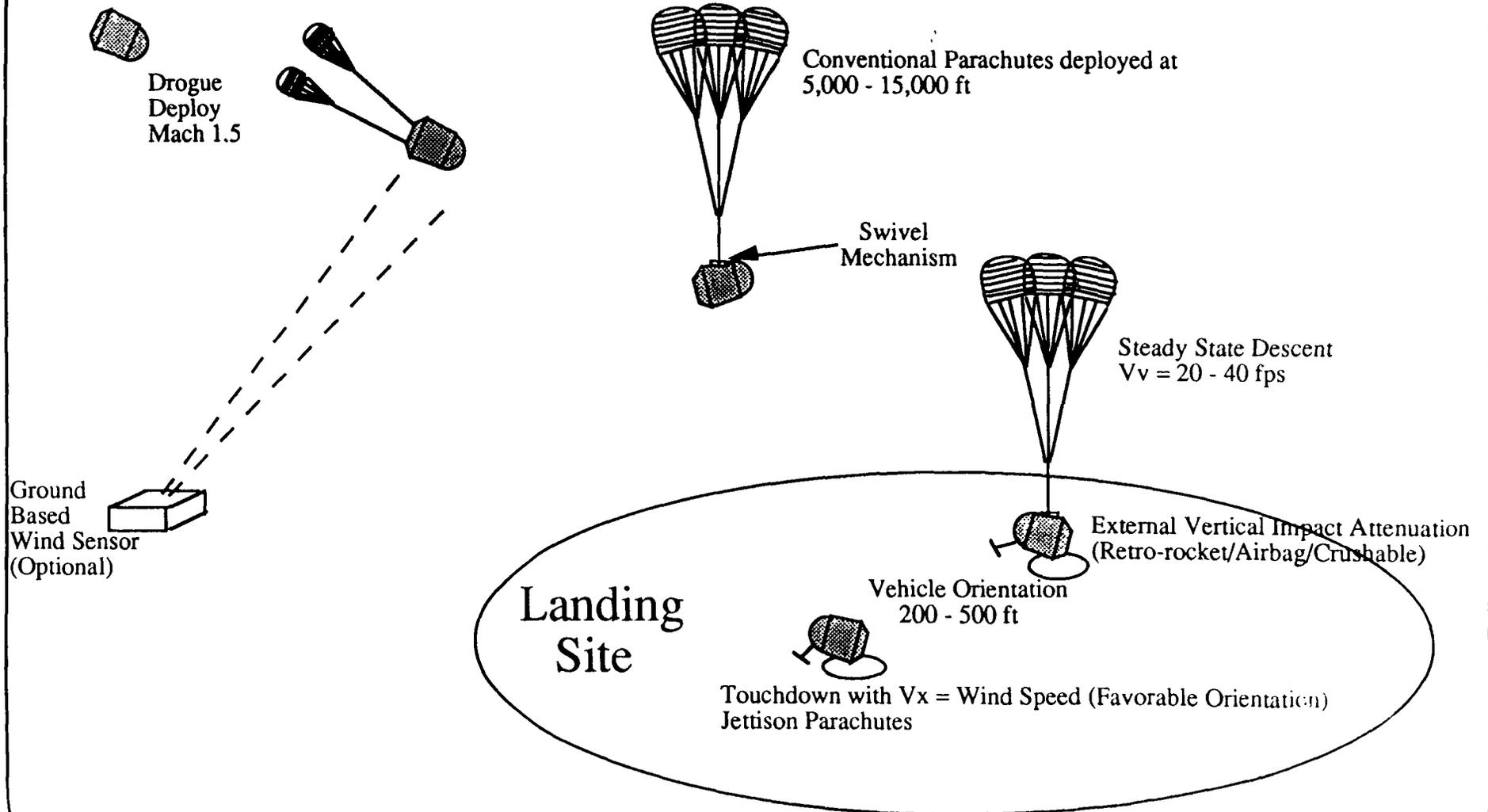at 5,000 - 15,000 ft

Steady State Descent
Vv = 20 - 40 fps

Landing
Site

External Vertical Impact
Attenuation
(Retro-rocket/Airbag/Crushable)

Touchdown with Vx = Wind Speed
Jettison Parachutes

Option 3 - Advanced Apollo with Orientation

Drogue Deploy Mach 1.5

Conventional Parachutes deployed at 5,000 - 15,000 ft

Swivel Mechanism

Steady State Descent Vv = 20 - 40 fps

Ground Based Wind Sensor (Optional)

External Vertical Impact Attenuation (Retro-rocket/Airbag/Crushable)

Vehicle Orientation 200 - 500 ft

Landing Site

Touchdown with Vx = Wind Speed (Favorable Orientation) Jettison Parachutes

R. E. Meyerson/NASA JSC EG3
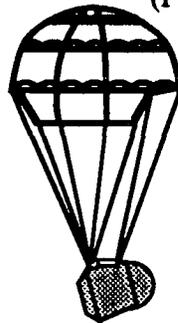
Option 4 - "Gemini Soft Lander/Apollo Land Lander"

Drogue
Deploy
Mach 1.5
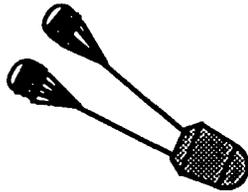
Low to Mid-L/D Parachute System
(Parasail/Cloverleaf/Tojo-slot)

Steer to Site
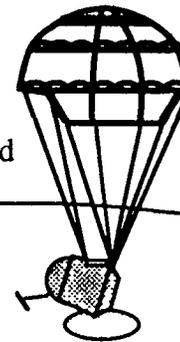
Steer into Wind

Landing
Site

External Vertical Impact Attenuation
(Retro-rocket/Airbag/Crushable)

Touchdown with Vx = Glide Speed - Wind Speed
Jettison Parachutes

Option 5 - Advanced Recovery System

Drogue
Deploy
Mach 1.5

Deploy and Disreef
High Glide System
(Parafoil/Parawing/Sailwing)
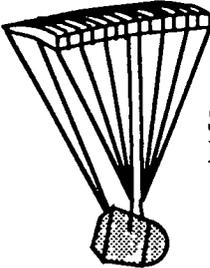
Steady Glide and
Maneuver to Site
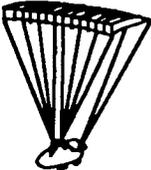
Steer into Wind
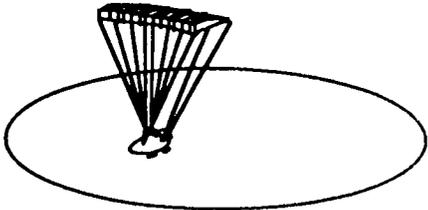
Perform Landing Flare Maneuver
(Vx < 30 fps, Vz < 10 fps)

R. E. Meyerson/NASA JSC EG3

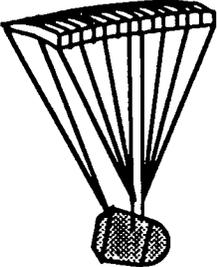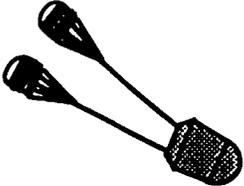|	**Development of Activities Plan**	Navigation, Control & Aeronautics Division Aeroscience Branch
		Robert Meyerson May 1992

- Evaluation of the 5 landing options will produce a list of technical and system integration "holes"
  - Does the system meet the requirements?
  - Can we design/analyze/test the system today?

- These "holes" will be prioritized based on a criteria to include the following:
  - Technology/Development risk and cost
  - Applicability to future NASA programs
  - Performance Improvement

- Ongoing programs will be identified which are addressing these "holes"

- A development plan will be produced to address remaining "holes" including the following:
  - Proposed activity
  - Performing organization
  - Requirement/Justification
  - Schedule
  - Cost estimate
  - Impact if not completed

Issues with Conventional Systems	Navigation, Control & Aeronautics Division Aeroscience Branch	
	Robert Meyerson	May 1992

Parachute/Cluster Opening Loads - Major design issue on Apollo ELS

Orientation System Implementation - Where? How?

Reduce Landing Footprint - Improve guidance through entry. Vehicle based wind sensing would help.

Dynamics of Ground Contact and Energy Mitigation - Soil modelling. Impact performance prediction. Effects of shape on impact.

Issues with Gliding Systems	Navigation, Control & Aeronautics Division Aeroscience Branch	
	Robert Meyerson	May 1992

Inflation - Asymetry about one axis makes conventional loads management difficult.

