BOEING
CAGE CODE 81205

THIS DOCUMENT IS:

CONTROLLED BY **Advanced Space Transportation**
2-1610
ALL REVISIONS TO THIS DOCUMENT SHALL BE APPROVED BY THE ABOVE ORGANIZATION PRIOR TO RELEASE

PREPARED UNDER
[X] CONTRACT NO. **NAS8-37855**
[ ] IR&D
[ ] OTHER

PREPARED ON APPLE MACINTOSH II
DOCUMENT NO. **D658-10010-1**

FILED UNDER
MODEL **STV**

TITLE **Space Transfer Vehicle Concepts and Requirements Study Phase II Final Report**

ORIGINAL RELEASE DATE 10/29/92

ISSUE NO.	TO	DATE
		9/30/92

ADDITIONAL LIMITATIONS IMPOSED ON THIS DOCUMENT WILL BE FOUND ON A SEPARATE LIMITATIONS SHEET.

	SIGNATURE	ORGN	DATE
PREPARED BY:	J. H. Cannon	2-1612	10/22/92
SUPERVISED BY:	S D Goo for D.G. Andrews	2-1612	10/27/92
APPROVED BY:	S D Goo S.D. Goo	2-1610	10/27/92

(NASA-CR-192430) SPACE TRANSFER VEHICLE CONCEPTS AND REQUIREMENTS STUDY, PHASE 2 Final Report (Boeing Aerospace Co.) 300 p

N93-16660

Unclas

G3/16 0141151

BOEING

SPACE TRANSFER VEHICLE CONCEPTS AND REQUIREMENTS STUDY

Phase II Final Report

D658-10010-1
September, 1992

DPD NUMBER-709
DR NUMBER-4
CONTRACT NAS8-37855

Submitted to
The National Aeronautics and Space Administration
George C. Marshall Space Flight Center
By
Boeing Defense & Space Group
Seattle, Washington 98124

FOREWORD

This final report of the Space Transfer Vehicle (STV) Concept and Requirements Study was prepared by Boeing for the National Aeronautics and Space Administration's George C. Marshall Space Flight Center in accordance with Contract NAS8-37855. The study was conducted under the direction of the NASA Contracting Officer Technical Representative (COTR), Mr Donald Saxton from August 1989 to November 1990, and Ms Cynthia Frost from December 1990 to April 1992.

This final report is organized into the following three documents:

Part I DESIGN GUIDE

Part 2 STV PHASE I SUMMARY

PART 3 STV PHASE 2 SUMMARY

The following personnel were key contributors during the conduct of the study in the disciplines shown:

Study Managers	Tim Vinopal & Dana Andrews
Mission & System Analysis	Bill Richards, Gary Weber, Greg Paddock, Peter Maricich
Operations	Bruce Bouton, Jim Hagen
Configurations	Richard Kolesar, Craig Hosking, George Dishman, Mike Furlong, Bob Kiliz, Jack Olson
Propulsion	Wayne Patterson, Noel Christensen, Phillip Knowles
Avionics	Rich Flannagan, Tim Mosher, Carl Malec
Structures	Peter Rimbos, Martin Gibbins
Electrical Power	Chris Johnson
Cryo Fluid Management	Ogden Jones, Jere Meserole
Mass Properties	Jeff Cannon, David Raese, Karl Heilborn
Aerothermodynamics	Richard Savage, Peter Keller
Thermal Protection	Anna Baker, Paul Nedervelt
Guidance	Lisa Skalecki, Jere Bradt
Controls	Mark Castelluccio
Performance/Astrodynamics	Ted Hanson, Ralph Risdall, Steve Paris, Mark Martin
Aerodynamics	Stan Ferguson
Crew Systems	Tom Slavin, Brand Griffin, Bill Pogue, Gerry Carr
Station Accomodations	John Palmer, Ron Rao, Carl Case
Cost Estimating	Thom Walters, Al Peffley, Hal Boggs, Jim Owens
Programmatics	Al Peffley, Don Benson, Lori Todd, Bob Croken
Documentation Support	Symantha Rodenbach, Darlene Glubrecht

For further information contact:

Cynthia Frost
NASA MSFC/PT41
MSFC, AL 35812
(205) 544-0628

Dana Andrews
Boeing Aerospace
M/S 8C-09, P. O. Box 3999
Seattle, WA 98124-2499
(206) 773-0211

BOEING

TABLE OF CONTENTS

Space Transportation Vehicle Concepts and Requirements Study Final Report

Introduction

NASA is currently studying new initiatives of space exploration involving both piloted and unpiloted missions to destinations throughout the solar system. Many of these missions require substantial improvements in launch vehicle and upper stage capabilities. The Space Transfer Vehicle (STV) Concepts and Requirements Study provides a focused examination of the space transfer vehicles required to perform these missions using the emerging national launch vehicle definition, Space Station Freedom (SSF) definition, and the latest mission scenario requirements.

This final report is a compilation of the Phase 1 and Phase 2 study findings and is intended as an STV 'users guide' rather than an exhaustive explanation of STV design details. It provides a database for design choices in the general areas of basing, reusability, propulsion, and staging; with selection criteria based on cost, performance, available infrastructure, risk, and technology.

The report is organized into the following three parts:

Part 1: Design Guide
Part 2: STV Phase 1 Concepts and Requirements Study Summary.
Part 3: STV Phase 2 Concepts and Requirements Study Summary

The overall objectives of the STV study were to: 1) define preferred STV concepts capable of accommodating future exploration missions in a cost-effective manner, 2) determine the level of technology development required to perform these missions in the most cost effective manner, and 3) develop a decision database of programmatic approaches for the development of an STV concept.

By direction of the NASA, all concepts were limited to using high performance cryogenic propellants (LO2/LH2), and the timing of the study was such, that special emphasis was given to examining the lunar exploration scenario in support of the Presidents Space Exploration Inititative. This was entirely appropriate because the lunar exploration mission with its Lunar Transportation System (LTS) provides the only near-term justification for a new upper stage vehicle.

The STV Concepts and Requirements study was conducted in two phases. Phase 1 of the study, from August 1989 to April 1991, focused on lunar and evolutionary mission performance, as well as use of the Space Station Freedom (SSF) as an assembly or refurbishment node. Phase 1 results showed that use of SSF enable use of smaller, more affordable launch vehicles, but that the additional on-orbit infrastructure and operations were very expensive, and were not cost effective for the low flight rates associated with currently proposed exploration scenarios.

Accordingly, Phase 2, from April 1991 to April 1992, focused on the use of larger launch vehicles derived from the National Launch System (NLS) family, with less emphasis placed on mission performance, more emphasis on transportation cost, and no use of the Space Station Freedom as an assembly or servicing node. Phase 2 results showed that the design-to-cost approach could save roughtly twenty percent of the total transportation Life Cycle Cost (LCC) by: 1) minimizing the number of elements developed, 2) using existing technologies were practicable, and 3) making certain program elements reusable. Unfortunately, one key issue, whether to go to the moon with a single launch of a massive booster twice the size of the Saturn V, or two launches of an NLS derived booster with on-orbit rendezvous and docking was left unresolved. Accurate facilities and ground operations cost data was not available before the conclusion of Phase 2.

1. Design Guide

In general, a transportation system delivers a payload from one point to another. In the case of a space transportation system, the purpose is to deliver payload from the Earth's surface, where it is produced, to an orbit or destination in space, and possibly return a payload or part of the system itself back to the Earth's surface. The 'best' space transportation system does this in the most efficient manner using the resources available, with the lowest cost, and with the least amount of risk to life or mission success. Figure 1.0-1 shows a summary of this design process, beginning with an objective (purpose), and limited by constraints or requirements (resources).

For example, if the goal is to establish a permanent base on the Moon, a rather large amount of oversized cargo must be delivered to the lunar surface, and people to assemble and checkout that cargo must be safely transported and returned periodically. Characteristics of the optimum transportation system design (basing location, number of stages, degree of reusability, and crew module design) for this mission objective depend on available funding, prospective infrastructure (facilities, launch vehicle, LEO node, etc.), and technology (or the willingness to pay for enhancements in these areas).

The purpose of this design guide is show the relationships between program objectives, mission requirements, and design characteristics, as derived from results of the STV Concepts and Requirements Study, Phases 1 and 2. This guide will also point out areas, such as cost and risk, that provide discrimination between design concepts, but that cannot be determined with total accuracy at this stage of design development. Ideally, this guide is meant to be used as a decision making tool to help program planners determine which LTS concepts are worth pursuing in future Phase B studies.

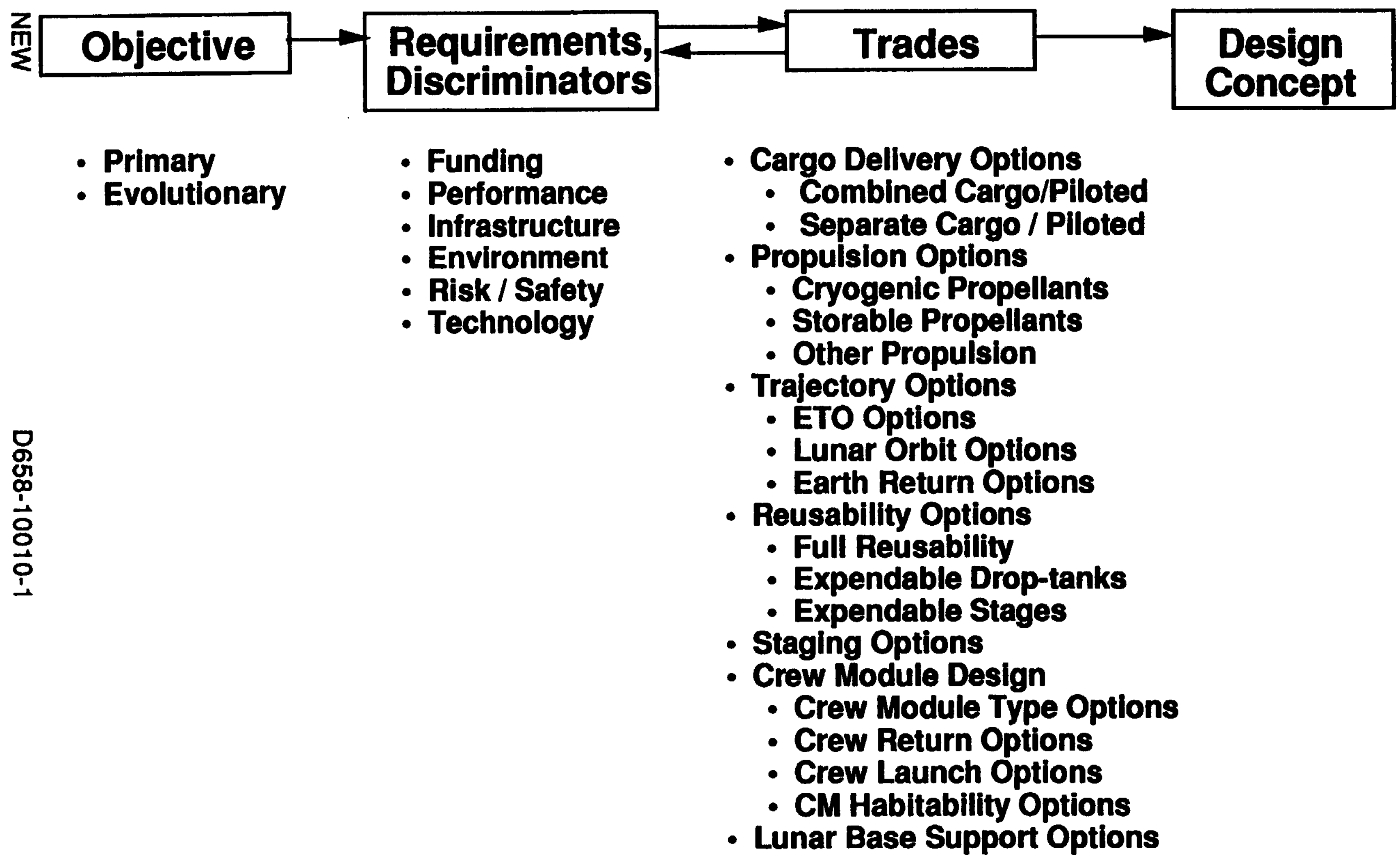


Figure 1.0-1 STV Design Process

1.1 STV Program Objective

In general, the objective for a space transportation system is to support one or more of the following missions: 1) Advancement of Science, 2) Human Exploration or Knowledge Building, 3) Expansion or Protection of the Human Habitat, and 4) Space / Planetary Resource Utilization

The STV Concepts and Requirements Study objectives are summarized as follows:

STV PRIMARY OBJECTIVE - Provide a cost effective lunar transportation system (LTS) capable of supporting a human exploration program which results in a manned outpost on the moon.

STV EVOLUTIONARY OBJECTIVES - Provide an evolvable transportation system capable of supporting high energy upper stage missions such as boosting planetary probes and delivering geosynchronous or other high orbit satellites beginning about the year 2000; and also provide the basis for an evolvable transportation system capable of supporting a human exploration program leading to a manned outpost on Mars.

1.2 Requirements / Discriminators

The requirements that influence the nature of an STV design include funding availability, mission requirements, infrastructure availability, environmental requirements, and technology availability. An additional requirement, that the system use cyrogenic propulsion, was a study groundrule imposed by the NASA. Additional analyses, done under Boeing funding, examined alternate propulsion systems and determined that LO_2/LH_2 propulsion was indeed the correct choice for the lowest cost, lowest risk lunar transfer system. Discriminators between concepts include: cost, mission performance, risk and safety, and technological advancement.

Based on the study objectives, the best STV design will support a lunar program with the lowest cost, best performance, and in the safest manner possible, while not precluding the capability of performing other missions, and while adding to the technology and infrastructure needed for going to Mars. There appear to be several concepts which meet the requirements and objectives, and do so with reasonable costs.

1.2.1 Mission Requirements / Performance

Mission performance is a measure of how well a concept meets mission requirements such as delivered payload and mission delta-velocity. In general, the best performing vehicle concept will deliver the most payload to the desired location for the least amount of mass delivered to orbit. A summary of the study mission requirements in terms of delivered payload and total required delta-velocity are shown in Figure 1.2.1-1. The missions fall into one of three categories - piloted or unpiloted lunar outpost support, unpiloted delivery, and piloted or unpiloted delivery and recovery. For the lunar outpost support missions, the overall program consists of both piloted and unpiloted missions, as shown in Figure 1.2.1-2. In this case, concept performance is not just based on piloted missions or cargo missions, but on the mixture of both piloted and unpiloted missions that meet the program objective. A typical lunar mission reference trajectory, with mission times and delta-velocities is shown in Figure 1.2.1-3.

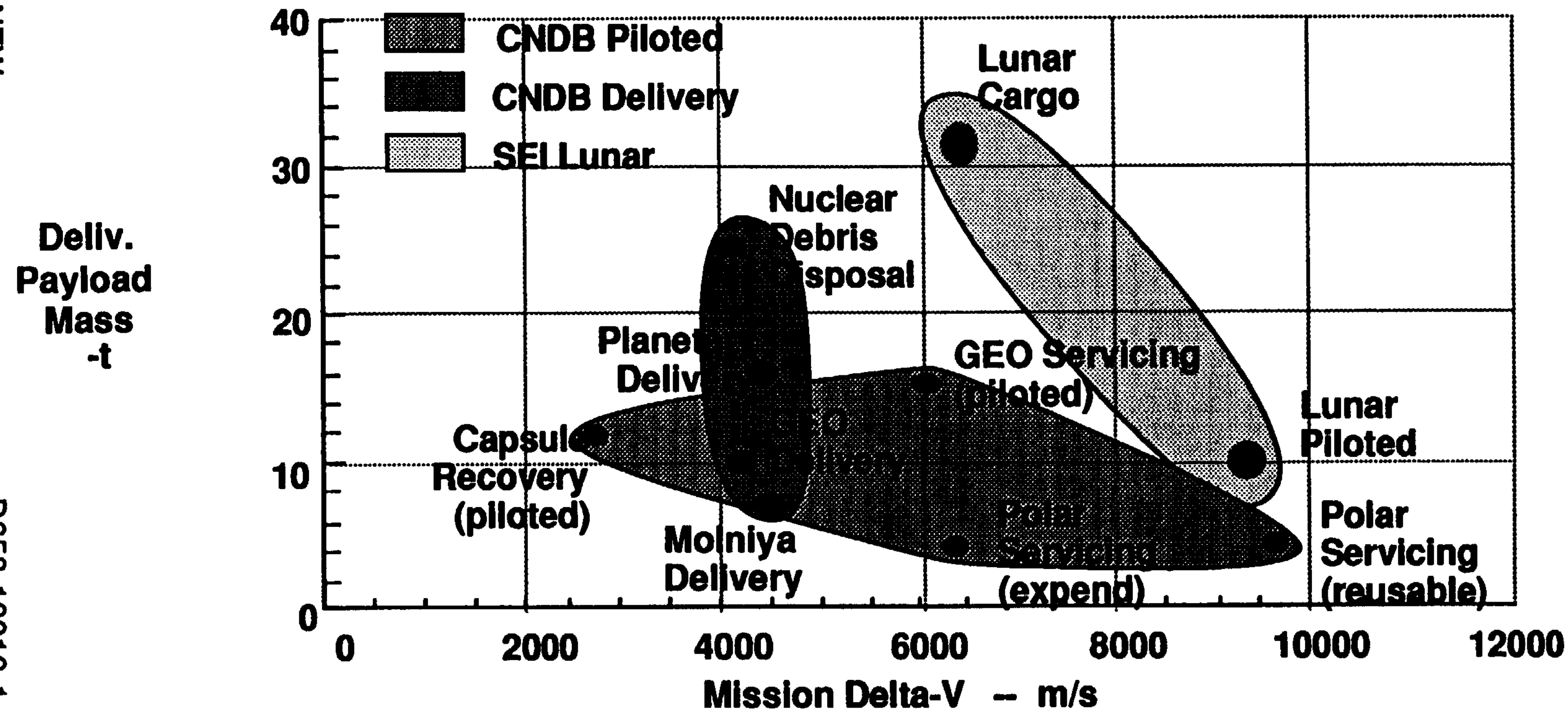
a. Lunar Mission - Lunar missions include two-way piloted and one-way cargo-only missions to the lunar surface. For the cargo missions, it is assumed that the lander remains on the lunar surface after delivery of the cargo.

b GEO Delivery Mission - Geosynchronous Earth Orbit (GEO) delivery missions are cargo-only delivery missions to GEO, with the transfer stage returned or placed into a collision-avoidance trajectory after delivery of the cargo.

c Molniya Delivery Mission - Molniya delivery missions are cargo-only delivery missions to a 12-hr Molniya orbit, with the transfer stage returned or placed into a collision-avoidance trajectory after delivery of the cargo.

d Planetary Delivery Mission - Planetary delivery missions place a planetary probe on a specified escape trajectory, after which the transfer stage is returned or placed into a collision-avoidance trajectory.

e Nuclear Disposal Mission - Nuclear disposal missions are one-way missions to retrieve and dispose of spent space reactors or debris on a trajectory to the sun.