Reliability - Large use database on personnel size systems. Never demonstrated in the 15,000 - 20,000 lb size. Not scalable. How do we deal with redundancy?

High Horizontal Velocity at TD - Higher wing loading required for weight efficiency.

Landing Flare Maneuver - Maneuver requires precise timing. Autonomous system never demonstrated at any scale.

Reduced Knowledge Base - Wealth of knowledge developed during Apollo program is no longer available.

	Ongoing Programs	Navigation, Control & Aeronautics Division Aeroscience Branch	
		Robert Meyerson	May 1992

- **Low Altitude Retrorocket System (U.S. Army Natick/AAI Corp.)**
  - Air Delivery system for unmanned Army payloads up to 60,000 lbs
  - Designed for deployment at 250 KEAS and 300 ft AGL
  - Includes riser mounted rocket, laser altitude sensor, processor

- **F-111 Crew Escape Module (U.S. Air Force)**
  - Recertification of parachutes and airbags
  - Impact testing performed at NASA Langley IDRF (Fall 1991)
  - Airdrop testing at NAWC - China Lake (Summer 1992)

- **Advanced Recovery Systems Program (NASA MSFC/U.S. Army)**
  - Demonstrate the capability to soft land, in a precisely controlled manner, a large object selected to simulate a launch component, such as P/AM.
  - Several tests performed with W < 13,600 lbs and S = 3600 sq ft.
  - Phase II Complete with no current plans to fund Phase III

- **Dryden Autoland Experiments (Joint NASA DFRF/JSC)**
  - Develop and demonstrate an autonomous landing flare maneuver using a simulated spacecraft with a gliding parachute and GPS
  - Tests to begin in June 1992

Johnson Space Center – Houston, Texas

Summary & Recommendations	Navigation, Control & Aeronautics Division Aeroscience Branch	
	Robert Meyerson	May 1992

- Landing Systems knowledge base is being developed

- Issues with land landing systems are being identified for a variety of parachute system options

- There are few **technology** issues with a conventional parachute with retrorocket system
  - This may not meet the system requirements

- Gliding parachutes may provide the most operational flexiblity but also create large technology risks
  - Reliability will always be an issue with current gliding concepts

- Land landing **CAN** be done

RECOMMENDATION: Address the following issues...
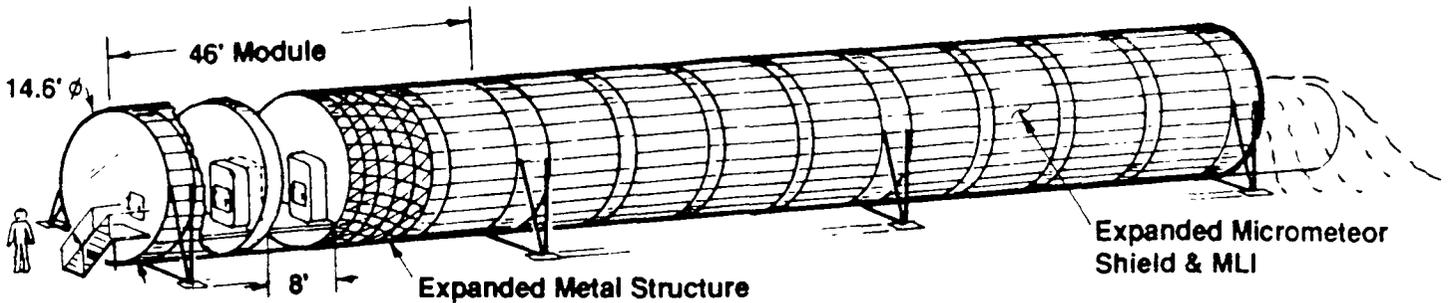Landing Footprint/Site Selection
Ground impact dynamics
Gliding parachute reliability/redundancy

PNEUMATICALLY ERECTED RIGID HABITAT

The pneumatically erected rigid habitat concept consists of a structure based on an overexpanded metal bellows. The structure is reinforced with rings and stringers in typical airframe construction technique. The achievable expansion ratio is easily five to one thus allowing a small volume during launch. An integrated micrometeorite shield and MLI consists of an overexpanded titanium honeycomb. Each module consists of four sections each separated by pressure bulkheads and airlocks. The module also has a self contained life support system. The module is designed so that it can be chained to other modules to construct a full base station.

The basic concept of a pneumatically erected rigid habitat incorporates the advantages of both the inflatable and rigid structures. The inflatable approach allows the number of launches needed to orbit a mission to be greatly reduced. The heavy launch vehicles in use can orbit a full payload volume worth of collapsed hardware, thus reducing the number of launches needed by a factor of 6x, and the cost proportionately. The prime disadvantage to this approach is problems with high maintainability times and the need for frequent repairs of inflatable structures. The aluminum rigid structure is extremely durable and low maintenance but has an extremely large launch volume. The pneumatically erected rigid habitat thus addresses the launch volume and provides a durable low maintenance structure.

The pneumatically erected rigid habitat is built up of standard aluminum construction techniques using rings and stringers, incorporating normal rip stop safety protection, but provides a structure that can be erected from a compressed shape. The structure would be built in the collapsed form, much as a set of metal bellows. During the deployment the structure would be over pressurized internally, yielding the bellow structure radially into a standard tank or fuselage structure. Because of the yielding which takes place, the process is not reversible, and the same strength and other structural properties are achieved as if the structure was fabricated in the normal manner. Figure 1 shows the collapsed launch configuration, the non extended configuration on the lunar surface and erected structure.

Pneumatically Erected Rigid Habitat

PATENT PENDING

96" Between Bulkheads

Elliptical Bulkheads

14' 7" Diameter
13' 7" Inside Living Diameter

Air Lock Between Sections

Titanium Honeycomb Micrometeorite Shield Deployed As Erected

Radiation Shelter

Aluminum Skin

Load Ring Stringers

Module, Independent Life Supporting Subassembly Made of 4 Sections

Module Interface Ring, Gas Storage and Life Support System for 4 Sections

FULLY EXTENDED HABITAT ON LUNAR SURFACE

Living Wall, Zeppel Coated Nomex

Load Ring

Removable Floor Panels

Expanded Metal Joists

Micrometeorite Shield

Pressure Wall, Aluminum .025 Thick

WALL DETAIL

NON EXTENDED ON SURFACE

MODULE DRY WEIGHT 3450 LBS.
(No Equipment, Life Support, or Tankage Included)

15'

Module (4 Sections)

24"/Section

61' 6"

86'

LAUNCH CONFIGURATION
TITAN IV D
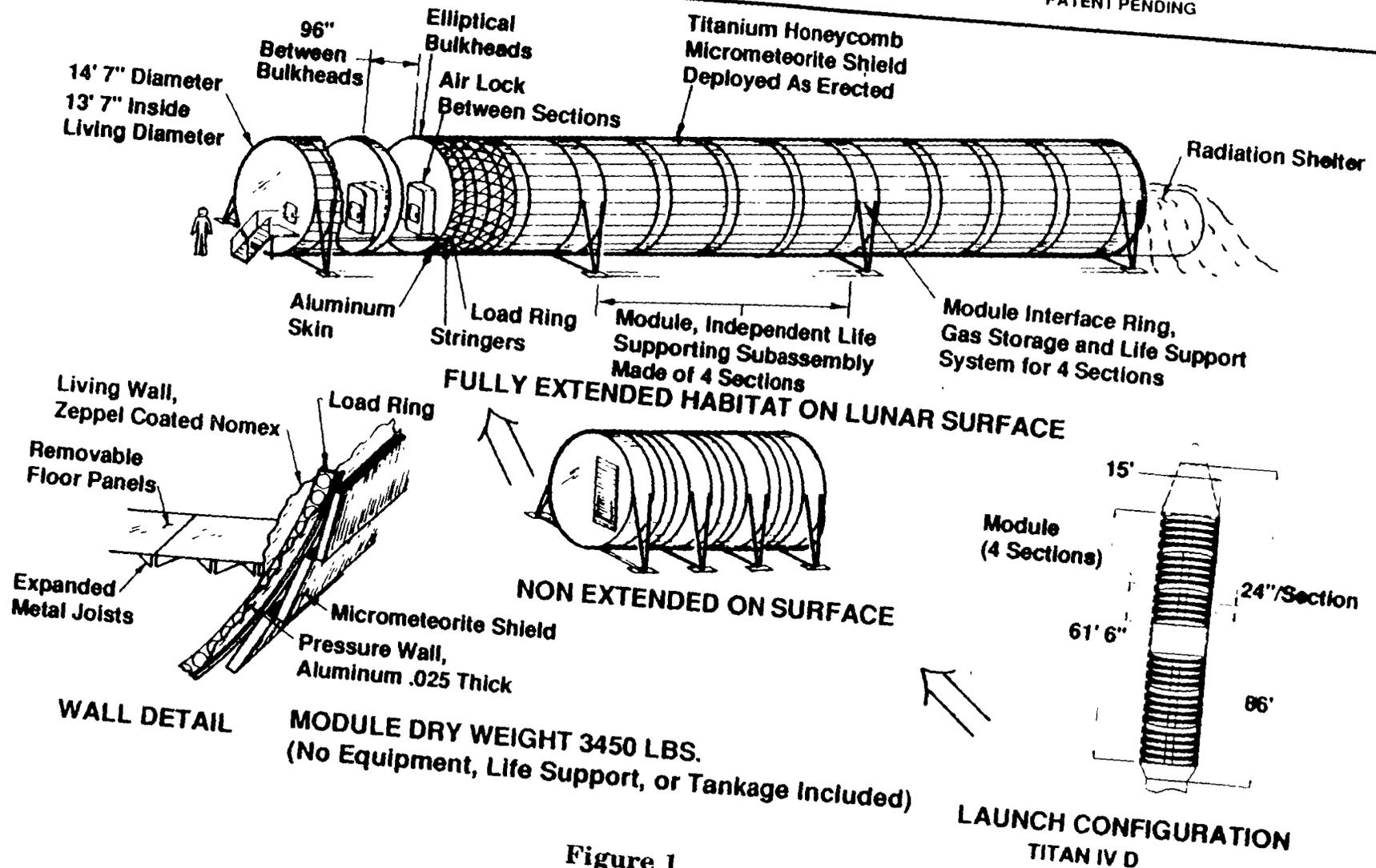
Figure 1.

Key Features

The key features of the concept, shown in figure 1 are: The multiple sections separated by airlocks for redundant depressurization protection. The expanded transfer rooms, allowing easy, full suit transfer. The expanded floor joists with removable floor panels, allowing easy access below decks and allowing a rigid floor to be erected. The elliptical pressure bulkheads which isolate each section and provide stowage space and attachment for equipment and the floor panels during launch. The inner living wall, which gives a redundant pressure wall, and provides protection to the structural wall from inadvertent puncture from within. The expanded titanium honeycomb micrometeor shield and MLI which provides a light weight "bring from home" protection that deploys autonomously during the habitat erection. To provide radiation protection a smaller expandable section is shown which can be covered with lunar soil. This would provide protection for the crew during periods of high radiation, but would not require bringing a lunar bulldozer. The dry weight of the concept is 3450 pounds per module (four sections) with no equipment, life support systems or tankage included. A total of six sections could be put into low earth orbit using a Titan IV D, with weight capacity for the modules, life support systems, tankage, and some modest equipment.

Erection Details

The primary design driver in developing the Pneumatically Erected Rigid Habitat (PERH), is forming the collapsed structure in a smooth bellow like fashion with no sharp bends, and maintain this no sharp bend or kink configuration throughout the erection process. The other design driver is to limit the elongation of any portion of the structure in yield to 50% of the allowable for the materials being used.

By using a bellows type structure shown in Figure 2, with rings on the outside of alternate crests or fold and a triangular stringer arrangement, the erection process takes place in three distinct well controlled steps. In the first step, at low pressure, the inflated habitat section elongates axially. During this stage stresses in the rings are negligible, stresses in the skin are below yield, but the stringers are yielding into a near straight condition. During the second stage of inflation the skin begins to yield outward, allowing the stringers to carry almost all the axial load of inflation and yielding them into near final condition. The last stage of inflation the skin is yielded and continues to bulge outward between the rings and stringers in diaphragm loading until the diaphragm deflection lowers the skin stress below yield. This process is thus self regulating and fairly insensitive to either inflation pressure tolerances or variations in material properties from one lot to the next. After the final inflation stage the pressure is lowered back to normal inhabitation pressure. The excess gas is use to inflate the next section of the habitat.

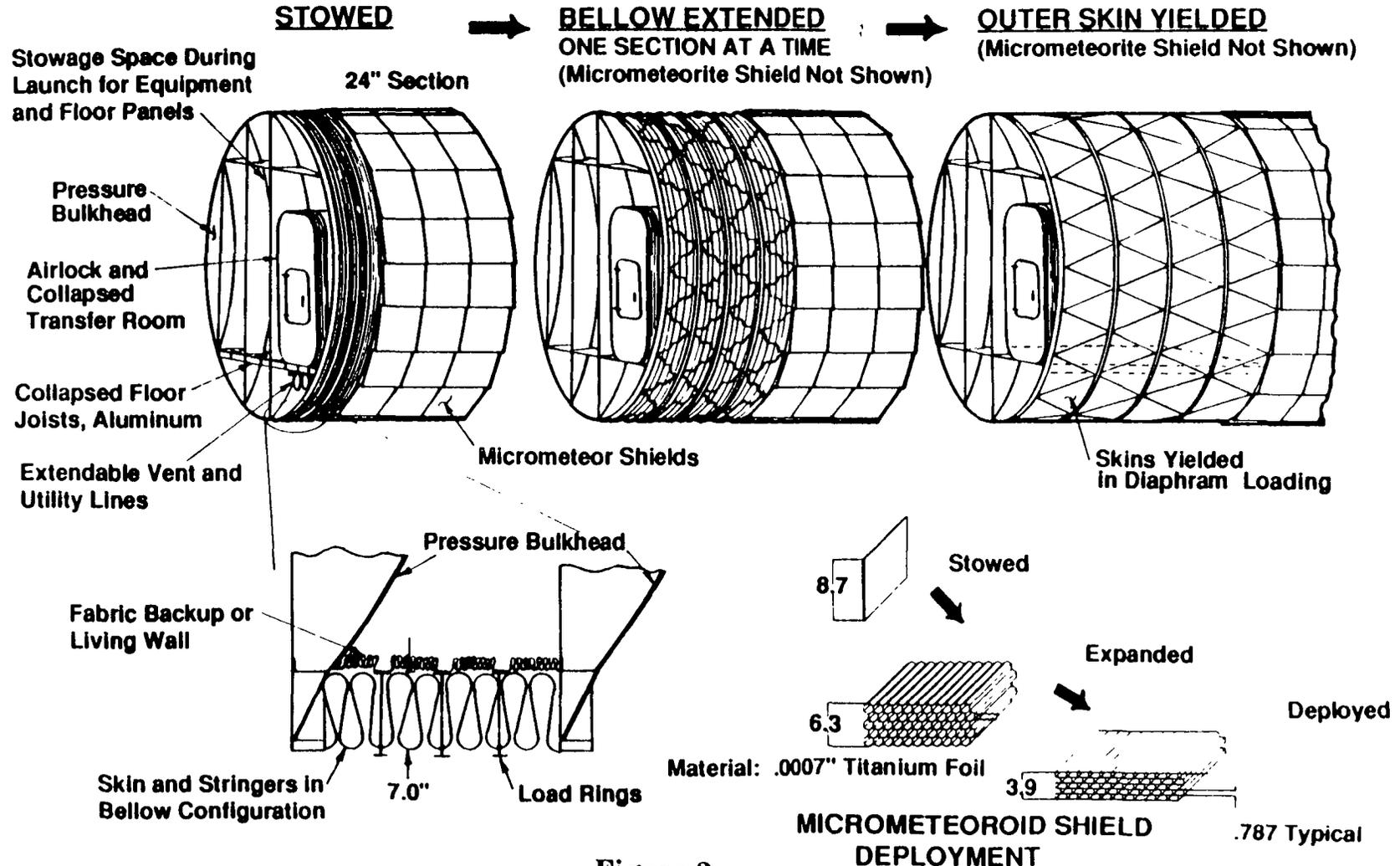# Erection Details, Pneumatically Erected Rigid Habitat

PATENT PENDING

STOWED ➡ BELLOW EXTENDED ➡ OUTER SKIN YIELDED

BELLOW EXTENDED
ONE SECTION AT A TIME
(Micrometeorite Shield Not Shown)

OUTER SKIN YIELDED
(Micrometeorite Shield Not Shown)

Stowage Space During
Launch for Equipment
and Floor Panels

24" Section

Pressure Bulkhead

Airlock and
Collapsed
Transfer Room

Collapsed Floor
Joists, Aluminum

Extendable Vent and
Utility Lines

Micrometeor Shields

Skins Yielded
in Diaphram Loading

Pressure Bulkhead

Fabric Backup or
Living Wall

Skin and Stringers in
Bellow Configuration

7.0"

Load Rings

Stowed

8.7

Expanded

Deployed

6.3

Material: .0007" Titanium Foil

3.9

MICROMETEOROID SHIELD
DEPLOYMENT

.787 Typical

Figure 2.