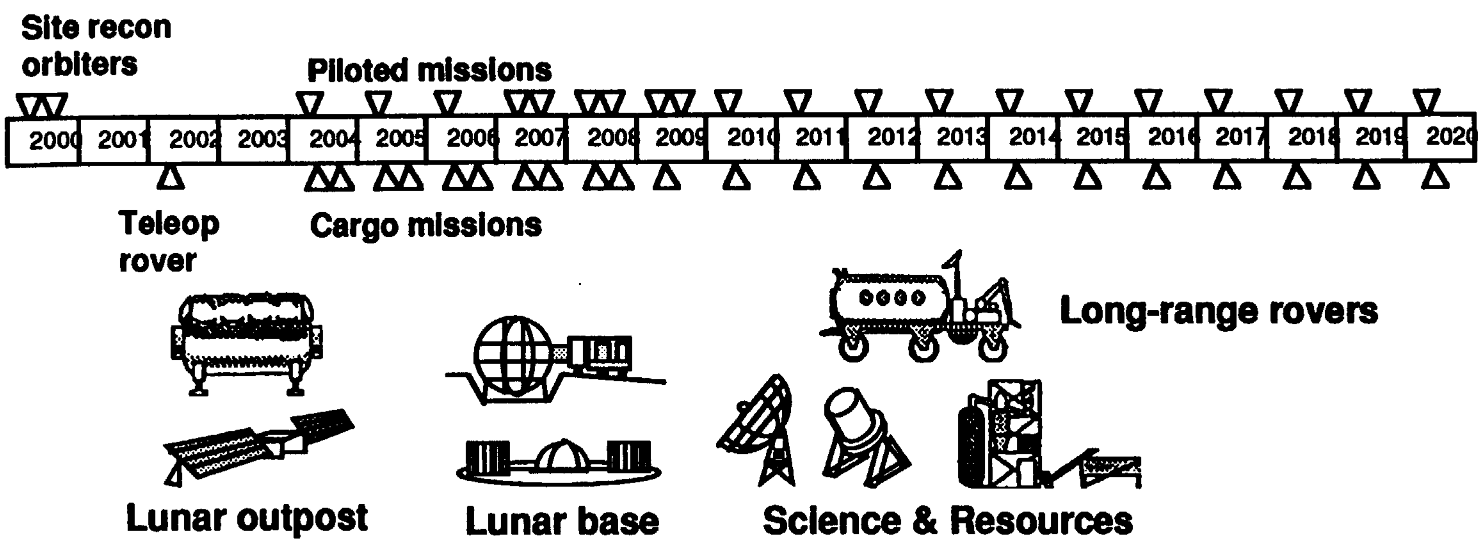


Assumptions:
- All unpiloted missions, except polar platform servicing (S1) for space-based, are delivery only.
- 1989 Civil Needs Data Base, excluding Mars missions, as follows:

	Mission	Delta V (m/s) Delivery	Delta V (m/s) Return	Deliv. Cargo (kg)	Return Cargo (kg)
N1	Nuclear Debris Disposal	4175	0	25000	0
G1	GEO Delivery	4207	0	10000	0
P1	Planetary Delivery	4451	0	16000	0
D1	Molniya Delivery	4499	0	6800	0
S1	Polar Platform Servicing (GB)	6356	0	4500	0
S1	Polar Platform Servicing (SB)	6356	3428	4500	4500
L4	Lunar Cargo Delivery	6390	0	34000	0
C1	Capsule Recovery - Piloted	2696	40	0 **	500 **
G2	GEO Servicing - Piloted	4202	1862	4000 **	4000 **
L3	Lunar Excursion - Piloted	6390	2913	12500 **	500 **

** Excluding Mass of Crew Module

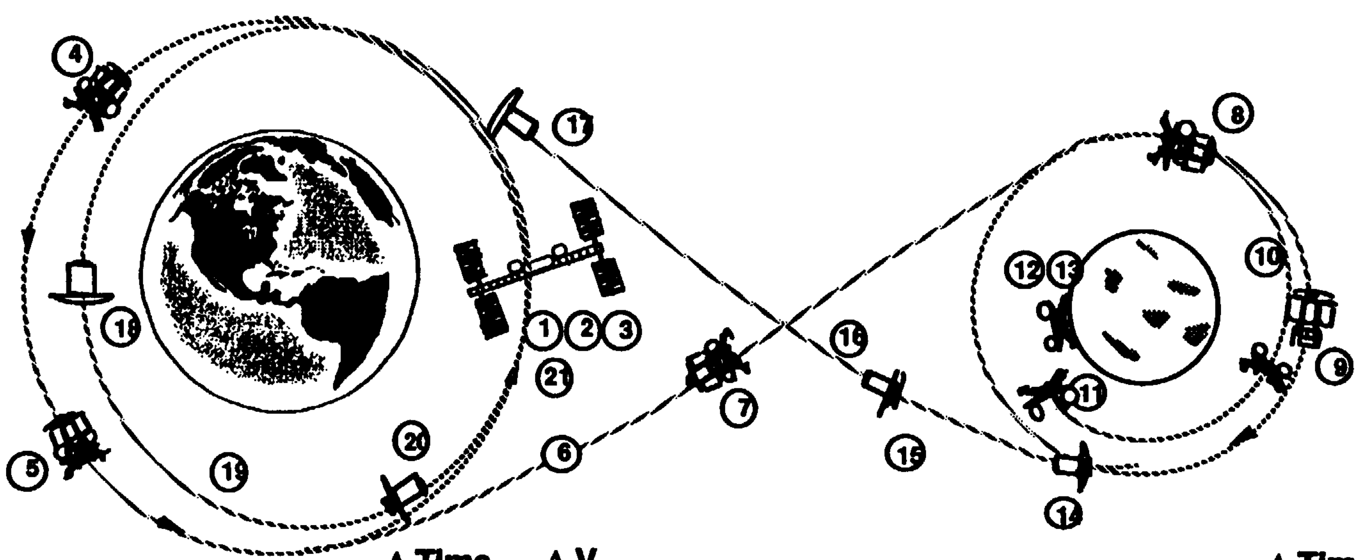
Figure 1.2.1-1 STV Mission Requirements



GO - TO - STAY PHILOSOPHY

- **418 ton cargo delivered to surface by 2020**
- **30 LTS flights (2004 - 2020)**
  - **Ground processing peaks at 4/year, averaging 1.9/year**
- **Permanently manned base initiated by 2004**
  - **Selected by 2 site recon orbiters and 1 teleop rover**

Figure 1.2.1-2 Baseline Lunar Architecture



	Event	Δ Time, hours	Δ V, mps
1)	Vehicle assembly	4 weeks	
2)	Payload Integration	2 weeks	
3)	Separate from SSF	0.0	
4)	Coast	0.8	
5)	TLI burn	0.3	3300
6)	Trajectory correction maneuvers	84.0	10
7)	Transit/coast	0.1	
8)	LOI burn	0.1	1100
9)	LEV/payload separation	0.0	
10)	LEV deorbit burn	0.05	50
11)	Descend to Lunar surface	2.0	2000

	Event	Δ Time, hours	Δ V, mps
12)	Landing	0.0	
13)	Cargo offload, Ascent	1.0	1900
14)	LTV TEI burn	0.1	1100
15)	Trajectory correction maneuvers	0.1	10
16)	Transit/coast	84.0	
17)	Pre aero correction maneuver	0.1	
18)	Aeromaneuver	0.1	
19)	Post aero correction maneuver	0.1	310
20)	Earth orbit circularization	0.1	
21)	Rendezvous/docking with SSF	1.0	45

Figure 1.2.1-3 Lunar Mission Trajectory

f Polar Servicing Mission - Polar servicing missions are two-way servicing missions to satellites in a polar orbit from the Space Station. These are not applicable for ground-based concepts

g Capsule Recovery Mission - Capsule recovery missions are two-way piloted missions to retrieve sample capsules from highly-elliptical Earth orbits.

1.2.2 Funding

The United States has entered a period of limited financial resources for space exploration. The political viability of any proposed lunar exploration program will depend in part on the amount of money that is required for development, test, and operation of an STV/LTS program. In the absence of a specified set of funding constraints, estimated program life cycle cost is used as a measure of merit for doing system and architecture trade studies.

There is a definite trade between nonrecurring and recurring costs as a function of vehicle reusability. In Phase I, we examined relative Life Cycle Costs for LTS options ranging from almost totally reusable one-and-half and two-and-a-half stage concepts to totally expendable three and four stage concepts. These trade studies were accomplished using parametric, factoring, and analogy cost estimating methods, and showed that only the manned crew module should be reused, given the low flight rates predicted. All cost comparison data was developed in constant-year 1989 or 1991 dollars.

An example of a typical unconstrained funding profile from the Apollo program, escalated to 1991 dollars using NASA cost inflation indices, is shown in Figure 1.2.2-1 The funding profile includes Apollo program S-IV stage, Command Service Module (CSM), and Lunar Module (LM) vehicle development and hardware fabrication costs. Recurring operation and support costs were not accessable by Boeing analysts during the STV study. The cost data shown for the Apollo program is one reference for the study. Review of the current NASA funding request and authorization allocation listings gives the study members another reference of cost data for comparisons of cost estimates in the STV study.

Total life cycle cost (LCC) and peak funding are of interest in evaluating STV program and design alternatives. As the system trade studies progress and better program

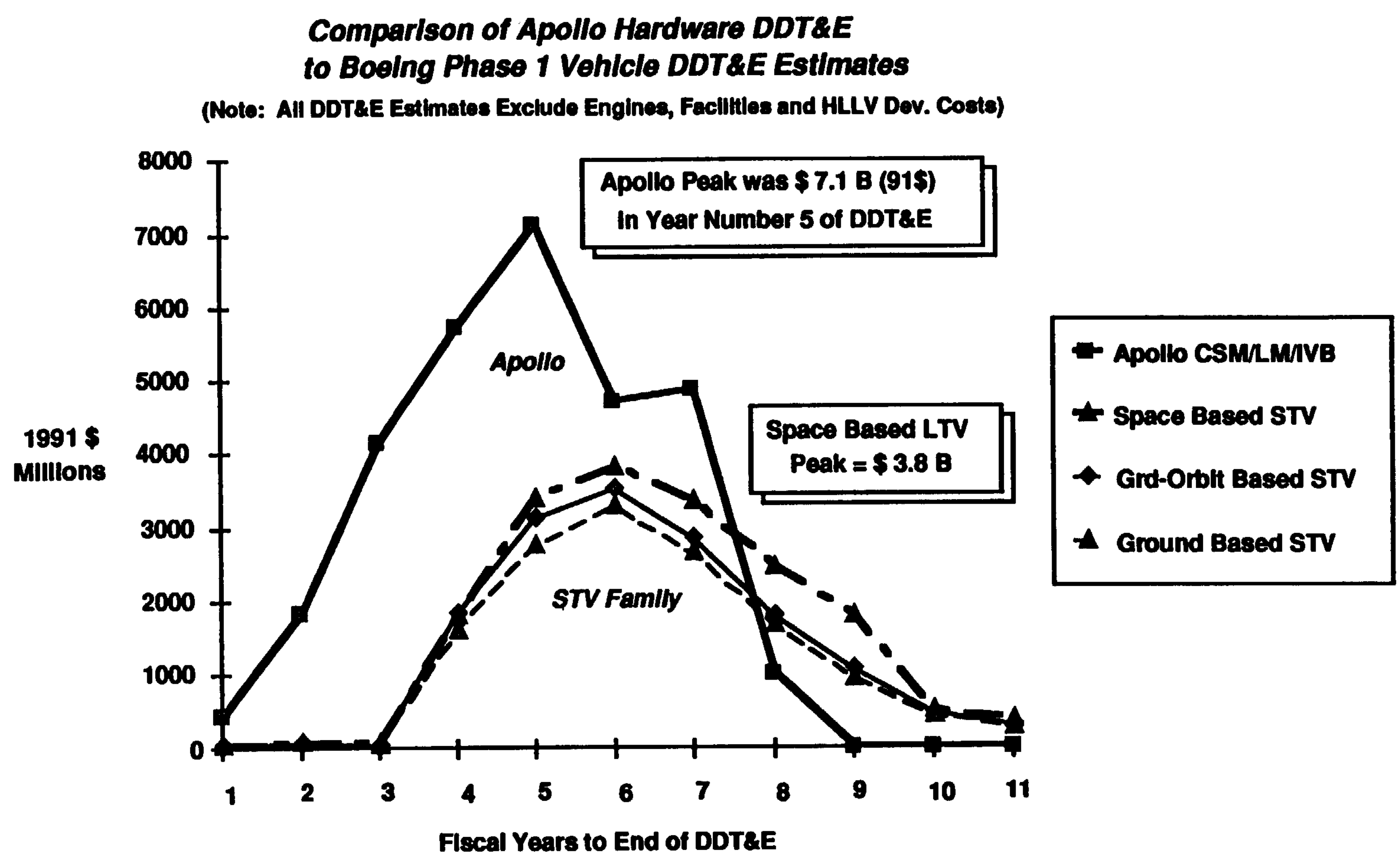


Figure 1.2.2-1 Apollo / STV DDT&E Funding Profile

planning data is developed, the peak funding profile attribute becomes more definitive and is included as part of the funding discriminator.

a. Program Non-Recurring Costs – Non-recurring development costs and production tooling costs are the "up-front" costs associated with design, development, testing, and evaluation (DDT&E) of the STV engineering tasks, flight hardware, software, and system facilities. Usually, the development cost estimates for system evaluation exercises include launch site preparation, operational facilities (including any space station operations provisions required for STV support), and some sort of "pathfinder" program where the facilities and flight hardware are certified for initial operational capability (IOC).

During the STV study these costs were estimated using several contractor parametric cost models (Boeing and Martin Marietta proprietary cost models), the NASCOM-H model (used for government estimates by NASA only), and the GE-Price cost modeling system (for some hardware and software subsystem estimates). When available, reliable hardware planning estimates were obtained from the hardware manufacturers (for example, Pratt & Whitney personnel provided most of the RL10 engine derivative estimates used for the design and system trade study estimates.)

Figure 1.2.2-2 is a typical hardware description sheet used for documenting inputs to these parametric cost models during the STV study. Notice that the majority of the paramatric cost model inputs on the sheet are information from the design and mass properties engineers. The parametric cost models are driven by the platform level (systems specification level assumptions) and the complexity and physical design descriptions of the system flight elements. The success of the whole cost estimating process is dependent upon cooperation and accurate communication between the concept designer/configurator, performance analyst, mass properties estimator, program planner, management, and the cost analyst (or estimator.)

b. Program Recurring Costs – Recurring program costs are associated with manufacturing production articles, preparing the flight hardware and crews for launching, launching the system, and performing ownership functions of the STV systems over the last three life cycle phases of the program. The three program

Boeing Cost Analysis Input Data Worksheet
(Description Sheet)

Title: STV Main Engine Propulsion - Lander Stage **Date:** 12-14-91

WBS No.: 7.2.3.2.1.3.1 (Ascent/descent eng.) **Make, Buy, or GFE?** GFE

Description of Item: Pratt & Whitney RL10 model derivative -B2 engine rated at 468.3 Isp @ 22,000 lbs. of thrust. Requires nozzle extension and retraction modifications, estimated expansion ratio of 330:1, a high reliability rating, and a high confidence level of testing for application to STV Lunar missions.

Similar Historic Items: RL10A-3-3A used on Atlas/Centaur program.

Dry Weight (pounds) 422 lbs. (w/o ctlr.) **Platform Spec. Level** Space Launch System
Volume (ft^3) 625 cu. ft. **PARAM** ______

Quantity per Assembly: Dev. Test 2 Production 2
Integration Factors (INTEGE/S) Electronics ______ Structural ______

Component Descriptors:

Mechanical/Structural

Structure Weight ______ Surface Area 166 in. long; 91 in. diam.
Manuf. Complexity Level 2 Percent New Design 20% (OTS)
Tech. Maturity 1993 Design Repeat TLI stage engines

Electrical/Electronics

Density (lbs./ft^3) ______ Technology Year ______
Type of Circuit (Digital/Neural/Analog) ______ Scale (LSI/VSLI) ______

Other Descriptors: ______ Power Requirement ______ (Watts)
Manuf. Complexity: ______ Percent New Design ______ (OTS)
Tech. Maturity ______ Design Repeat ______

Estimated Thru-Put (K)$: (FY 1991) DDT&E $ 265 M Production TFU $ 2.9 M

Manuf. Curves Selected: DDT&E 95% Production 100%

Schedule/Task Complexity:

	Development (DDT&E)		Production	
Start Date	1994		1999 (long lead)	
1st Unit Avail./Delivered	1999		2001	
Complete Date	2001		2018	
Complexity (Price Model)		(ECMPLX)		(MCPLX)
Integ. & Test Complexity	Level 6	(E/SPLANS)		
Develop Test Units Qty.	(10 in $ 265 M)			
Flight Units Qty.	4 + 1 spare	(Proto.)		
Qualification Units Qty.	2			

Figure 1.2.2-2 Example of STV Cost Analysis Input Sheet for Lunar Lander Stage Ascent Moduk

phases after DDT&E are Production, Operation and Support, and Disposal. The Disposal phase was not addressed in the STV system trade studies.

(1) Production – These recurring costs are estimated using the parametric cost models previously described. The parametric cost models are used to produce Theoretical First Unit (TFU) cost estimates. The TFU estimates are extended into full production lot buy estimates using cost improvement (learning) curves and spares and manufacturing support factors. NASA-provided program level factors are added at the life cyle cost summary level to the production estimates (in constant-year dollars.) Reusability and flight rate assumptions greatly influenced the production cost estimates in the STV studies.

(2) Operation and Support (O&S) – These recurring costs are not estimated by the traditional parametric cost modeling systems. Because of the unique characteristics of O&S tasks in the life cycle, no standardized models exist for cost estimating STV O&S phase activities in the trade studies. The O&S estimating process consisted of estimating O&S phase elements based on prior functional flow experience on Apollo and NSTS Shuttle missions, estimating relationship formulas from Kennedy Space Center databases, task manloading estimates, and preliminary hardware refurbishment and maintenance concept descriptions. The O&S estimating process success is dependent upon cooperation and acurate communication between the end user representatives, contractor system concept designer/configurator, operations analyst, program planner, management, and the cost analyst (or estimator.)

c. Life Cycle Cost Summaries – The non-recurring and recurring costs are summarized after the application of program level factors. The program level factors for the STV study cost estimates are specified by the NASA MSFC study office. The three program level factors include: a requirements change factor (30 to 35% management contingency); a prime contractor fee factor (8 to 10%); and a government program support factor (5 to 15% adminstration, analysis, and government laboratory program support.) Some of the early system cost trade studies were conducted without the application of the program level factors; however, as the range of preferred options was narrowed, the program factors were applied to the final phase I and all subsequent phase II cost estimates for STV systems. The LCC and non-recurring/recurring cost estimate relationships (magnitudes and ratio to total LCC) change as the trade studies progress. This document will attempt to describe these ratio and relative cost magnitude changes, in summary format, for each step of the STV "design for cost" trade study process.

1.2.3 Space Program Infrastructure

STV design is highly dependant on the rest of the space program infrastructure, especially the launch vehicles available, the manufacturing and operations facilities, and space node facilities if required.

a. ETO capability - In the analyses described here, the reference launch vehicle could launch 130 t to LEO, assuming a suborbital burn of the Trans-lunar Injection (TLI) stage. This approach required the payloads from two launches to rendezvous and dock in LEO prior to the TLI burn. We also analyzed several variations of the proposed large Heavy Lift Launch Vehicle (HLLV) which could place 210 t in LEO and perform the lunar mission in one launch. All launch vehicles were configured with 10 m diameter shrouds.

b. Space node facilities - Based on results in the Phase I analyses, no space assembly, servicing, or storage nodes were used in the reference missions. This includes SSF in LEO and possible storage nodes in LLO. In both cases the cost of developing the node exceeded the recurring cost savings from the improved performance, given the relatively small number of flights in the lunar exploration model (17 manned missions).

1.2.4 Environment

The physical environment encountered by the STV affects both subsystem design and vehicle configuration .

a. Subsystem Requirements - Since the STV will be manned on some missions, subsystems must be designed for two sets of requirements.

(1) Crew Requirements – In the piloted missions, provisions must be made to keep the crew alive and well for the duration of the mission, as well as provide for safe return in the case of a mission abort. Functional requirements for crew module design are shown in Figures 1.2.4-1, and 1.2.4-2, human metabolic requirements are given in Figure 1.2.4-3.

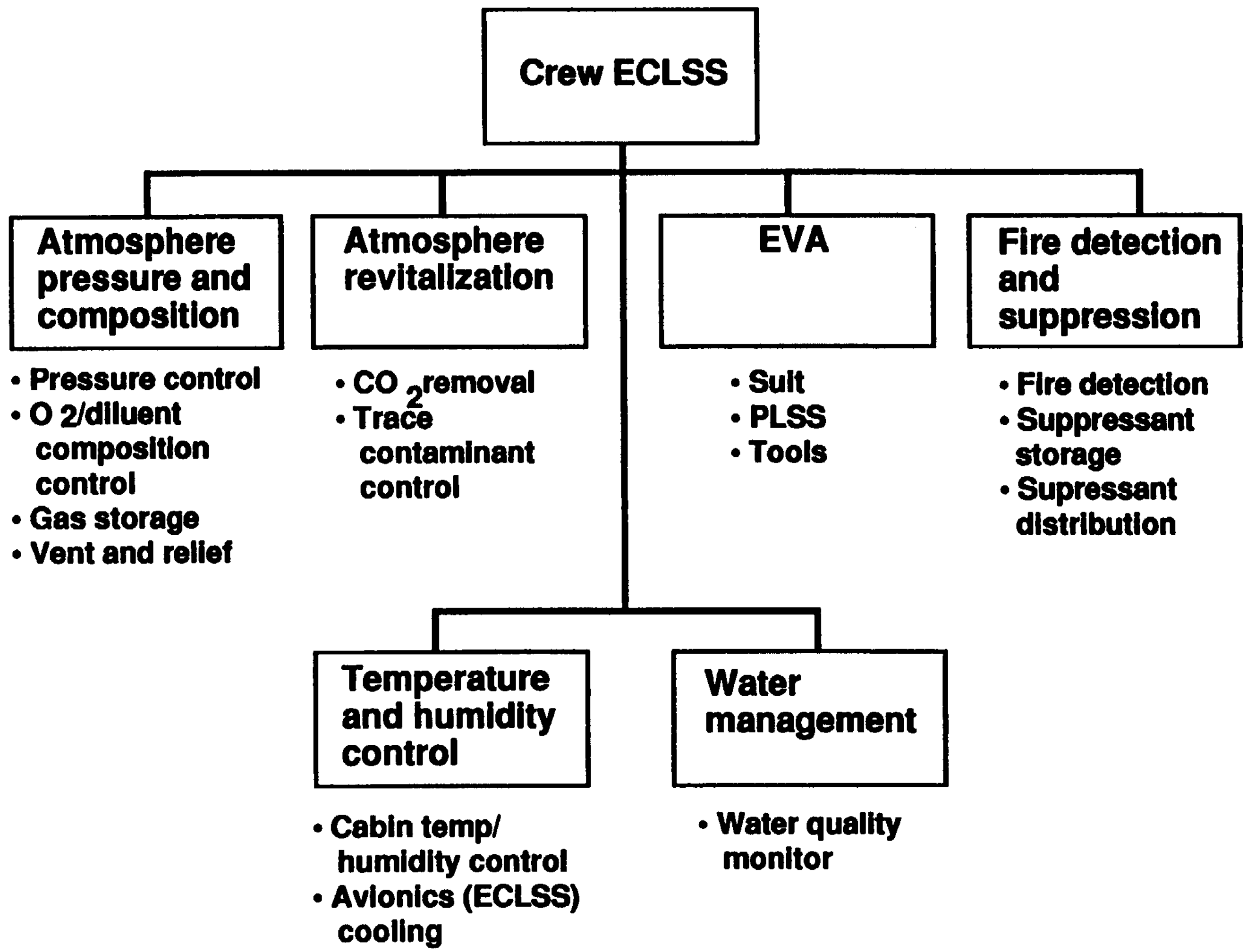


Figure 1.2.4-1 ECLSS Functional Diagram

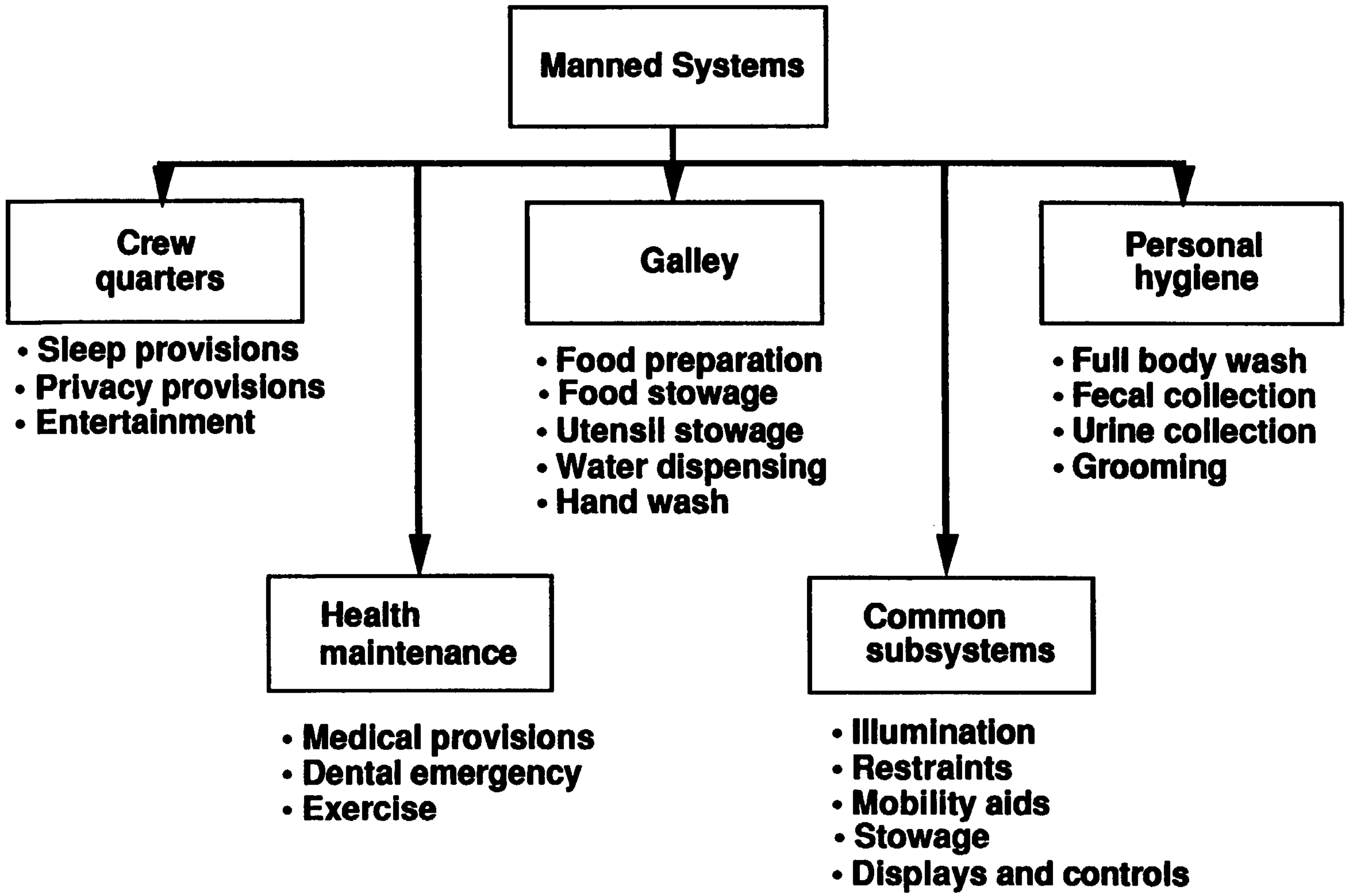


Figure 1.2.4-2 Personnel Provisions Functional Diagram

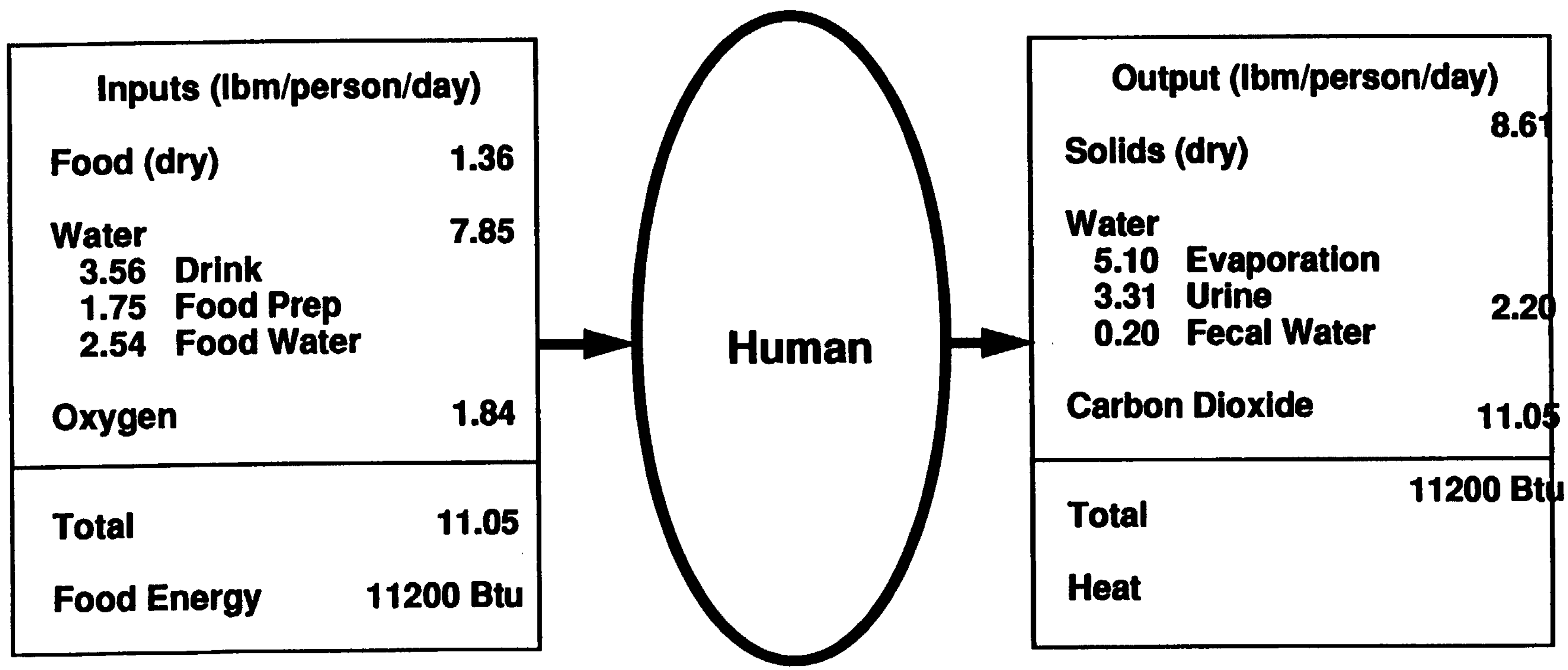


Figure 1.2.4-3 Human Metabolic Balance

(2) Vehicle Flight Requirements – Environmental concerns affecting the vehicle design include thermal environments, exposure to vacuum, exposure to lunar dust and debris, exposure to orbital debris and meteoroids, as well as exposure to solar events.

b. Configuration Constraints – Vehicle configurations are constrained by launch requirements (shroud size and launch escape provisions), engine gimbal requirements for engine-out capability, and lunar landing requirements.

(1) Shroud Size – In order to minimize impact on the launch vehicle design due to aerodynamic stability and control considerations, it is desirable to make the shroud as short and as close to the launch vehicle diameter as possible, and also to avoid extremely blunt shroud shapes. A shroud diameter of ten meters appears necessary to accommodate balanced lunar lander designs.

(2) Launch Escape Provisions – When the crew is launched aboard the STV, provisions must be made for escape from the launch vehicle in the case of a launch abort. In order to escape, the crew module must be at or near the top of the launch stack and must meet the requirements of launch escape g-levels, atmospheric deceleration, and emergency water landing.

(3) Engine gimbal requirements – Engine-out capability is one way to improve crew safety and mission success probability. To minimize required engine gimbal angles in an engine-out case, the vehicle C.G. needs to be as far as possible from the engine gimbal point. This drives the vehicle to longer, smaller diameter shapes; or to alternate control methods such as thrust balancing with throttable engines.

(4) Lunar Landing – Lunar landing requirements are opposite from the engine gimbal and launch requirements, in that it is preferable to have a short distance from the crew module to the lunar surface, with a large vehicle diameter and short c.g.-to-surface distance to minimize landing gear requirements.

c. Ground Facilities - The STV systems must also withstand the environment prior to launch. This environment includes not only the heat, humidity, and salt spray at the KSC, but also the dust, bumps, and shakes common during transit from the factory to the launch site.