TRACOR

Conclusions

The expanded metal structure of the pneumatically erected rigid habitat and its micrometeor shield allows a large inhabitable structure to be transported to the moon with a minimum launch cost. The structure draws upon the excellent flight experience with aluminum structures, without paying a large cost penalty or being forced to use excessively compact living quarters.

PLUME INDUCED ENVIRONMENTS ON FUTURE LUNAR MISSION VEHICLES

Bill Rochelle and Ruston Hughes
Lockheed Engineering & Sciences Co., Houston, TX

Steve Fitzgerald
NASA/Johnson Space Center, Houston, TX
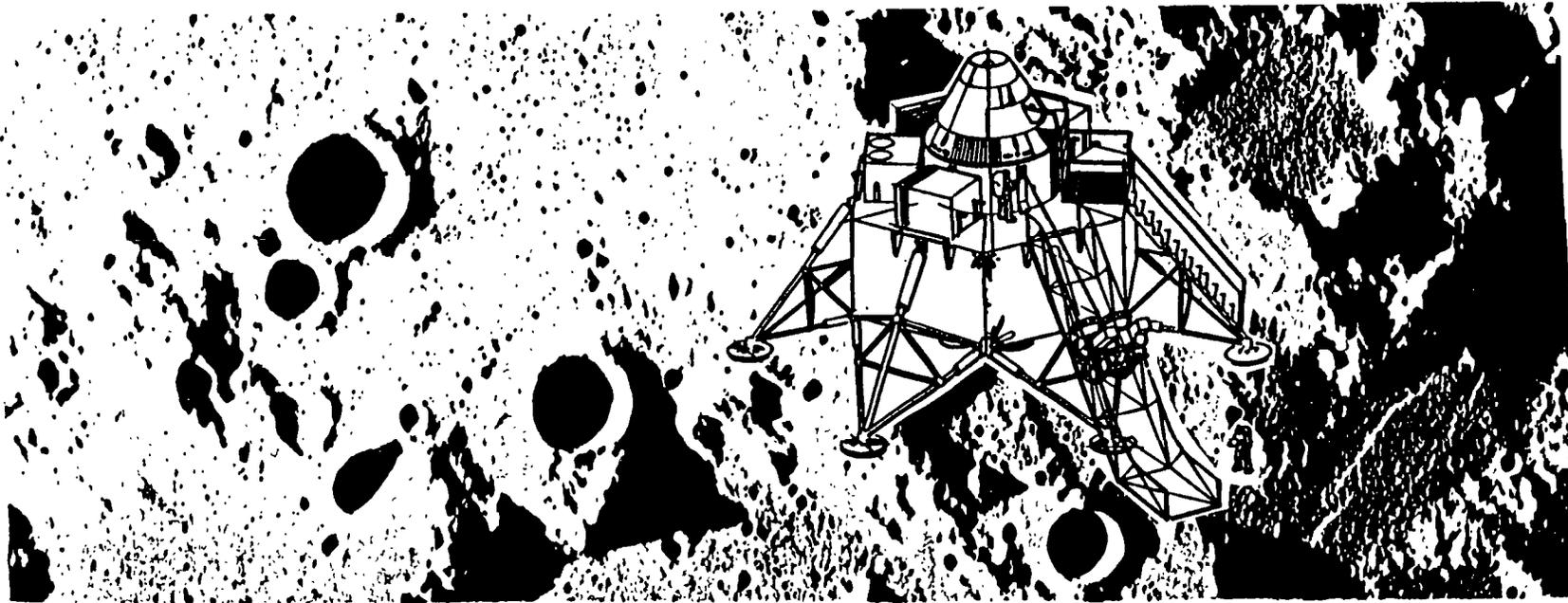
SEI 3RD TECHNICAL INTERCHANGE MEETING
MAY 5-6, 1992
HOUSTON, TX

OVERVIEW OF PRESENTATION

- **Objective:** To identify potential plume heating/impingement problem areas on vehicles used for future lunar missions

- **Basis of Experience:** Comparison with Lunar Module plume investigations performed during 1968 - 1971

- **New Vehicles Considered:**
  - Common Lunar Lander (Artemis)
  - First Lunar Outpost (FLO)
    - Cargo Lander
    - Crew Lander

- **Engines Considered:**
  - Reaction Control System (RCS)
  - Lunar Descent Stage
  - Lunar Ascent Stage

- **Plume Technology:**
  - Available Computer Codes
  - Available Test Facilities

LUNAR MODULE PLUME INDUCED ENVIRONMENT EFFECTS

A. <u>Vehicle Design Issues</u>

- **RCS engine plume impingement on sides and base of LM (installation of plume deflector or "coal chute")**

- **Descent engine plume impingement to LM landing gear**

- **Descent engine plume soil erosion on lunar surface**

- **Descent engine high base pressure during touchdown**

- **Ascent engine plume impingement on Early Apollo Scientific Experiment Package (EASEP)**

- **Ascent engine plume impingement on Apollo Lunar Scientific Experiment Package (ALSEP)**

- **Ascent engine plume impingement on top of LM descent stage**

B. <u>Ground Testing Used for Vehicle Design</u>

- **Full scale ground tests of RCS engine plume in JSC TTA and Chamber A facilities**

- **Scale model ground tests of descent stage engine plume at Grumman detonation facility**

FIG. 8 CLOSE-UP OF LM RCS PLUME DEFLECTOR FOR APOLLO 16
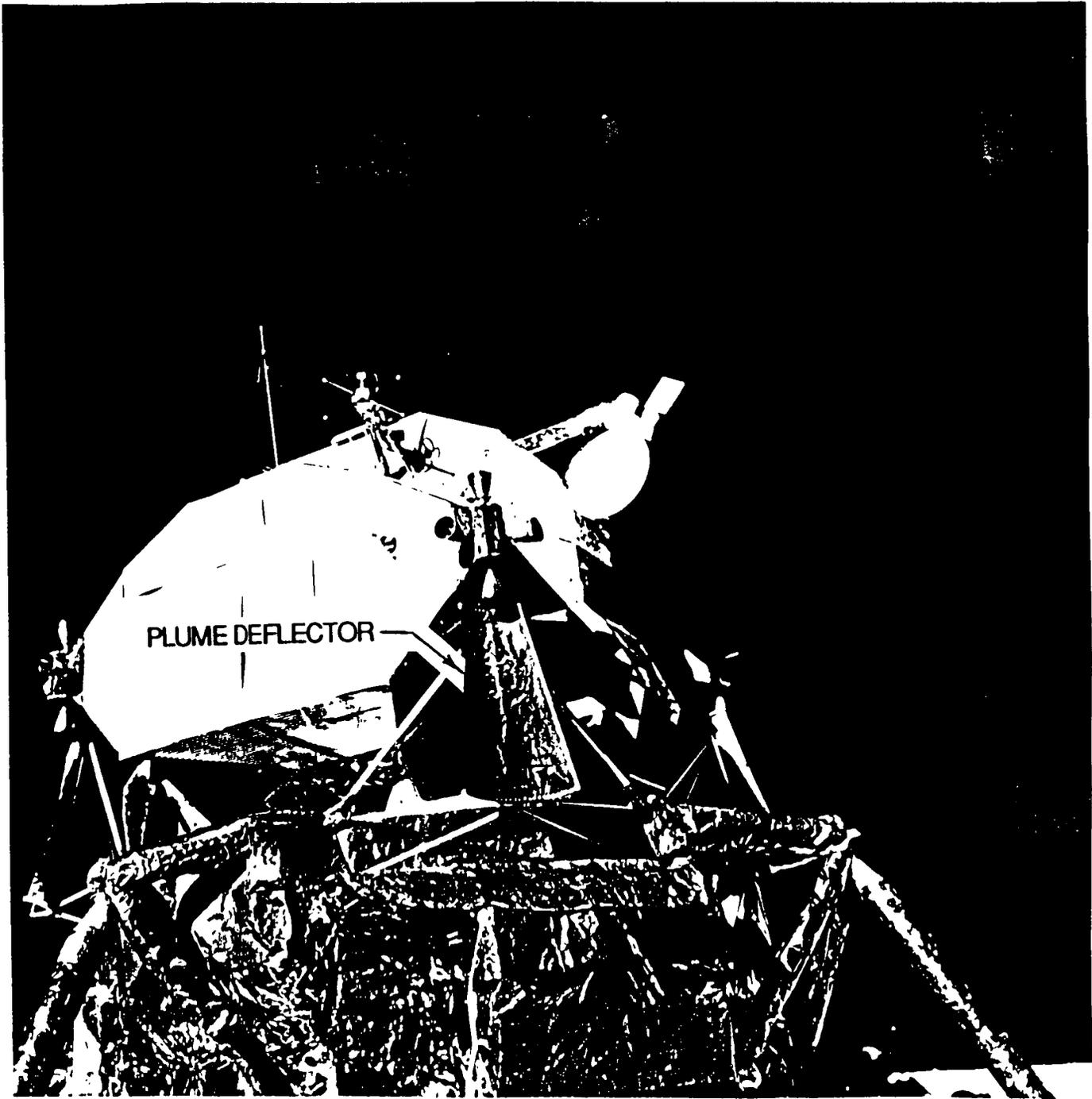