1.2.5 Risk / Safety

Risk is a measurement of contributions to mission or program failure, including technical and programmatic risks.

a. Technical risk – Operational and development phase risks are driven by STV technical definition uncertainties, including mission performance and operations risks (margin reduction, failure to meet mission objective), lunar mission crew safety risks (safe abort capability), hardware reliability (redundancy, reliability), etc. Technical risk also includes the uncertainty associated with the accomplishment of STV development testing. The STV development tests must be performed to the specified (or assumed) STV operational requirements, and also to a reasonable level of confidence. The development technical design and test schedule risks are synergistic.

b. Programmatic risk – Life cycle program risks include cost, management, schedule, and other development risks. These may be expressed by a cost uncertainty analysis, a selection of programmatic risks from a maturity scale (usually from levels 1 to 10), or a combination of both methods. Both of these methods were used to some degree in the STV study. Overall scoring was accomplished using the maturity scale method for margins and risk evaluation.
A cost uncertainty model, the Boeing-proprietary Ranger model, was used at the end of phase I to evaluate the development cost risk. Ranger was used to calculate cost uncertainty ranges for several STV system design flight element finalists to provide the customer with some comparison to the NASA-supplied requirements change (management reserve) factor. The commercially-available @Risk application software tool is a viable alternative to the Ranger method and it has more flexibility to use different curve types for the system cost element distributions.

1.2.6 Technology

The relationship of technology to vehicle design is shown in Figure 1.2.6-1. The level of technology available for use by the STV system designers is a design constraint. For example, a STV design using aerobraking might have better performance than one that returns to the ground. However, the aerobraked STV configuration design effort may be delayed in development due to problems associated with aerobraking technology demonstrations or with testing to the level of confidence desired. Also the additional aerobrake development may be very costly.

Technologies may also be advanced by the STV program for use in other applications. In the scoring of technology applications for prior STV designs, the technology advancement was measured in terms of: (1) the number of technology areas expected to advance; (2) weighted by the extent of the expected advancement in those selected areas; and then (3) ranked by the criticality of those technologies to the intended STV application (enabling; enhancing). Later in the phase I study process, a NASA technology maturity scale for statusing these technologies was adopted for STV study use. The up front cost of technology and advanced development projects, the potential of payback from technology areas advanced by STV, as well as the possibility of achieving the technology level required by the program preliminary design review (PDR) date, are all concept discriminators.

a. Application to Mars Missions – Most of the technologies applicable to STV can be applied to development of a mission to Mars, as well as to other space program applications. Mars applications include those technologies that will benefit directly the development of a mission to Mars. These may include aerobraking technology, low-g propellant transfer, long-term cryogenic storage, crew module life support systems improvements, advanced guidance and landing systems, etc.

b. Other Applications – The other applications technologies category includes those technologies that contribute to other space programs like satellite servicing systems, commercial industry "free flier" space platforms, or space science applications. Examples of these other technology inprovement applications might include solid state fuel cell technology, improved vehicle health management subsystems, smart structure, neural network software, new robotics, etc.

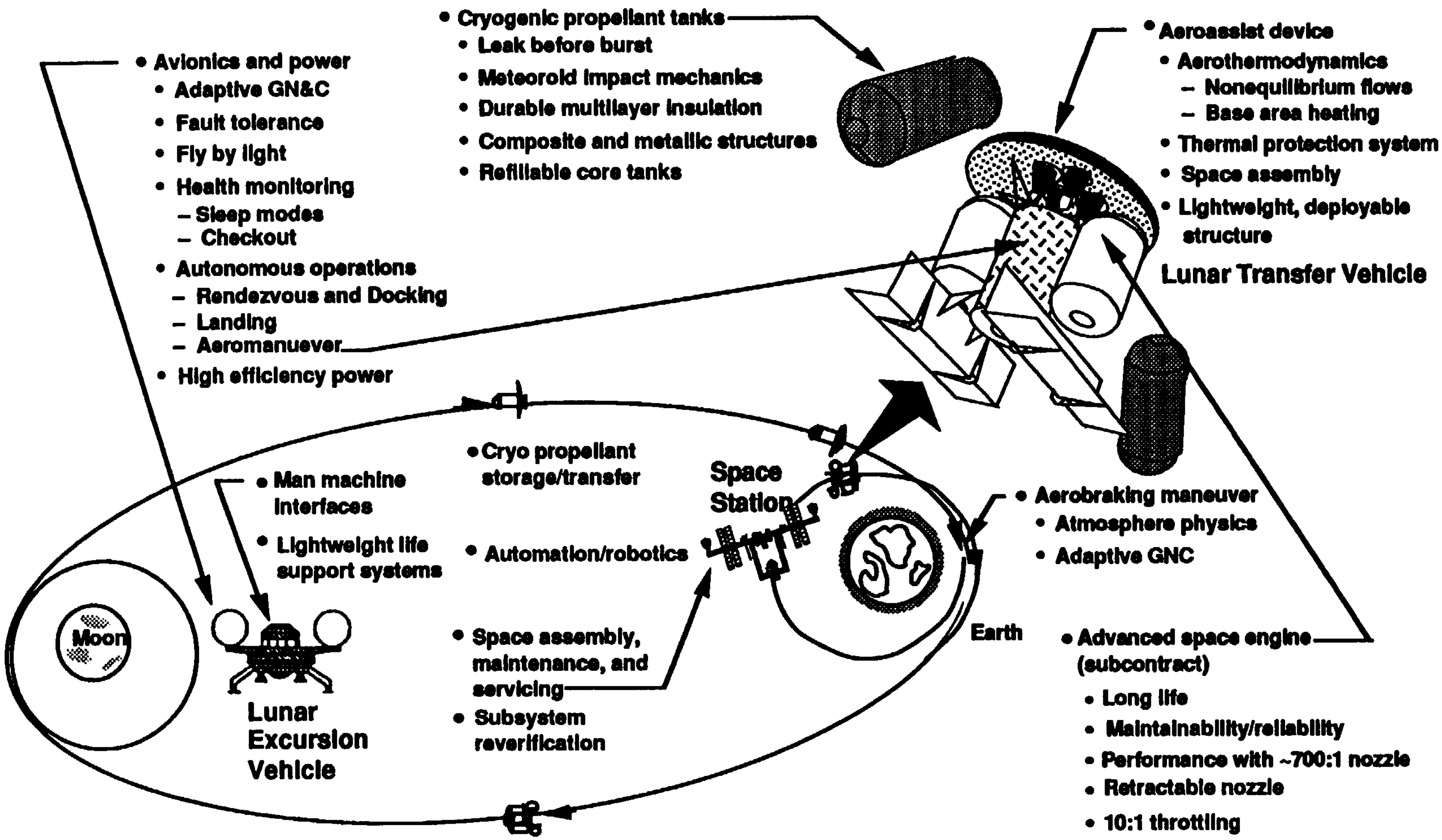


Figure 1.2.6-1 LTS Vehicle Technology Needs

c. Maturity Scale – Figure 1.2.6-2 contains the NASA maturity scale adopted by the STV technology research team (NASA MSFC, Boeing, and Martin Marietta) during Phase I of the STV study. The symbols on the scale are used for milestones on technology and advanced development schedules. The use of these symbols helped to quickly assess the status of various hardware and software subsystems within a Space Exploration Initiative (SEI) and STV technology area.

d. Critical Technologies Summary Example – A summary of critical technologies developed during phase I, associated with generic STV space-based and ground-based designs, is presented in Figure 1.2.6-3. Each of the critical technology areas was researched in some detail for the SEI Technology liaison office at MSFC. A technology priority listing (like the one shown) will change, depending upon the operational flight characteristics and integration complexities of the STV candidate system(s). See section 2.1.3 for a further discussion of the technologies associated with STV phase I designs and advanced development.

These schedule symbols were coordinated with MMC and Don Saxton:

Selected Schedule Symbols	Level	Level Description
□	1	Basic Principles Observed and Reported
◩	2	Conceptual Design Formulated
■	3	Conceptual Design Tested Analytically or Experimentally
◇	4	Critical Function/Characteristic Demonstration
⬙	5	Component Brassboard Tested In Relevant Environment
◆	6	Prototype/Engineering Model Tested In Relevant Environment
○	7	Engineering Model Tested In Space
◒		"Flight-Qualified" System
●		"Flight-Proven" System

Technology Development

Advanced Development

Flight Systems

Figure 1.2.6-2 Negotiated Maturity Level Symbols

Categories previously presented:	(Priority Selected) Space	Ground
1. Aerobrake (modeling, design, M&P)	Highest	None
2. Cryogenic Primary/Auxiliary Propulsion	Highest	Highest
3. Cryogenic Fluid Supply Systems	Highest	Highest
4. Avionics, Power, & Software	Highest	Highest

The additional categories are ranked as:

	Space	Ground
5. Structure, Tankage, Auxiliary Equip.	High	High
6. Crew Module (structures, controls, TPS)	Higher	Highest
7. Environmental Control & Life Support	Higher	Higher
8. STV Fabrication & Assy./Processing	High	Higher
9. In-Space Orbit Launch & Mission Ctrl	Highest	High
10. STV In-Space Servicing & Processing	Highest	High
11. STV Flight Operations & LES	Higher	Highest
12. In-Situ Resources → Mars Program Candidates		
13. Nuclear Propulsion → Mars Program Candidates		
14. Artificial Gravity → Mars Program Candidates		
15. Other..........	TBD	TBD

Figure 1.2.6-3 STV Technology Priority by Vehicle Type

1.3 STV/LTS Design Options

Several concept characteristics can be traded to allow an STV design to meet key design drivers.

1.3.1 Cargo Delivery Options

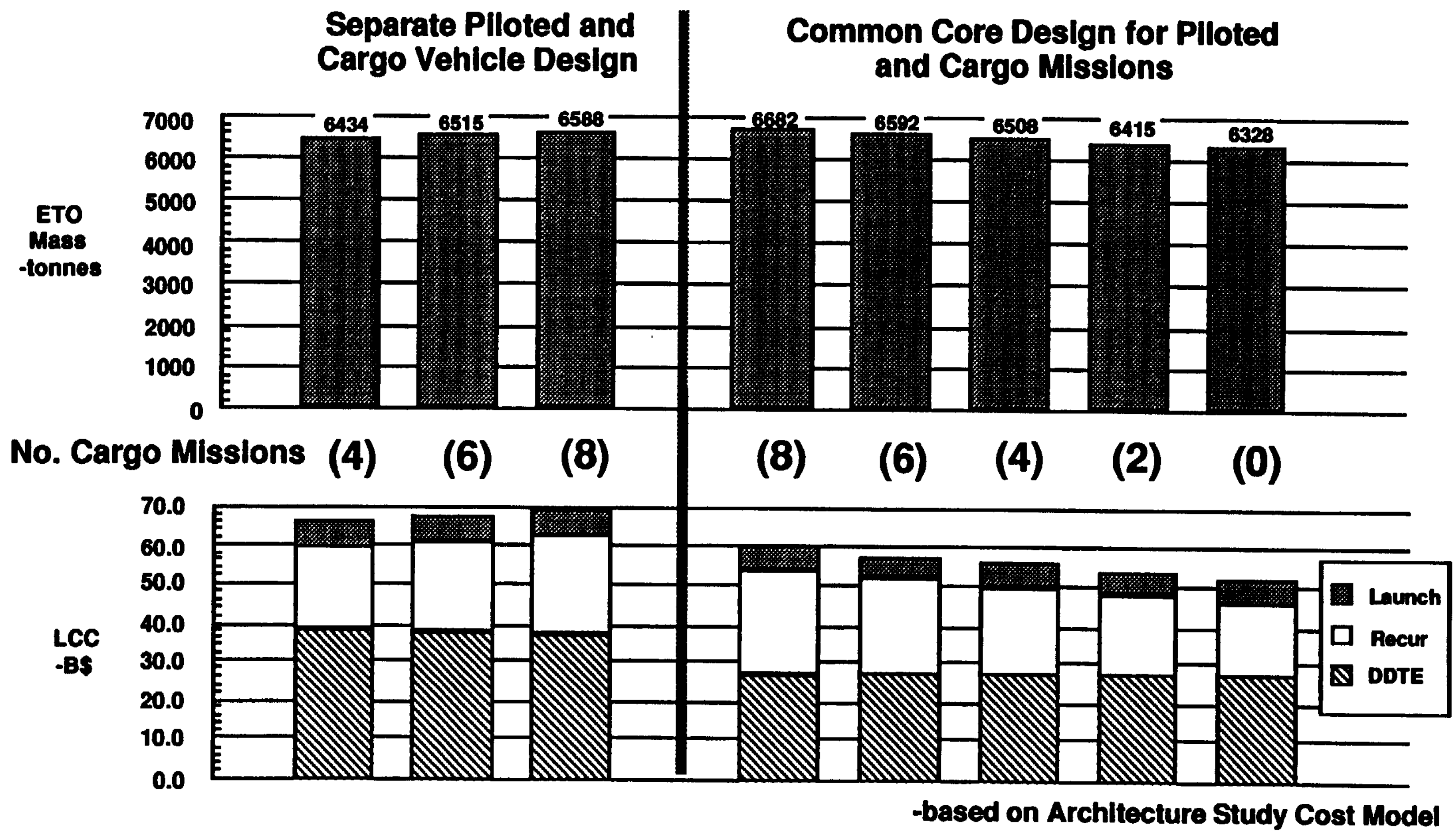
Lunar cargo delivery design options include:

(1) Separate vehicle designs for the piloted and cargo missions (small piloted vehicle and large cargo vehicle), and

(2) Common vehicle design for both piloted and cargo-only missions (optimized cargo split).

Recommendations – Although performance per mission favors an increased number of missions with smaller cargo on each mission, both overall performance and life cycle cost (LCC) favor the least number of cargo flights with a common piloted / cargo mission design, as shown in Figure 1.3.1-1.

- SB-1.5 Configuration
- For 418 t Total Cargo to Lunar Surface; 21 Piloted Missions



Common Core Design with Minimum Cargo Flights is Optimum

Figure 1.3.1-1 Cargo Delivery Option Comparison

1.3.2 Propulsion Options

Propulsion system variables that can affect performance, cost, risk, etc include main engine type and propellant (engine Isp, engine thrust level, and engine throttling capability). The requirements for chemical propulsion are given in Figure 1.3.2-1.

a. Orbit Transfer – Although this study was groundruled to use chemical propulsion systems, other engine types and propellants are available for lunar transfer, where low thrust, high Isp engines can provide significantly increased cargo delivery for the same ETO launched mass. This is effective because cargo is not as time-limited as piloted systems. Options included are shown in Figure 1.3.2-2. Use of more advanced propulsion does reduce the number of ETO launches required, but it is more economical to develop a low cost launch system and use a chemical LTS, than develop nuclear or electric vehicles and struggle with inefficient ETO launch system.

b. Lunar Landing – For lunar landing, thrust levels may be required to maintain near-hover conditions as well as lunar ascent thrust levels, so engines with some throttle capability are required, as shown in Figure 1.3.2-3.

c. Ascent / Return Propulsion – If the piloted vehicle is required to stay on the lunar surface for an extended period of time, one method of avoiding boiloff of cryogenic propellant is to use storable propellants in a separate ascent stage. The drawbacks include poorer performance due to lower engine Isp and additional propulsion systems inert weight. Some higher performance storable propellant options, such as Aluminum Hydride gel fuel / hydrogen peroxide oxidizer, are also possible, but development costs increase significantly. A comparison of the possible design options is shown in Figure 1.3.2-4, with associated performance and cost comparisons given in Figure 1.3.2-5.

d. Sensitivity to Engine Isp – A twenty second increase in the engine specific impulse of the TLI stage increases lunar cargo by 380 kilograms. A twenty second increase in in both the TLI and lander stages increases lunar cargo by 1000 kilograms as shown in Figure 1.3.2-6.