PLUME DEFLECTOR

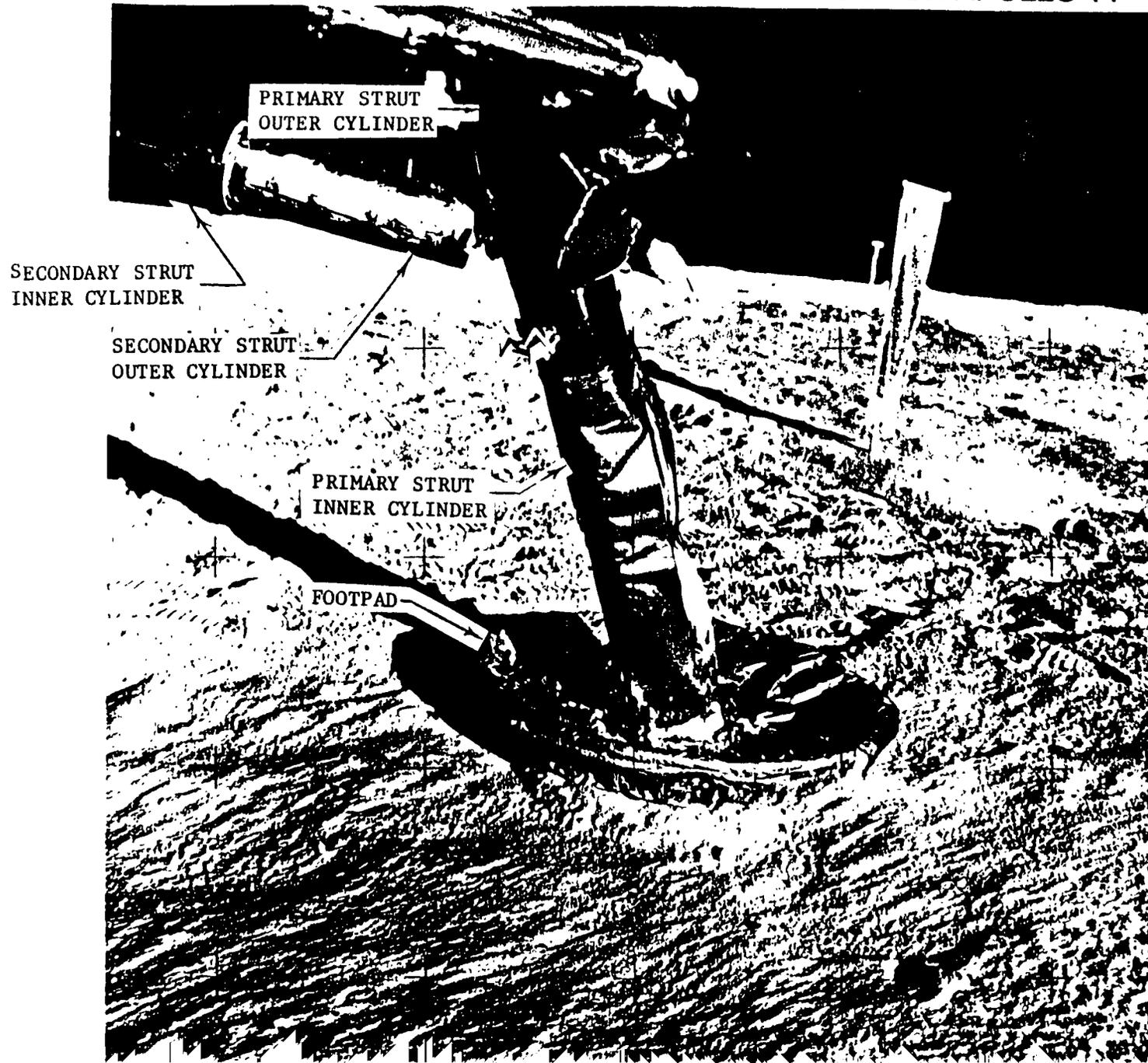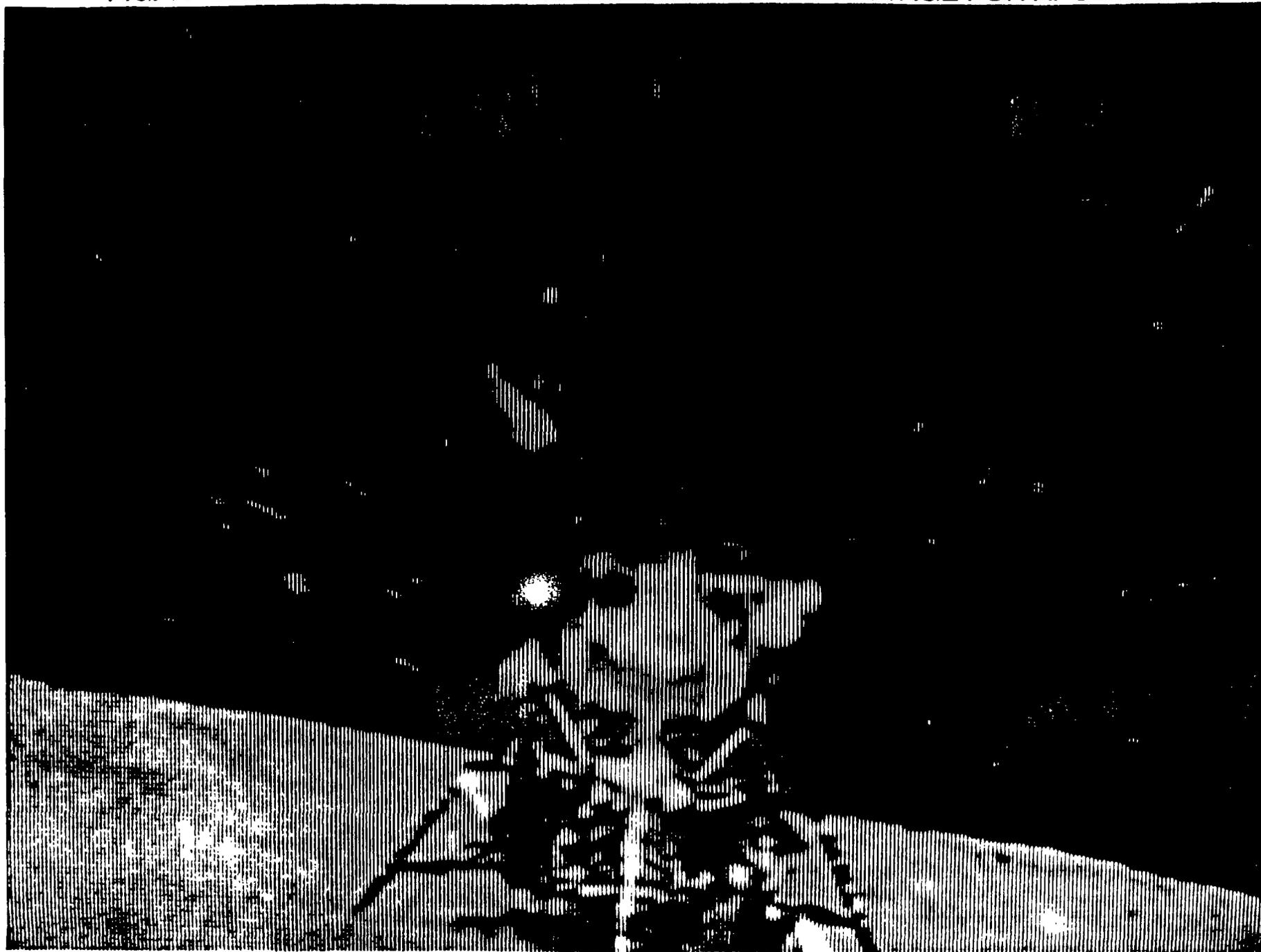FIG. 9 LM LANDING GEAR ON LUNAR SURFACE FOR APOLLO 11