1. CRYOGENIC OXYGEN AND HYDROGEN PROPELLANTS.

2. COMPATIBLE WITH SPACE ENVIRONMENT FOR 6-MONTH LIFE.

3. SIX-MONTH LIFE WITH 10 STARTS AND 5 HOURS BURN TIME.

4. RELIABILITY CONSISTENT WITH MAN RATING.

5. TANK HEAD IDLE OPERATION FOR PROPELLANT SETTLING.

6. PROVIDE AUTOGENOUS O2 AND H2 PRESSURIZATION.

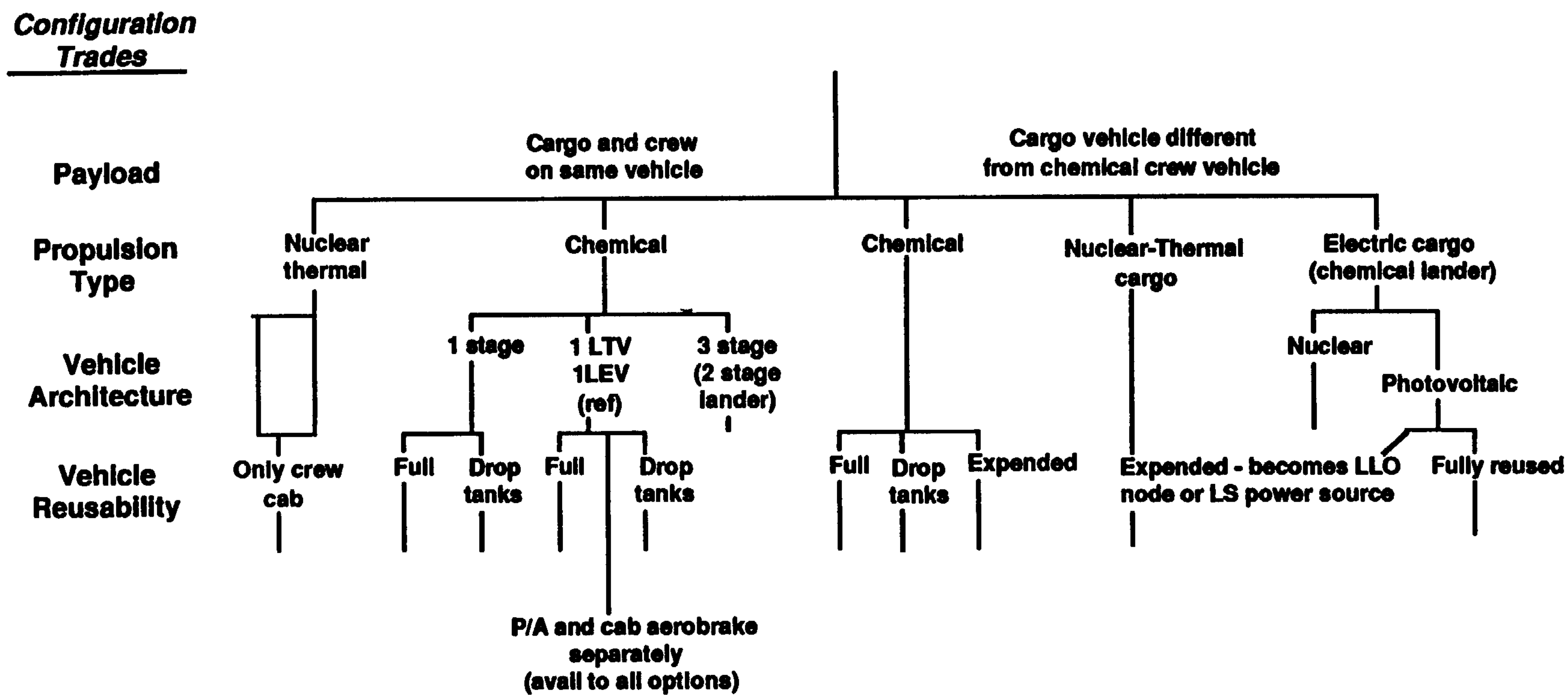
7. LUNAR EXCURSION VEHICLE (LEV) REQUIRES THROTTLING

8. THROTTLING IS NOT REQUIRED FOR THE LUNAR TRANSFER VEHICLE (LTV).

9. PREVENT FAILURE PROPAGATION TO OTHER SYSTEMS BY RETAINING FRAGMENTS OF FAILED TURBINE MACHINERY OR OTHER COMPONENTS WITHIN THE ENGINE ENVELOPE.

10. PROVIDE DATA FOR HEALTH MONITORING SUFFICIENT TO IDENTIFY INCIPIENT FAILURES AND DEFINE PREVENTATIVE REMEDIAL ACTIONS

Figure 1.3.2-1 Lunar Mission Engine Requirements



Many configuration & propulsion options are available, with different evolutionary implications.

Figure 1.3.2-2 Orbit Transfer Design Options

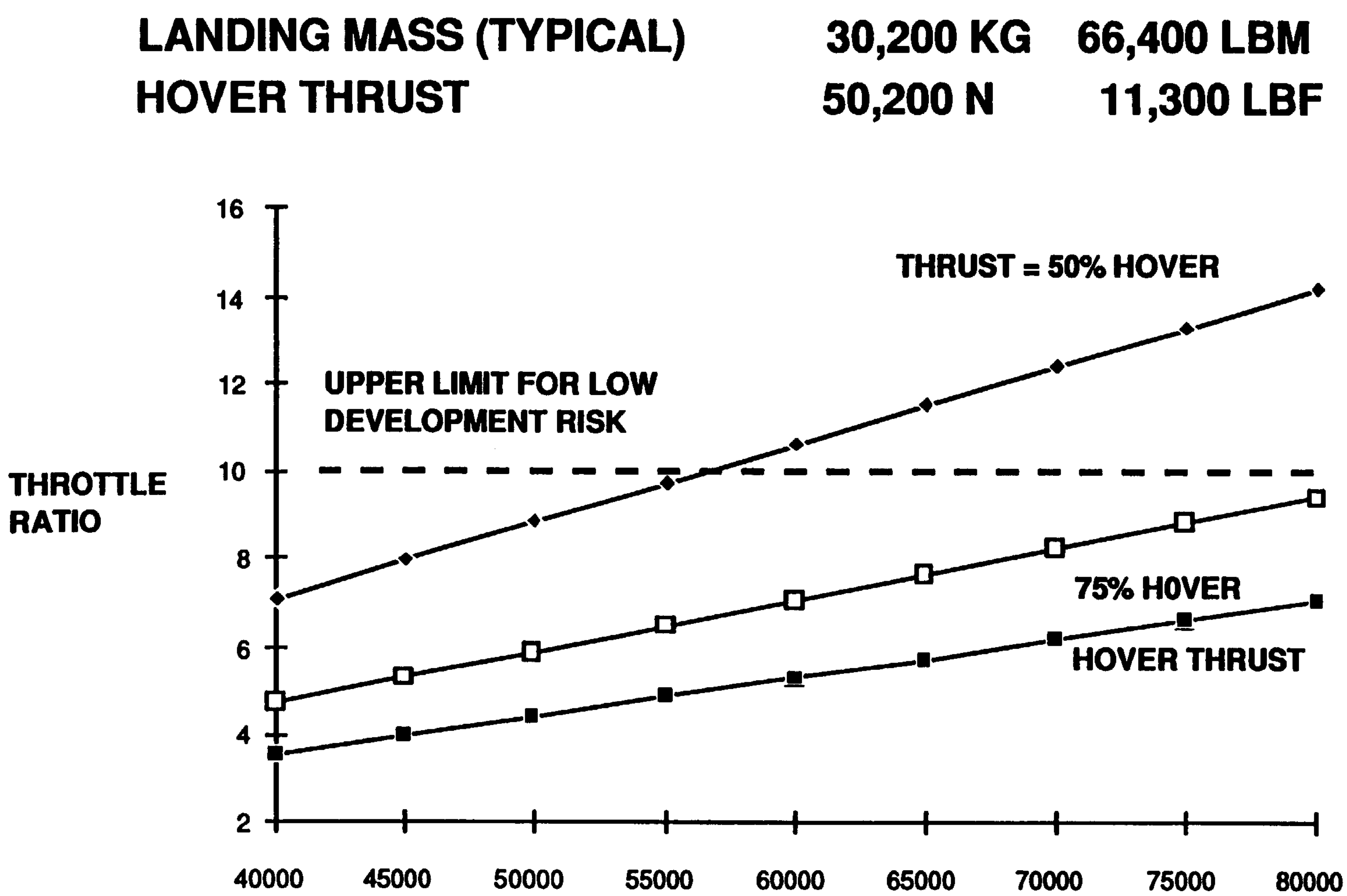
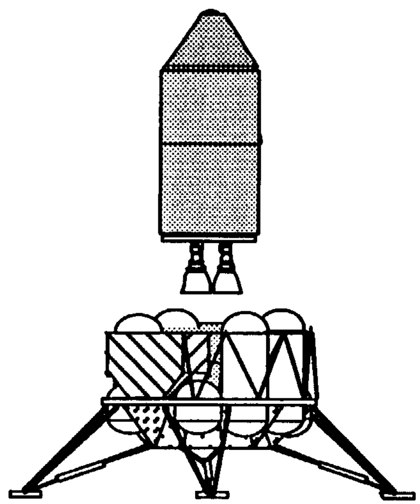


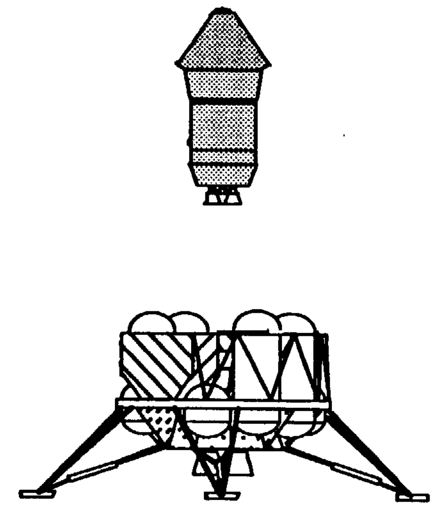
Figure 1.3.2-3 LEV Lunar Landing Propulsion

Cryo Ascent Stage Propellants



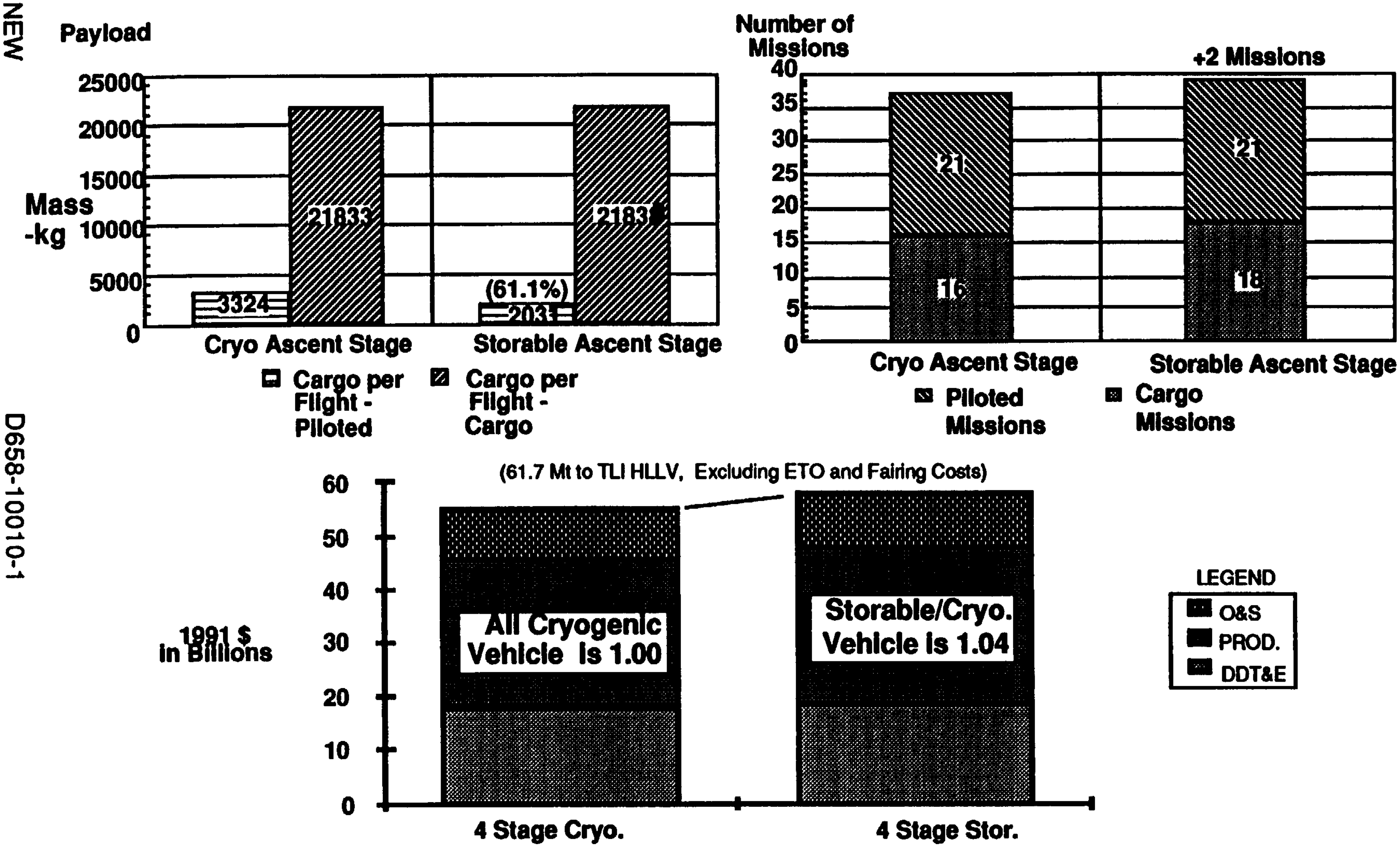
- Ascent engines: RL10 derivatives (Isp=444 sec, 15 klb thrust each)
- LO2 / LH2 ascent propellant

Storable Ascent Stage Propellants



- Ascent engines: XLR-132 derivatives (Isp=342 sec, 3750 lb thrust each)
- NTO / MMH ascent propellant

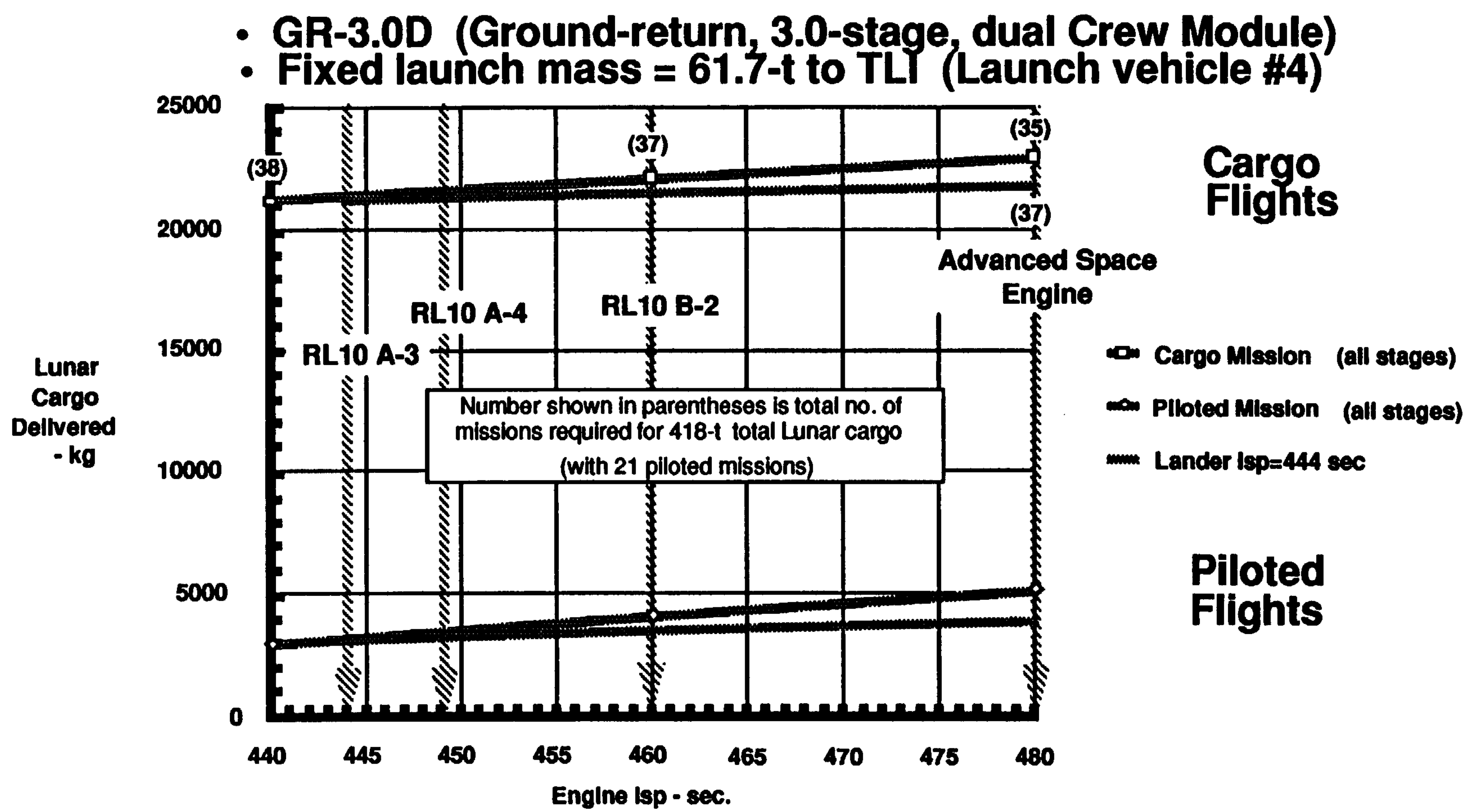
Figure 1.3.2-4 Lunar Lander Propellant Options



Aerobraked, 4.0-Stage, Dual Crew Module Vehicles

Launch Vehicle Option #4 with PoD Shroud (Capability to TLI - 61.7 mt)

Figure 1.3.2-5 Ascent Propulsion Trade Summary



A 20-sec increase in the Isp of only the transfer stage increases the lunar-delivered cargo by about 380 kg, but an increase in all stages results in a 1000 kg increase in delivered cargo.

Figure 1.3.2-6 Lunar Cargo Sensitivity to Engine Isp

1.3.3 Trajectory Options

a. Earth-to-Orbit Options – Options for the mission phase extending from launch to the start of trans-lunar injection are shown in Figure 1.3.3-1 and include single launch, dual launch with Earth orbit Rendezvous, and multiple launch with LEO assembly. A comparison of the abort availability for each option is shown in Figure 1.3.3-2.

(1) Single Launch – For this option the STV with cargo, crew module, and crew is launched complete in one launch, similar to Apollo. This option would require development of a large booster (>200 metric ton class).

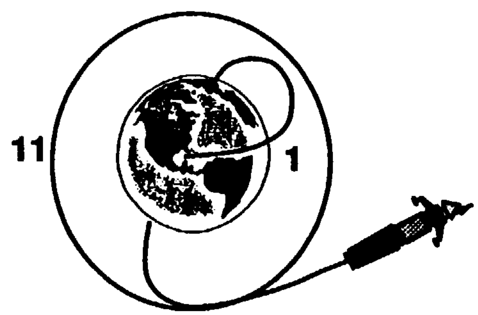
Abort opportunities are available during any phase of the launch with the use of an Apollo-type launch escape system (LES). The crew can safely escape from the launch vehicle and return to a water or land landing.

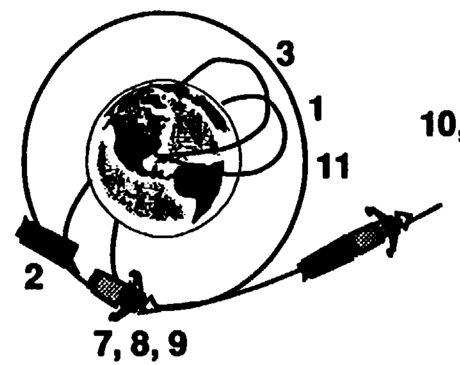
(2) On-Orbit Rendezvous – This option has elements launched separately and, through a series of rendezvous and docking maneuvers, the STV and cargo is assembled autonomously in LEO.

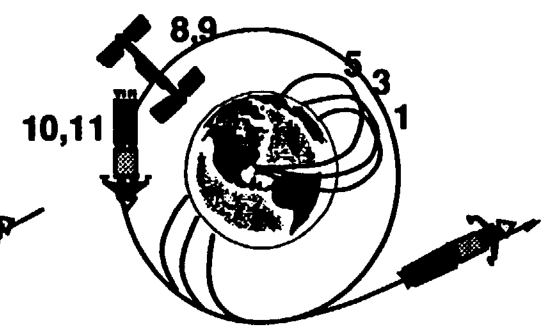
As for abort opportunities, if the crew is launched aboard a ground-return crew module with an LES, they can safely escape from the launch vehicle and return to a water landing during any phase of the launch or on-orbit operations. If launched aboard an aerobraked LEO-return crew module, an abort would require the crew to wait for an STS rescue.

(3) LEO Assembly – In this option, the vehicle, cargo, crew, and propellant or propellant tanks would be launched from Earth in multiple launches, assembled at a LEO node (SSF-assumed), and then depart from the LEO node.

Abort opportunities for this launch option are similar to the dual-launch scenario. In this case though, the LEO node could serve as a safe haven for the crew while awaiting an STS rescue launch.





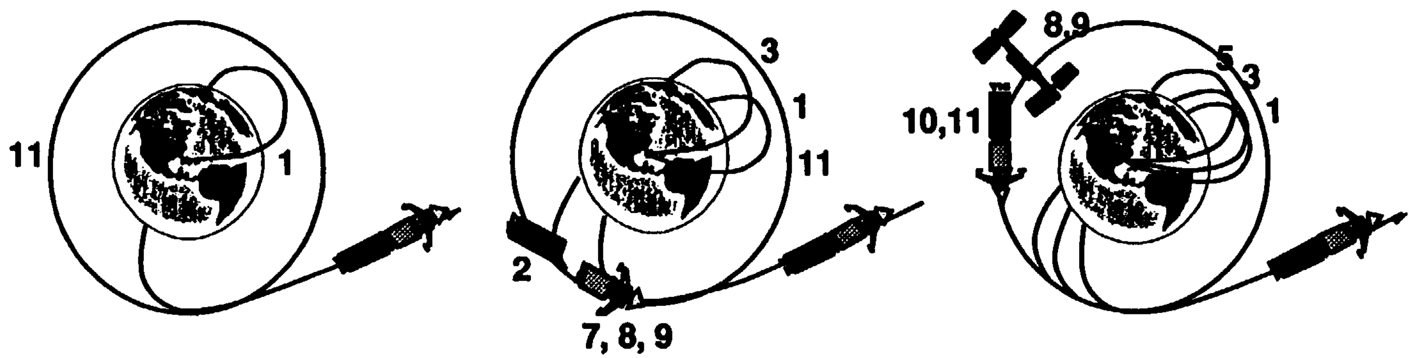


Earth to Orbit

	Event *	Single-Launch ∂-V (m/s)	Single-Launch ∂-T (hrs)	Single-Launch Enabling Technologies	Dual Launch, LEO Rendezvous ∂-V (m/s)	Dual Launch, LEO Rendezvous ∂-T (hrs)	Dual Launch, LEO Rendezvous Enabling Technologies	Multiple-Launch, LEO Assembly ∂-V (m/s)	Multiple-Launch, LEO Assembly ∂-T (hrs)	Multiple-Launch, LEO Assembly Enabling Technologies
1	HLV Launch #1	*2230*	0.2	Launch Escape	*2230*	0.2	none	*2230*	0.2	none
2	LEO Storage	0	0.0	n/a	0	624.0	Cryo Storage	0	624.0	Cryo Storage
3	HLV Launch #2	0	0.0	n/a	*1340*	0.2	Launch Escape	*1340*	0.2	none
4	LEO Storage	0	0.0	n/a	0	0.0	n/a	0	624.0	Cryo Storage
5	STS Launch	0	0.0	n/a	0	0.0	n/a	0	0.2	n/a
6	LEO Turn & Dock	*0*	*6.0*	*Rendezvous*	0	0.0	n/a	0	0.0	n/a
7	LEO Prox-ops	0	0.0	n/a	0	6.0	Rendezvous	0	6.0	Rendezvous
8	LEO Assembly	0	0.0	n/a	0	72.0	n/a	0	72.0	Teleop assy
9	LEO Checkout	0	0.0	n/a	0	96.0	VHMI	0	297.0	VHMI
10	LEO Node Sep	0	0.0	n/a	0	0.0	n/a	0	6.0	n/a
11	LEO Coast	0	4.5	none	0	4.5	none	0	4.5	none
		2230	10.7		3570	802.9		3570	1634.1	

* Optional ops in italics

Figure 1.3.3-1 ETO Options



Earth to Orbit

	Event *	Single-Launch: Abort Capability	Single-Launch: Return Time (hrs)	Dual Launch, LEO Rendezvous: Abort Capability	Dual Launch, LEO Rendezvous: Return Time (hrs)	Multiple-Launch, LEO Assembly: Abort Capability	Multiple-Launch, LEO Assembly: Return Time (hrs)
1	HLV Launch #1	LES, water recovery	<1	n/a	n/a	n/a	n/a
2	LEO Storage	n/a	n/a	n/a	n/a	n/a	n/a
3	HLV Launch #2	n/a	n/a	LES, water recovery	<1	n/a	n/a
4	LEO Storage	n/a	n/a	n/a	n/a	n/a	n/a
5	STS Launch	n/a	n/a	n/a	n/a	STS abort	<1
6	LEO Turn & Dock	*deorbit, water recovery*	*<5*	n/a	n/a	n/a	n/a
7	LEO Prox-ops	n/a	n/a	deorbit, water recovery	<5	STS abort	<1
8	LEO Assembly	n/a	n/a	n/a	n/a	SSF/STS return	1-144**
9	LEO Checkout	n/a	n/a	deorbit, water recovery	<5	SSF/STS return	1-144**
10	LEO Node Sep	n/a	n/a	n/a	n/a	n/a	n/a
11	LEO Coast	deorbit, water recovery	<5	deorbit, water recovery	<5	SSF/STS return	1-144**

* Optional ops in italics

** Depends on STS or PLS Availability

Figure 1.3.3-2 ETO Abort Options

(4) Recommendations – A summary of the Phase 1 findings concerning ETO options is shown in Figure 1.3.3-3. The single-launch and dual-launch LEO-rendezvous scenarios had the least risk and lowest cost, respectively, with overall scores favoring the dual-launch LEO-rendezvous option. The single-launch and dual-launch options were studied further in Phase 2.

b. Lunar Insertion / Landing – Options for the mission phase extending from trans-lunar injection to Earth orbit insertion are shown in Figure 1.3.3-4 and include Lunar Surface Direct, Lunar Orbit Direct (LOD), and Lunar Orbit Rendezvous (LOR). A comparison of the abort availability for each option is shown in Figure 1.3.3-5.

(1) Lunar Surface Direct – The direct approach is a single burn approach where the landing site is targeted and the STV performs a single landing burn. In this case, a safe abort could be accomplished up to the lunar targeting or insertion burn, which puts the vehicle onto an impact trajectory. This makes this option less safe for the crew than the other options, but could be a viable option for a cargo-only mission.

(2) Lunar Orbit Direct – This approach was conceived during evaluations of the lunar surface direct option to mitigate some of the safety concerns related to the lunar surface direct approach. In this scenario, the STV inserts into an elliptical lunar orbit and then, without leaving anything in orbit, performs a landing burn. The approach assumed, would be to burn into the transfer orbit, stay in this orbit for only a portion of a revolution, and then accomplish the lunar landing. The use of a fractional orbit may be ambitious in terms of navigational capability, so the option exists to stay in this elliptical orbit for some number of revolutions prior to landing. This would initially provide time for navigation updates while providing a growth path to the fractional orbit approach as navigation capabilities are verified.

(3) Lunar Orbit Rendezvous – The LOR approach was used for the Apollo missions. Depending on the vehicle concept, Earth-to-LLO transfer and/or return elements may be left in a LLO parking orbit while the lunar surface tasks are performed. Upon completion of the lunar surface stay, the lunar excursion portion of the STV would rendezvous and dock with the elements

Assumptions

- Ground based single launch scenarios incur $7 B delta for DDT&E of large 220 mt class booster
- Space based concepts incur $4.5 B (GD STIS) for SSF facilities and changes
- Reusable elements used for a total of 5 missions
  - Ground based - Crew module only
  - Ground based, LEO Assembly - Crew module only
  - Combination Space/Ground based - Earth return Crew mod. and Space based stage (less drop tanks)
  - Space based - Crew module and stage (less drop tanks)
- Trade criteria (weighting): Cost (50%), Margins & Risk (30%), Other Mission Capture (15%), and Benefits to Mars (5%)
- Trade study based on Lunar Missions only

Trade Scoring

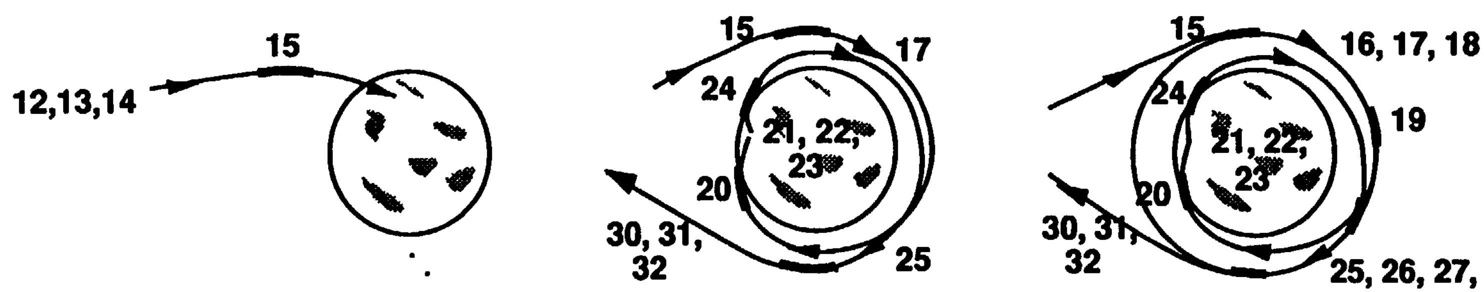
BASING	COST	M&R	MSSN CAP	MARS	TOTAL
Ground	2.37	1.56	2.82	4.55	1.74
Grnd-LEO Rend & Dock Assembly	1.67	2.34	2.78	3.71	1.51
Space-Stage Grnd-Crew Module	2.23	4.09	2.82	1.34	2.49
Space	2.12	4.41	3.00	1.52	2.60

- Scoring based on 1-5 scale with 1 being best
- Total scores (.5 Cost + .3 M&R, etc.) respread to 1-5 scale
- Cost score is 70% DDT&E + 30% LCC, based on Low boost cost ($1K/kg to LEO)

Conclusions

- Cost favors Ground basing with LEO assembly and Space basing (Space Station facilities costs will be further defined)
- Margins & Risk and Benefits to Mars criteria scores tend in the opposite direction
- Space basing, Ground based single launch, and Ground based with on-orbit assembly have been selected for further study
  - Indicated by overall scenario scoring

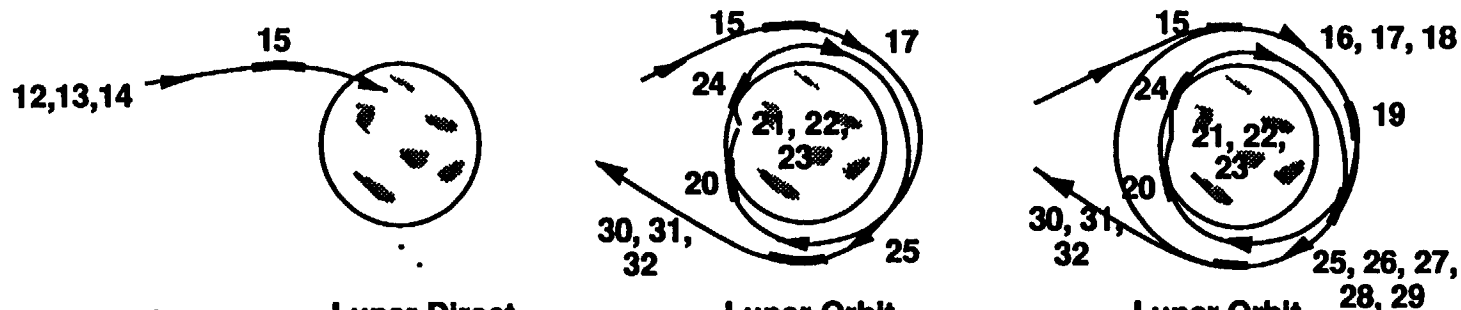
Figure 1.3.3-3 Phase 1 Basing / Reusability Trade Summary



	Lunar Transfer	Lunar Direct (Cargo Only)			Lunar Orbit Direct			Lunar Orbit Rendezvous		
	Event *	∂-V (m/s)	∂-T (hrs)	Enabling Technologies	∂-V (m/s)	∂-T (hrs)	Enabling Technologies	∂-V (m/s)	∂-T (hrs)	Enabling Technologies
12	Trans-lunar Inject	3277	0.2	none	3150	0.2	none	3150	0.2	none
13	Trans-lunar Coast	0	84.0	none	0	84.0	none	0	84.0	none
14	Correction	0	0.1	none	95	0.1	none	37	0.1	none
15	Lunar Orbit Insert	1075	0.1	none	1075	0.1	none	914	0.1	none
16	LLO Rendezvous	0	0.0	n/a	0	0.0	n/a	5	26.0	Rendezvous
17	LLO- Coast	0	0.0	n/a	0	0.0	n/a	0	116.0	none
18	Refuel Lander	0	0.0	n/a	0	0.0	n/a	*0*	*46.0*	*Low-g Transfer*
19	De-orbit, Brake	1367	0.8	none	1428	0.8	none	1642	1.8	none
20	Final Descent	492	0.2	Auto. Landing	492	0.2	Eng. Throttle	492	0.2	Eng. Throttle
21	LS-Crew In Cab				0	48.0	none	0	48.0	none
22	LS-Storage				0	4272.0	Cryo Storage	0	4272.0	Cryo Storage
23	LS- Checkout				0	8.0	VHMI	0	8.0	VHMI
24	Ascent				1822	0.2	Eng. Restart	1857	0.2	Eng. Restart
25	LLO Coast				0	0.8	none	0	0.8	none
26	LLO Rendezvous				0	0.0	n/a	48	2.0	Rendezvous
27	Refuel TEI Stage				0	0.0	n/a	*0*	*16.0*	*Low-g Transfer*
28	LLO Operations				0	0.0	n/a	0	20.0	none
29	LLO Coast- Abort				0	0.0	n/a	*0*	*144.0*	*none*
30	Trans-Earth Inject				1075	0.2	none	967	0.2	none
31	Trans-Earth Coast				0	72.0	none	0	72.0	none
32	Correction				16	12.0	none	3	12.0	none
		6211	85.4		9153	4498.6		9115	4869.6	

* Optional ops in italics

Figure 1.3.3-4 Lunar Orbit Options



	Lunar Transfer	Lunar Direct (Cargo Only)	Lunar Orbit Direct		Lunar Orbit Rendezvous	
	Event *	Abort Capability	Abort Capability	Return Time (hr)	Abort Capability	Return Time (hr)
12	Trans-lunar Inject	n/a	Free-return	84 - 804 **	Free-return	84 - 804 **
13	Trans-lunar Coast	n/a	Free-return	84 - 804 **	Free-return	84 - 804 **
14	Correction	n/a	Free-return	84 - 804 **	Free-return	84 - 804 **
15	Lunar Orbit Insert	n/a	TEI burn	84 - 804 **	TEI burn	84 - 804 **
16	LLO Rendezvous	n/a	n/a	84 - 804 **	Separate, TEI	84 - 804 **
17	LLO- Coast	n/a	n/a	84 - 804 **	Separate, TEI	84 - 804 **
18	Refuel Lander	n/a	n/a	84 - 804 **	Separate, TEI	84 - 804 **
19	De-orbit, Brake	n/a	TEI burn	84 - 804 **	Return to LLO	84 - 804 **
20	Final Descent	n/a	Ascent, TEI burn	84 - 804 **	Return to LLO	84 - 804 **
21	LS-Crew In Cab		Ascent, TEI burn	93 - 813 **	Return to LLO	131 - 851 **
22	LS-Storage		Ascent, TEI burn	93 - 813 **	Return to LLO	131 - 851 **
23	LS- Checkout		Ascent, TEI burn	93 - 813 **	Return to LLO	123 - 843 **
24	Ascent		Ascent, TEI burn	84 - 804 **	Return to LLO	122 - 842 **
25	LLO Coast		TEI burn	84 - 804 **	Return to LLO	120 - 840 **
26	LLO Rendezvous		n/a	84 - 804 **	Separate, TEI	120 - 840 **
27	Refuel TEI Stage		n/a	84 - 804 **	Separate, TEI	104 - 824 **
28	LLO Operations		n/a	84 - 804 **	Separate, TEI	84 - 804 **
29	LLO Coast- Abort		n/a	84 - 804 **	*Wait, TEI burn*	84 - 804 **
30	Trans-Earth Inject		TEI burn	84 - 804 **	TEI burn	84 - 804 **
31	Trans-Earth Coast		n/a	n/a	n/a	n/a
32	Correction		n/a	n/a	n/a	n/a

* Optional ops in italics

** Up to 720-hr STS/PLS Launch Delay or 144-hr SSF plane alignment delay

Figure 1.3.3-5 Lunar Orbit Abort Options stored in LLO and the return to Earth would be initiated. Between missions, some flight elements (e.g., LLO-based excursion stages and/or excursion crew modules) may be left in LLO to be refueled and used for the next lunar excursion. LOR abort issues are discussed in Figures 1.3.3-6 thru 1.3.3-9.