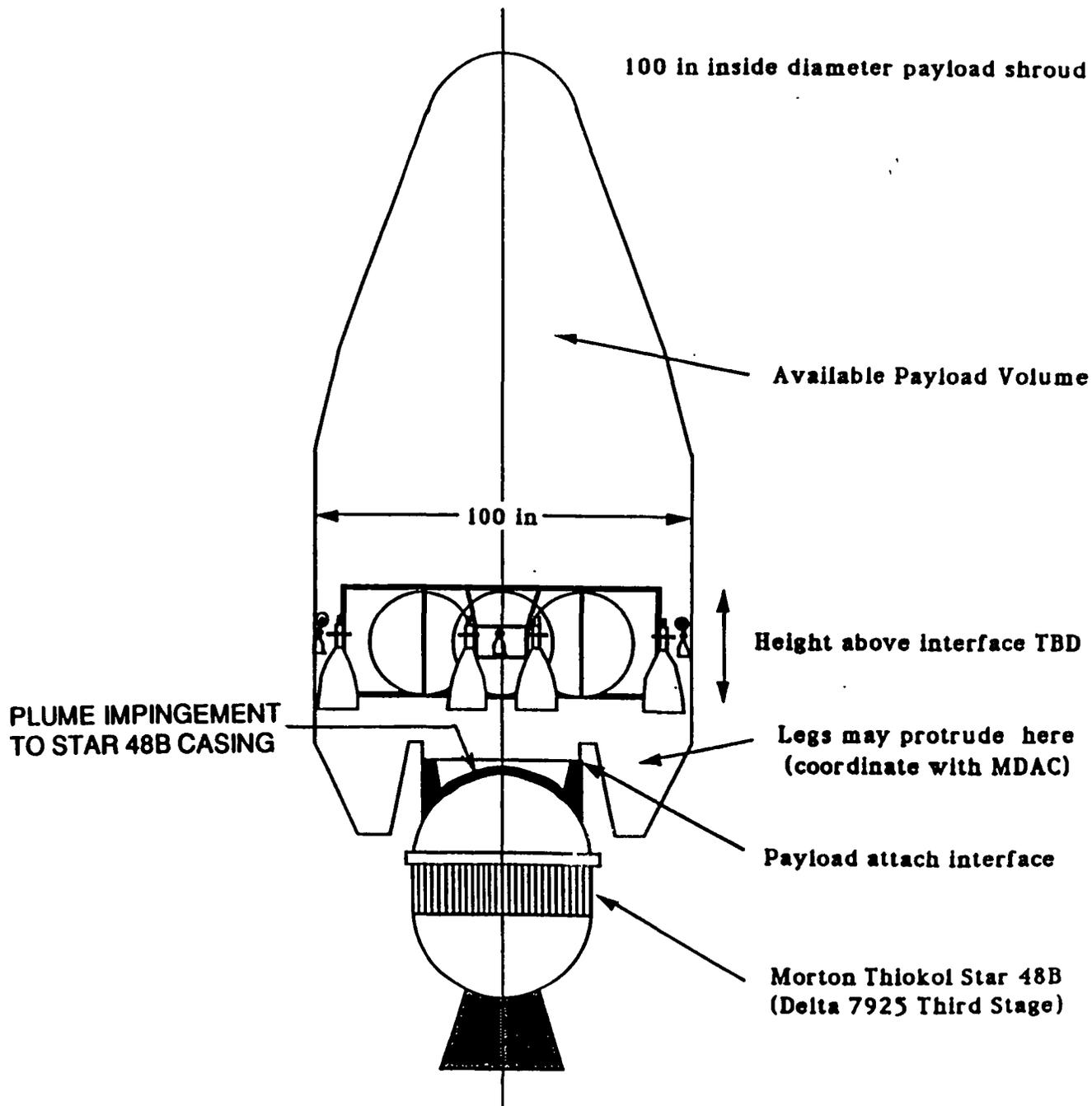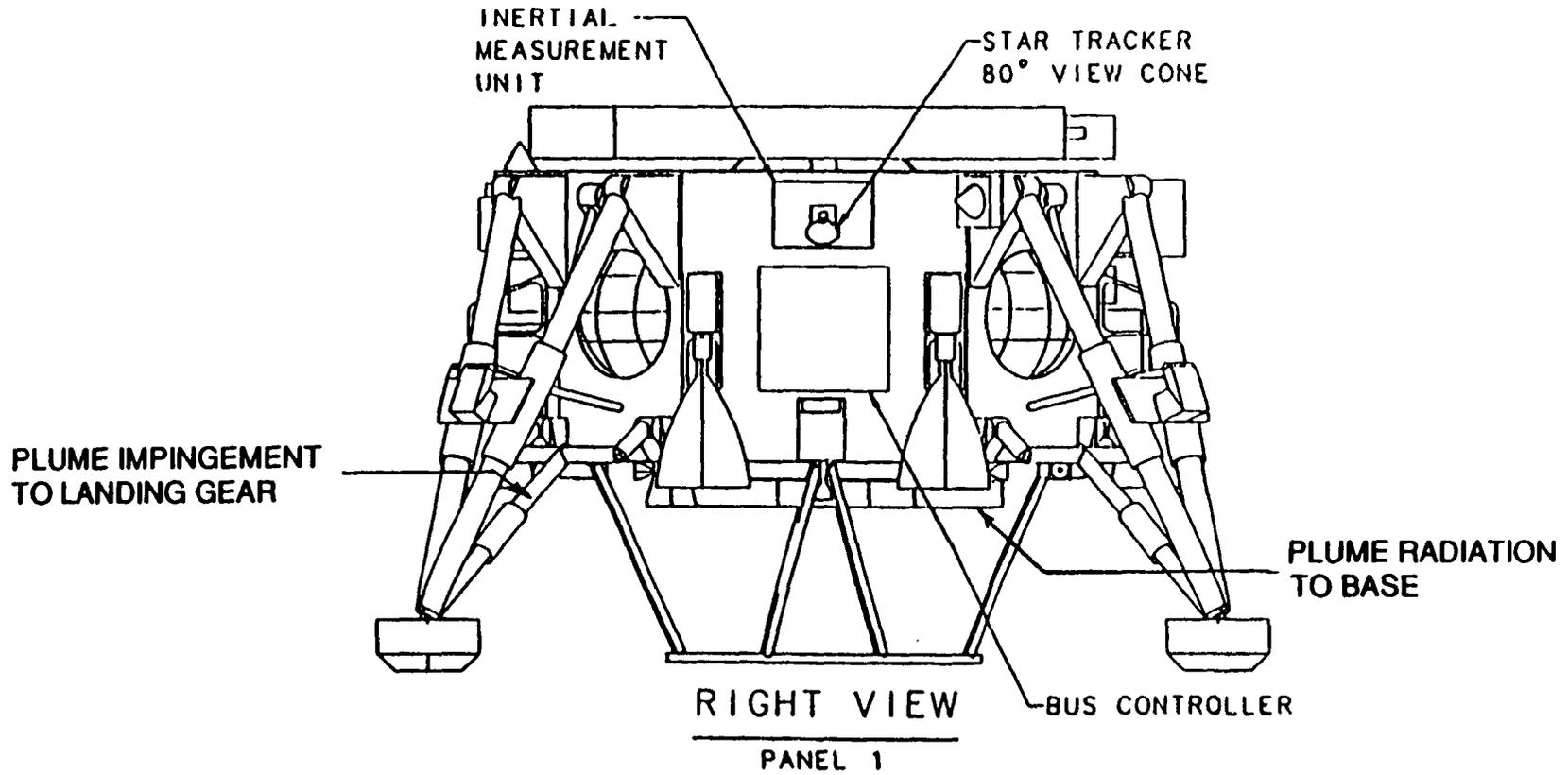FIG. 10 LM ASCENT PLUME IMPINGEMENT ON DESCENT STAGE FOR APOLLO 16