(4) Recommendations – A summary of the Phase 1 findings concerning lunar orbit options is shown in Figure 1.3.3-10. The facts are that LOR provides additional performance, but at unacceptable risk. This was not the case for the Apollo missions, because those missions all landed near the lunar equator, where there are frequent opportunities to launch and rendezvous with the TEI stage. Future missions will visit sites well off the equator, where rendezvous opportunities require almost impossible ΔVs for days to weeks at a time. This, combined with the cost of developing a TEI stage capable of autonomous operation in LLO for months at a time, led us to recommend the lunar orbit direct approach for manned lunar exploration. The potential savings of thirty to forty tons of launch mass with LOR are not worth the additional program cost and the risk of stranding astronauts on the lunar surface during an emergency.

c. Earth Return Options – Options for the return to Earth mission phase are shown in Figure 1.3.3-11 and include direct reentry to the ground, aerobraked return to LEO with STS recovery, aerobraked return to LEO with LEO node storage, and all-propulsive return to LEO. A comparison of the abort availability for each option is shown in Figure 1.3.3-12.

(1) Direct reentry to ground – In this option the crew module returns in a guided trajectory to the launch site for refurbishment and reuse. The method of abort during this mission phase is an abort to a water landing in case of partial parachute or guidance system failure.

(2) Aerobraked return to LEO, STS recovery – This option uses an expendable aerobrake to slow down by braking through the Earth's atmosphere. After achieving Earth orbit capture, the aerobrake is jettisoned (too large to fit in the shuttle payload bay), and a propulsive maneuver of approximately 310 m/s ΔV is required to circularize into a rendezvous orbit

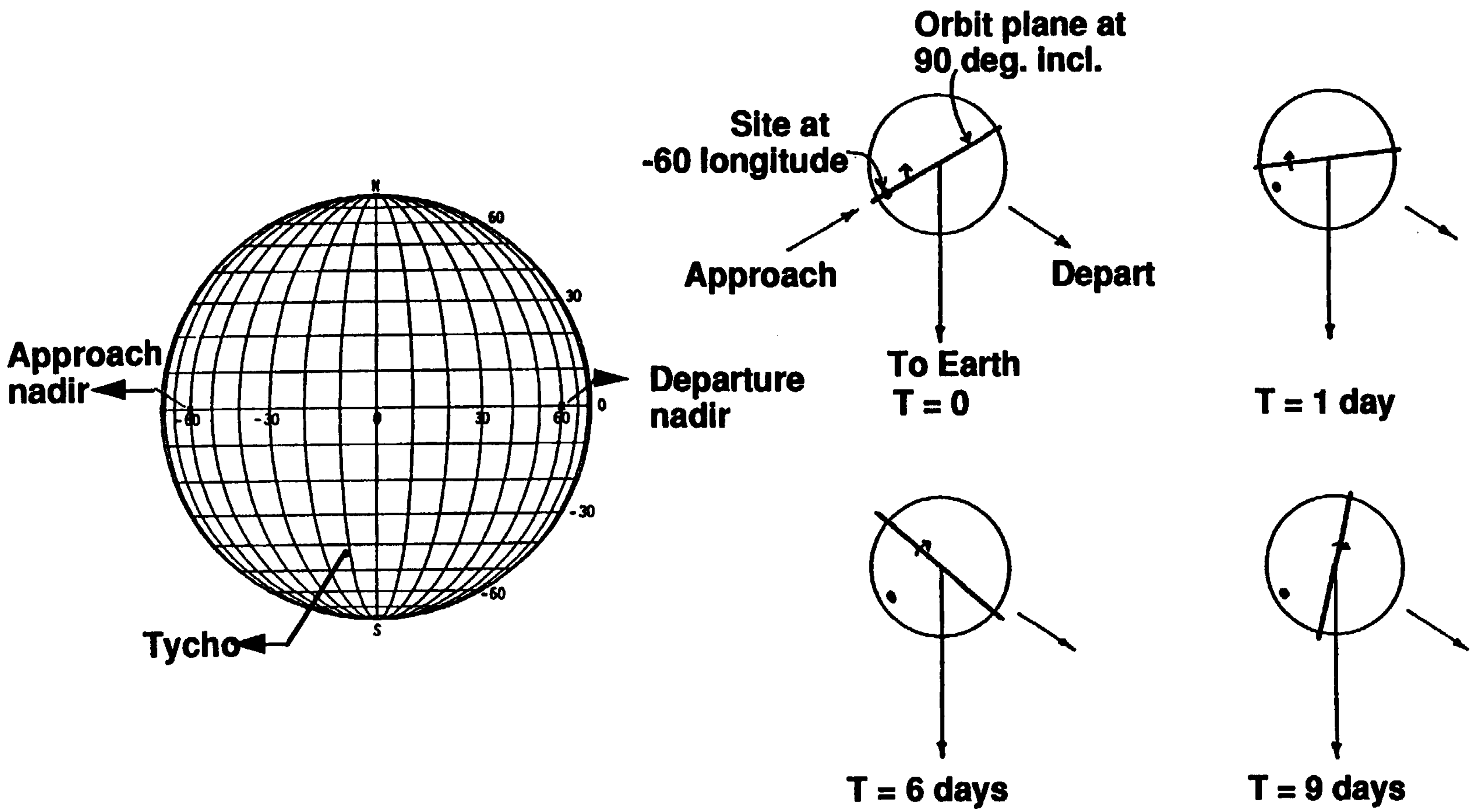
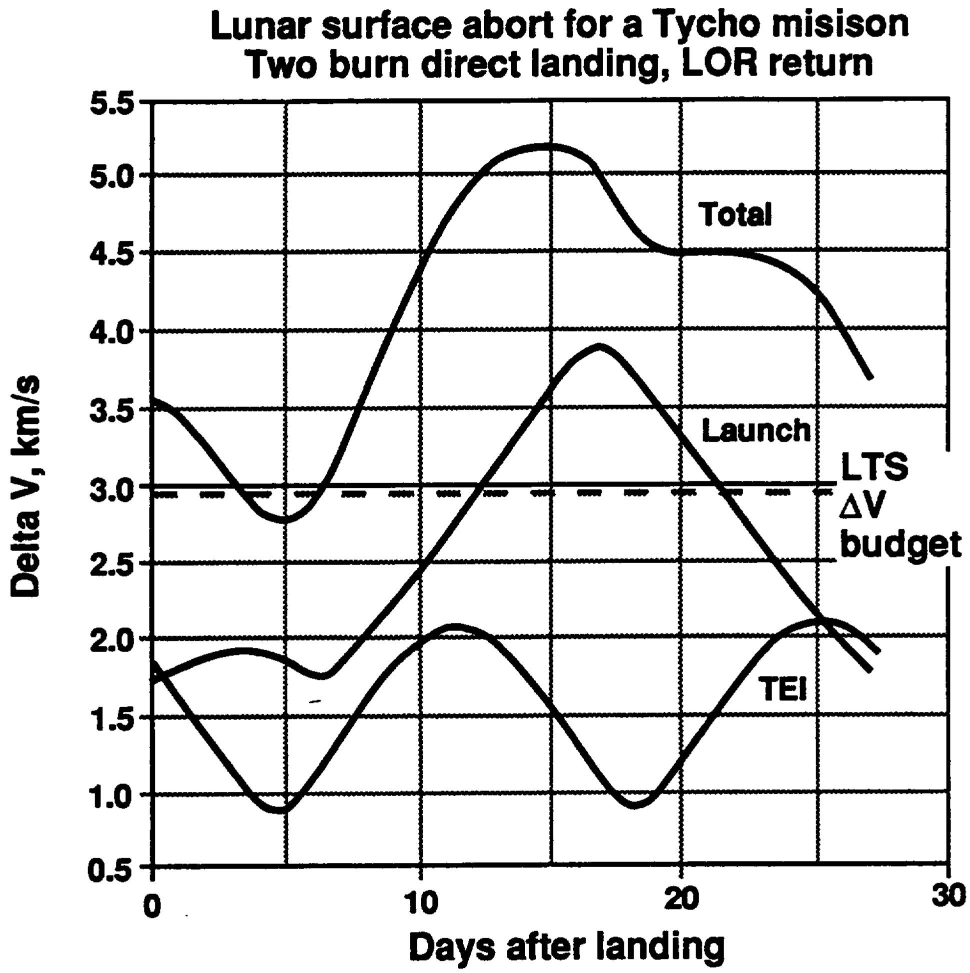


Figure 1.3.3-6 Lunar Geometry for LOR Abort Discussions



Return to Earth possible only during three day window once a month

Figure 1.3.3-7 LOR Abort Using Tycho Base Example

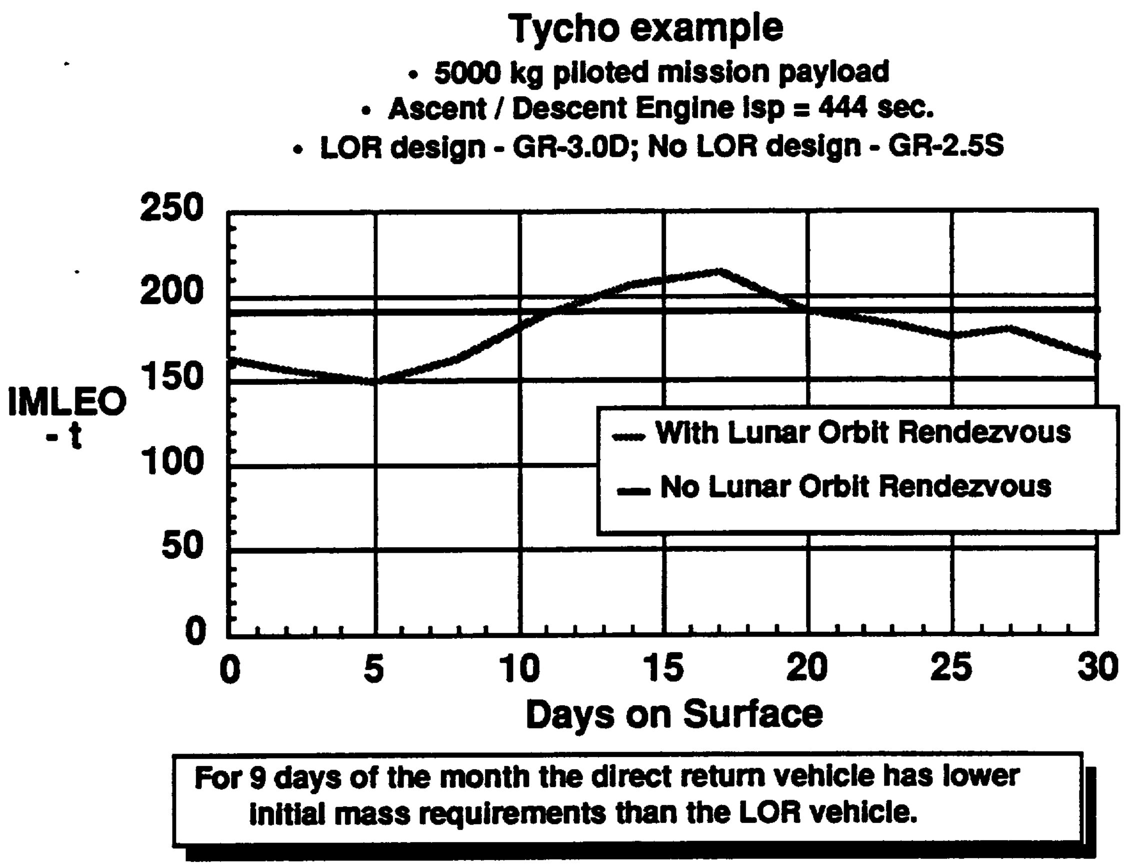


Figure 1.3.3-8 Launch Mass Requirements vs. Return Timing

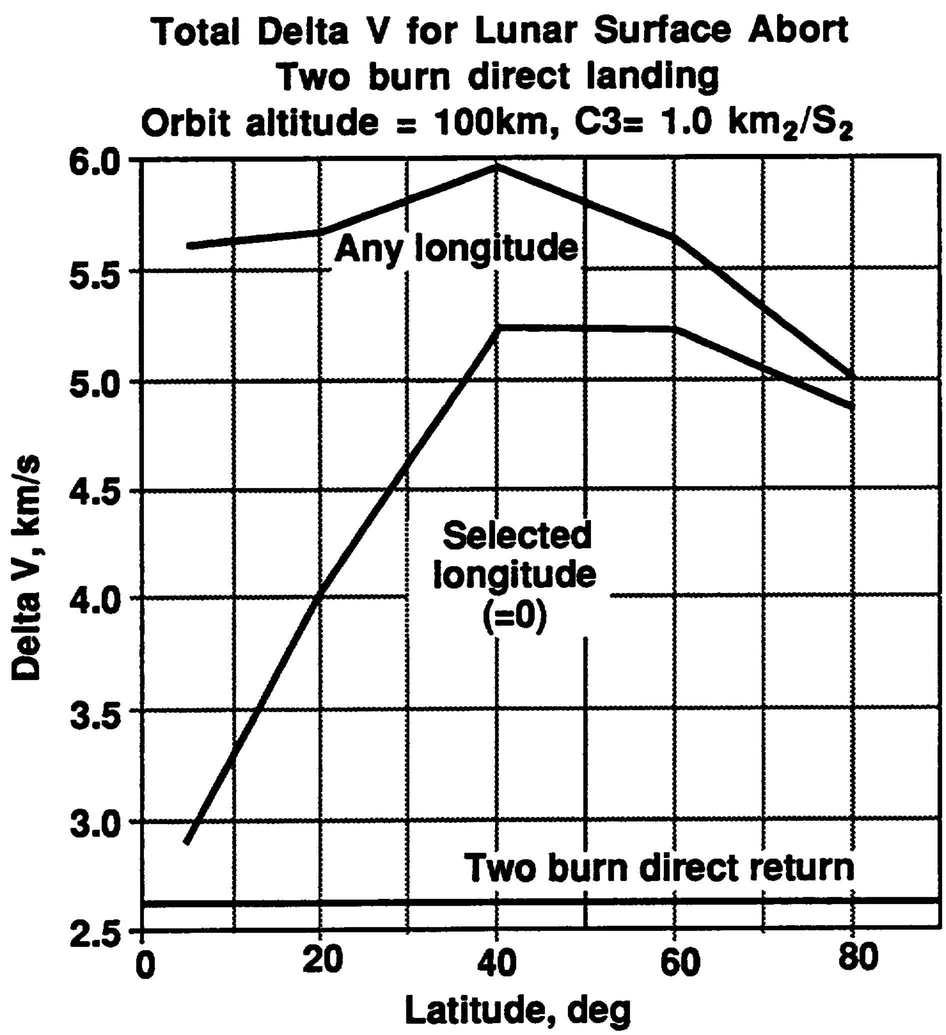


Figure 1.3.3-9 Comparison of Worst-Case Lunar Abort Delta-V's

Assumptions

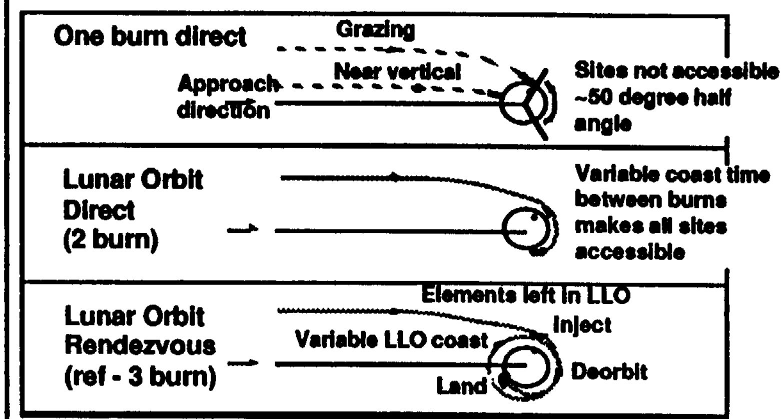
- Three approaches examined
  - Lunar Orbit Rendezvous leaves elements in LLO
  - Lunar Orbit Direct - nothing left in LLO, time in LLO is variable (generally fractional)
  - Direct one-burn approach targeted for landing site ~ 1 day prior to Lunar arrival
- All options initially on Free Return trajectory until retargeting
- Trade criteria (weighting): Cost (50%), Margins & Risk (30%), Other Mission Capture (15%), and Benefits to Mars (5%)