TABLE I. SUMMARY OF CLL PLUME HEATING/IMPINGEMENT EFFECTS

Engine	Vehicle	Mission Phase	Plume Effect
RCS	Lander	TLI & Descent	Impingement on bottom of lander
			Impingement on landing gear
Descent	Lander	TLI	Impingement on casing of STAR 48B SRM
		TLI & Descent	Impingement on landing gear
			Radiation to base
		Descent	Soil erosion on base and landing gear
			High base pressure at touchdown

CLL STAR 48B SRM AND PAYLOAD SHROUD

100 in inside diameter payload shroud

Available Payload Volume

100 in

Height above interface TBD

PLUME IMPINGEMENT
TO STAR 48B CASING

Legs may protrude here
(coordinate with MDAC)

Payload attach interface

Morton Thiokol Star 48B
(Delta 7925 Third Stage)

RIGHT SIDE VIEW OF CLL VEHICLE

INERTIAL MEASUREMENT UNIT

STAR TRACKER 80° VIEW CONE

PLUME IMPINGEMENT TO LANDING GEAR

PLUME RADIATION TO BASE

BUS CONTROLLER

RIGHT VIEW

PANEL 1

BOTTOM VIEW OF CLL VEHICLE

PLUME RADIATION
TO BASE

RCS JETS

ALTIMETER
RADAR

RCS JETS

PLUME IMPINGEMENT
TO LANDING GEAR

BOTTOM VIEW
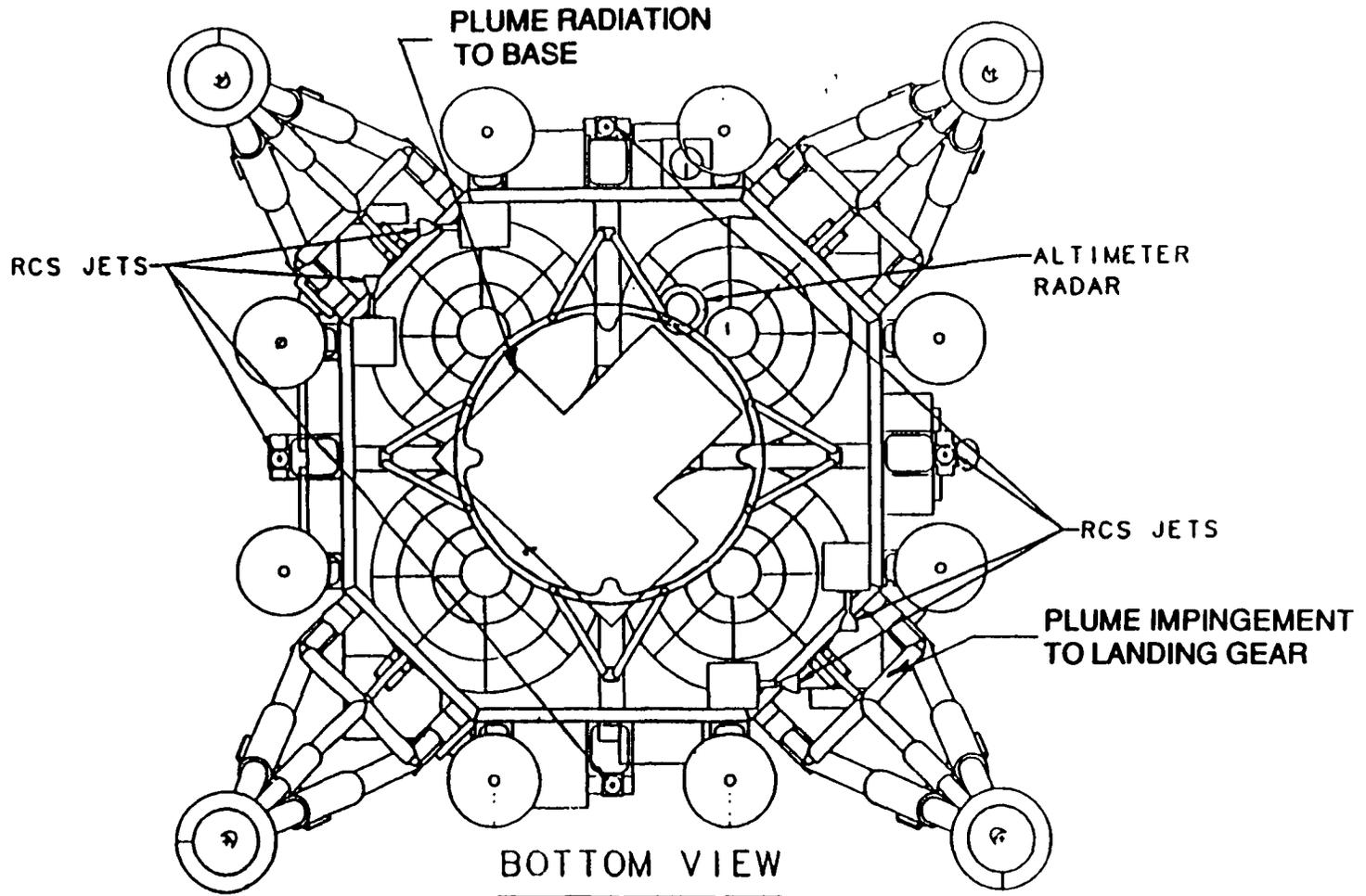
TABLE II. SUMMARY OF FLO PLUME HEATING/IMPINGEMENT EFFECTS

Engine	Vehicle	Mission Phase	Plume Effect
RCS	Cargo Lander	On-orbit	Impingement on habitat module
	Cargo Lander	On-orbit	Impingement on sides and base
	Crew Lander	On-orbit	Impingement on sides and base
	Crew Lander	Descent	Impingement on sides and base
	Crew Lander	Ascent	Impingement on sides and base
Descent Stage	Cargo Lander	Descent	Radiation to base
		Descent	Impingement on landing gear
		Descent	Soil erosion on base & landing gear
		Descent	High base pressure at touchdown
	Crew Lander	Descent	Radiation to base
		Descent	Impingement on landing gear
		Descent	Soil erosion on base & landing gear
		Descent	High base pressure at touchdown
Ascent Stage	Crew Lander	Ascent	Radiation to base
		Ascent	Impingement to top of descent stage
		Ascent	Soil erosion to comports left on lunar surface
		Ascent	High base pressure at lift-off
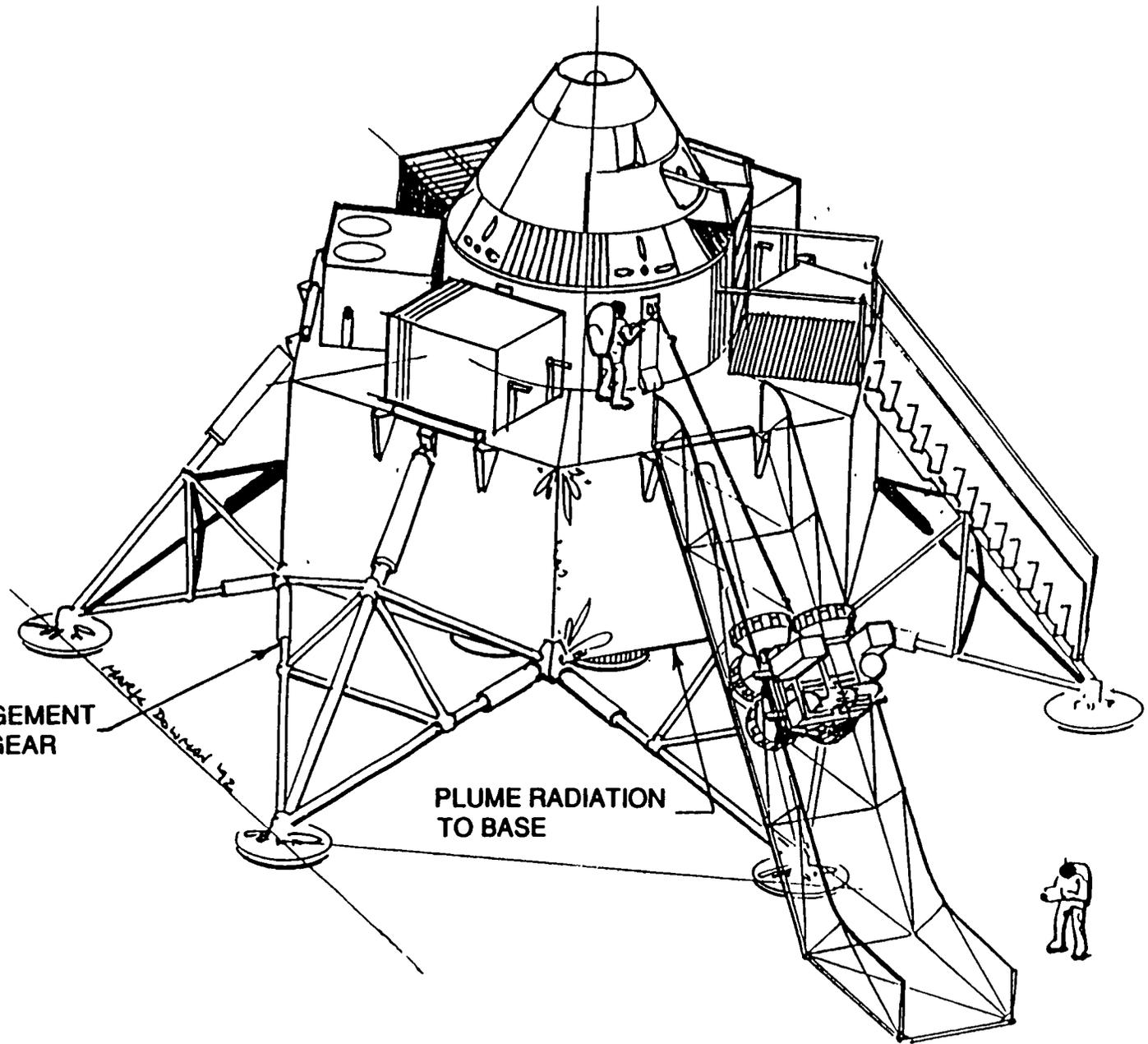
PLUME IMPINGEMENT TO LANDING GEAR

PLUME RADIATION TO BASE

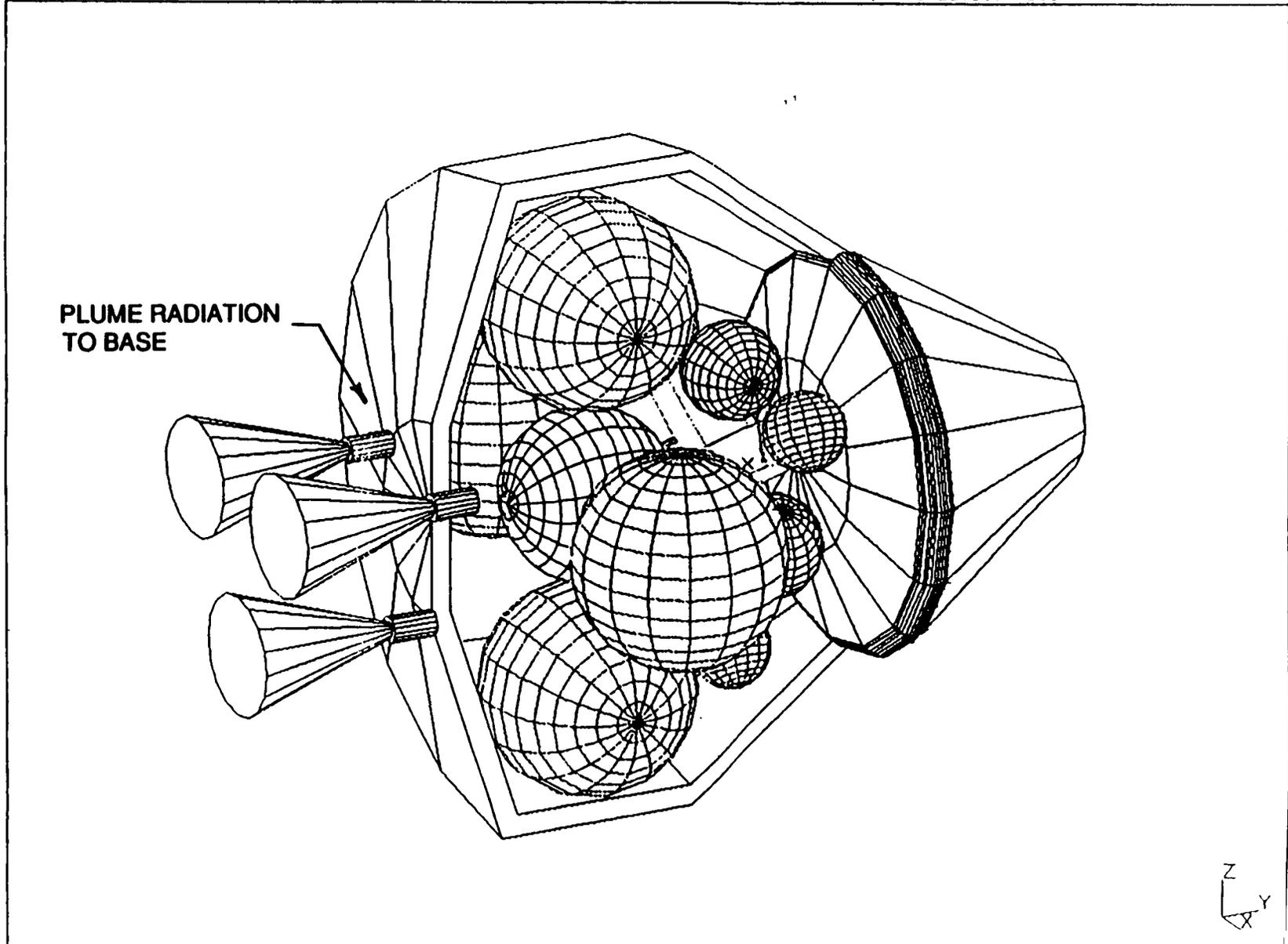# SIDE VIEW OF FLO CREW LANDER ASCENT STAGE

SDRC   I-DEAS V: Solid_Modeling   25-FEB-92   15:45:01

Database: Ascent/TEI Stage   Units   : SI
View   : ASCENT CUTAWAY   Display : No stored Option
  Task: ASSEMBLY   Bin:50-ASCENT STAGE
System: 101-CUT AS   Update Level: Full

PLUME RADIATION
TO BASE

Z
Y
X

GENERIC PLUME TECHNOLOGY NEEDED FOR LUNAR MISSIONS

Technology	Available Computer Codes/Facilities
• Flow Fields	
• Nozzle	CEC, RPP
• Plume	RAMP, CHARM, VISMOC, SFPLIMP, DSMC
• Surface Loads (Pressures, Forces, Moments, etc.)	RCSFORCE, PIM900, SFPLIMP, SFPDQ
• Heating	
• Radiation	SIRRM, CHARM
• Convection	VISMOC, BLIMP, PLIMP, CHENG, DSMC, Bridging Functions, etc.
• Soil Erosion	Roberts Theory
• Contamination	CONTAM, PROX, MOLFLUX
• Temperatures	SINDA
• Ground Tests	Full Scale: JSC/TTA & Chamber A Scale Model: CALSPAN, Grumman

CONCLUSIONS AND RECOMMENDATIONS

- CLL and FLO vehicles represent the first spacecraft to return to the moon in a generation

- Plume heating/impingement problems will exist on new lunar vehicles, similar to those of the LM 20-25 years ago

- Plume effects could result in potential damage to critical structures and components of the new lunar vehicles if these effects are not assessed prior to detailed design phase

- Current methodology exists to predict almost all plume effects

- For special cases, more extensive analyses (including CFD) may be required

- JSC/EG3 and LESC/Aerothermal Group have plume experience with numerous engines (ASRM, RSRM, SSME, PRCS, LM, SM, Saturn, etc.)

- Analytical tools are available to support design trade studies conducted by FLO and CLL system designers