Delta V Analysis Results

Burn	Lunar Orbit Rendezvous (NASA ref)	Direct–one burn		Lunar Orbit Direct	
		High thrust	Low thrust	High thrust	Low thrust
	m/s	m/s	m/s	m/s	m/s
TLI	3300	3300	3300	3300	3300
MC	10	115	115	95	95
LOI	1100	0	0	1075	1075
DESC	2000	3041	3151	1900	1920
LLO	50	N/A	N/A	N/A	N/A
ASC	1900	2964	3114	1752	1822
TEI	1100	0	0	1075	1075
MC	10	10	10	10	10
ADJ	6	6	6	6	6
	310	310	310	310	310
	9786	9746	10006	9523	9613

Assume free return initial targeting (maximum lunar approach energy = 2 km /s)
Translunar retargeting at 0.5 days from posigrade to retrograde flyby requires additional 35 m/s
High thrust = 1.0 Wo, low thrust = 0.6 Wo applies to descent and ascent
Descent hover allowance = 127 m/s deduced from NASA ref 2000 m/s budget

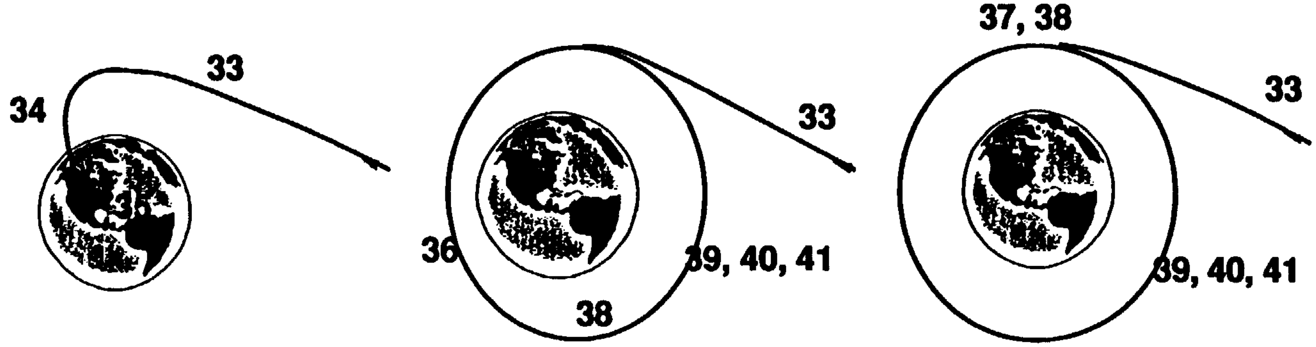
Lunar Approach Concepts



Conclusions

- Lunar Orbit direct (selected) has lowest Delta V and highest crew safety
- Lunar Orbit Rendezvous performance benefits outweighed by:
  - Abort risk: rendezvous & docking, multiple interface mate and verification required to pick up elements left in LLO prior to Earth return
  - Higher development costs (LLO station keeping elements)
- One-burn direct spends approximately 1 day on impact trajectory (risk) and has highest Delta V due to highest gravity losses

Figure 1.3.3-10 Phase 1 Use of LOR Trade Findings

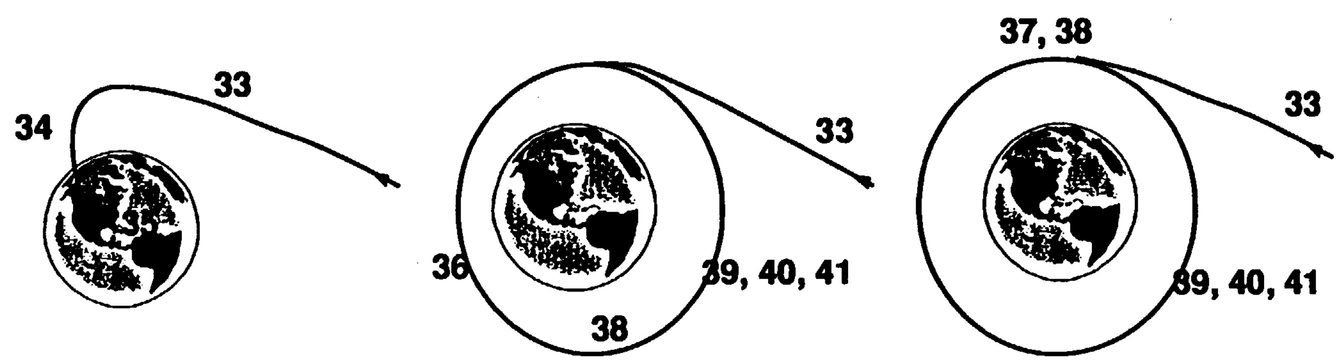


Earth Return

	Event *	Ground Return			Aerobrake, LEO Return			All-propulsive, LEO Return		
		∂-V (m/s)	∂-T (hrs)	Enabling Technologies	∂-V (m/s)	∂-T (hrs)	Enabling Technologies	∂-V (m/s)	∂-T (hrs)	Enabling Technologies
33	Trim	0	0.3	none	3	0.3	none	0	0.3	none
34	Earth Reentry	0	0.1	none	0	0.0	n/a	0	0.0	n/a
35	Dry Landing	0	0.0	Guidance, Chutes	0	0.0	n/a	0	0.0	n/a
36	Aeromaneuver				0	0.1	Aerobraking	0	0.0	n/a
37	LEO Insert				0	0.8	none	3150	0.8	none
38	LEO Circulariz				310	0.1	none	310	0.1	none
39	LEO Stay				0	48.0	none	0	48.0	none
40	*STS Launch Delay*				*0*	*720.0*	*Solar Power Sys*	*0*	*720.0*	*Solar Power Sys*
41	Rendezvous				0	6.0	Rendezvous	0	6.0	Rendezvous
		0	0.4		313	775.3		3460	775.2	

* Optional ops in italics

Figure 1.3.3-11 Earth Return Options



	Earth Return Event *	Ground Return Abort Capability	Ground Return Return Time (hr)	Aerobrake, LEO Return Abort Capability	Aerobrake, LEO Return Return Time (hr)	All-propulsive, LEO Return Abort Capability	All-propulsive, LEO Return Return Time (hr)
33	Trim	n/a	n/a	n/a	n/a	n/a	n/a
34	Earth Reentry	Return to Grnd	<1	n/a	n/a	n/a	n/a
35	Dry Landing	Water Landing	<1	n/a	n/a	n/a	n/a
36	Aeromaneuver			none		none	
37	LEO Insert			Abort to LEO	6 - 726**	none	
38	LEO Circularz			Abort to LEO	6 - 726**	none	
39	LEO Stay			STS Recovery	6 - 726**	STS Recovery	6 - 726**
40	*STS Launch Delay*			*Wait for STS*	*720 ***	*Wait for STS*	*720 ***
41	Rendezvous			STS Recovery	6	STS Recovery	6

* Optional ops in italics

** Up to 720-hr STS/PLS Launch Delay

Figure 1.3.3-12 Earth Return Abort Options with a waiting shuttle. The crew module is placed in the shuttle payload bay and returned to the ground for refurbishment and reuse.

For this option, the vehicle must be able to perform the aeromaneuver and return to a stable LEO orbit. In the case of an abort, the crew must wait on-orbit for a Shuttle rescue.

(3) Aerobraked return to SSF, LEO Storage – This option also uses a reusable aerobrake to slow down by braking through the Earth's atmosphere. After Earth orbit capture, the vehicle is circularized propulsively in the LEO transportation node orbit, and is refurbished and stored at the LEO node for reuse. The crew is returned to ground via the Shuttle or a PLS.

For this option also, the vehicle must be able to perform the aeromaneuver and return to a stable LEO orbit. In the case of an abort, the crew may return to the LEO node to await return to the ground.

(4) All-propulsive return to SSF or LEO – The all-propulsive return requires a 3,300 m/s ΔV main propulsion system burn for direct insertion into the required LEO. After Earth orbit capture, the vehicle is either refurbished and stored at the LEO node or returned via the Shuttle to the ground. The crew is returned to ground via the Shuttle or a PLS.

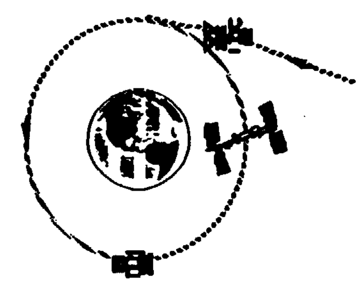
For this option also, the vehicle must be able to perform the Earth orbit capture and return to a stable LEO orbit. With a LEO node available, the crew may return to the LEO node to await return to the ground.

(5) Recommendations – A summary of the Phase 1 and Phase 2 findings concerning earth return options are shown in Figures 1.3.3-13 and 1.3.3-14, respectively. Even though the LTS would be a golden opportunity to demonstrate aerobraking before committing to an aerobraked Mars mission, our data indicates it would be significantly cheaper operationally (and safer) to return to the launch site using a semiballistic capsule with high glide parachutes, rather than stop in LEO and wait for pickup.

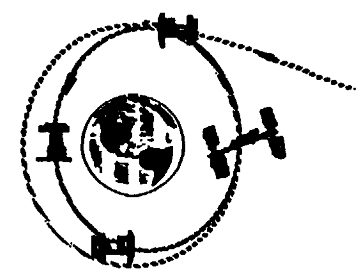
Assumptions

- Mission scenarios examined for this trade all used an LLO node for storage of Aerobrake (A/B) or return propellant during mission
- A/B refurbished and reused for a total of 5 mission uses
- All-propulsive return tanks dropped and replaced via wet launch for each mission
- Trade criteria (weighting): Cost (50%), Margins & Risk (30%), Other Mission Capture (15%), and Benefits to Mars (5%)
- Trade study based on Lunar Missions only

Concepts



- All-propulsive return to LEO node
- Return propulsive Delta V = 3300 m/s



- Aerobrake used through Earth atmosphere to slow vehicle
- Return propulsive Delta V = 310 m/s for LEO circularization post-A/B maneuver

Trade Scoring

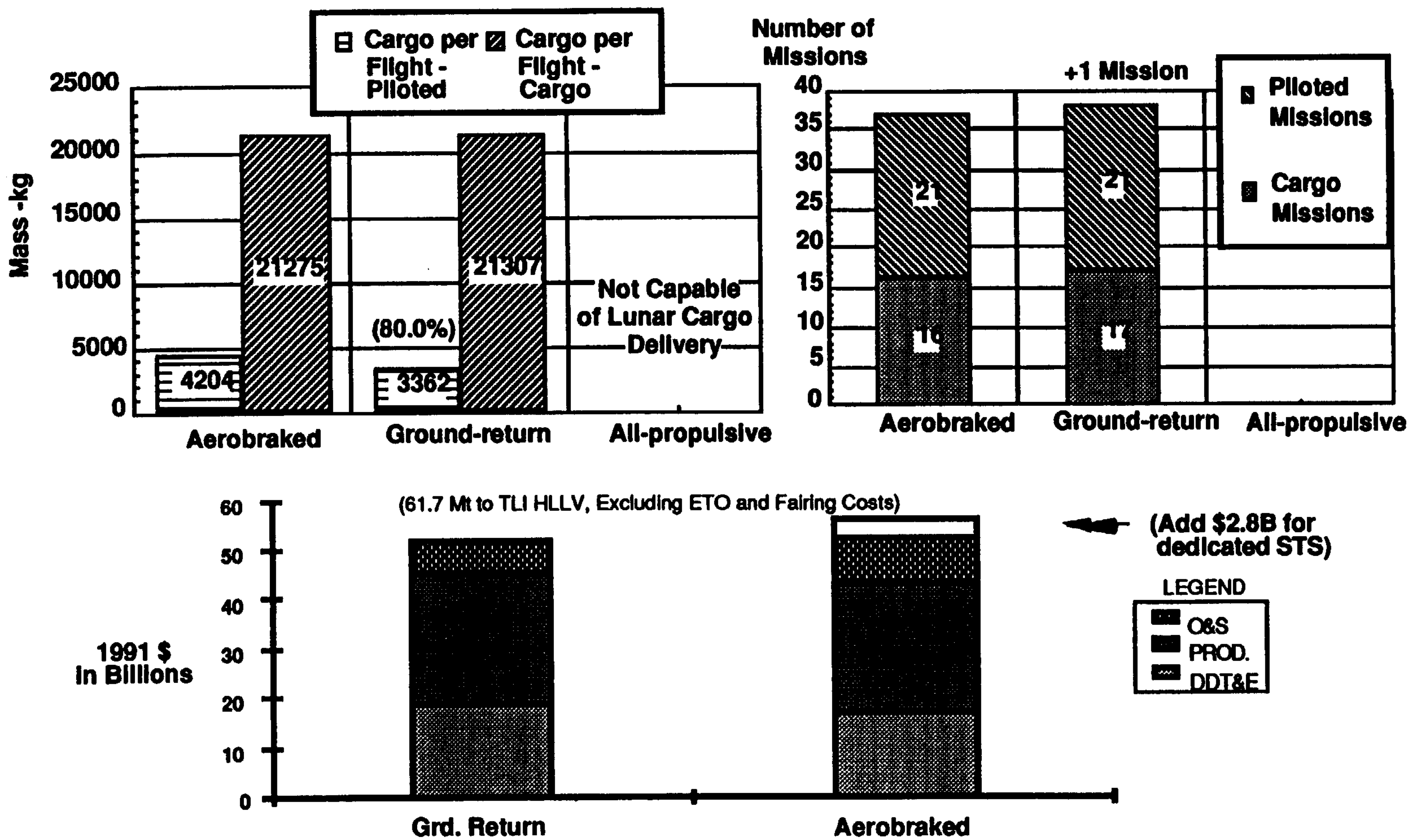
RETURN APPROACH	COST	M&R	MSSN CAP	MARS	TOTAL
Aerobrake	2.23	4.51	3.04	1.19	2.71
All-Propulsive	2.11	3.99	2.18	1.97	2.27

- Scoring based on 1-5 scale with 1 being best
- Total scores (.5 Cost + .3 M&R, etc.) respread to 1-5 scale
- Cost score is 70% DDT&E + 30% LCC, based on Low boost cost ($1K/kg to LEO)

Conclusions

- Cost slightly favors All-propulsive: influenced by high weighting of DDT&E
- A/B required for Mars, STV is the only SEI opportunity to prove out Aerobraking
- Margins & Risk and Benefits to Mars criteria tend in opposite directions: new technology which benefits Mars is accompanied by higher risk
- Trade did not examine Lunar Direct scenarios or questions of RL-10 + A/B vs All-propulsive with ASE, etc.
- Ranking by Total Score indicates A/B should be retained (All-propulsive available as backup (pending AFE))

Figure 1.3.3-13 Phase 1 Earth Return Findings



3.0-Stage, Dual Crew Module Vehicles

Launch Vehicle Option #4 with PoD Shroud (Capability to TLI - 61.7 mt)

Figure 1.3.3-14 Earth Return Trade Performance Summary

d. Suborbital Staging at Launch – One method of increasing delivered payload capability from a launch vehicle is to deploy the upper stage suborbitally, taking advantage of the typically higher upper stage specific impulse. Historically, upper stages that had an Isp comparable or higher than that of the launch vehicle have been deployed suborbitally to maximize payload capability. Examples include the SIVB stage and Centaur upper stage. Lower Isp upper stages such as the IUS typically have been launched on orbit and do not benefit as much from suborbital deployment. Because the STV/LTS designs are assumed to have high Isp and relatively high thrust, they could benefit from suborbital deployment based on these criteria.

Optimization of the ΔV split between launch vehicle and LTS over a range of upper stage thrust levels, for four of the launch vehicles potentially available for SEI, is shown in figure 1.3.3-15. Higher upper stage thrust levels result in larger staged weight, with more ascent delta-velocity provided by the upper stage, and more mass available for the TLI burn. For a ground-based, 2.5 stage concept, the effect of increased upper stage (in this case, the TLI stage) thrust on performance is shown in figure 1.3.3-16. In general, increased upper stage thrust results in large increases in upper stage propellant, but yields only small delivered cargo improvement. The cost optimum TLI stage is probably the one with five RL10 engines, since it can deliver as much payload as the far larger J2 powered stage, but has engine out capability.

e. Launch Vehicle Integration – Vehicle integration for launch depends on whether the flight crew is launched aboard the vehicle, the number of flight elements included in each launch, the amount of vehicle on-orbit assembly required, and the degree of flight element reusability, as shown in Figure 1.3.3-17. For example, designs that include a crew aboard the launch vehicle require the crew module to be at the top of the stack in order to allow launch escape. The number of flight elements can also affect the launch configuration in the case of a single-launch system. If a separate lunar lander and transfer stage are launched together, as on Apollo, an on-orbit turn-and-dock manuever must be performed to allow crew transfer between stages.

On-orbit assembly of an aerobrake or other flight element also affects the integration of the STV in the launch vehicle. The launch configuration may be a series of cargo

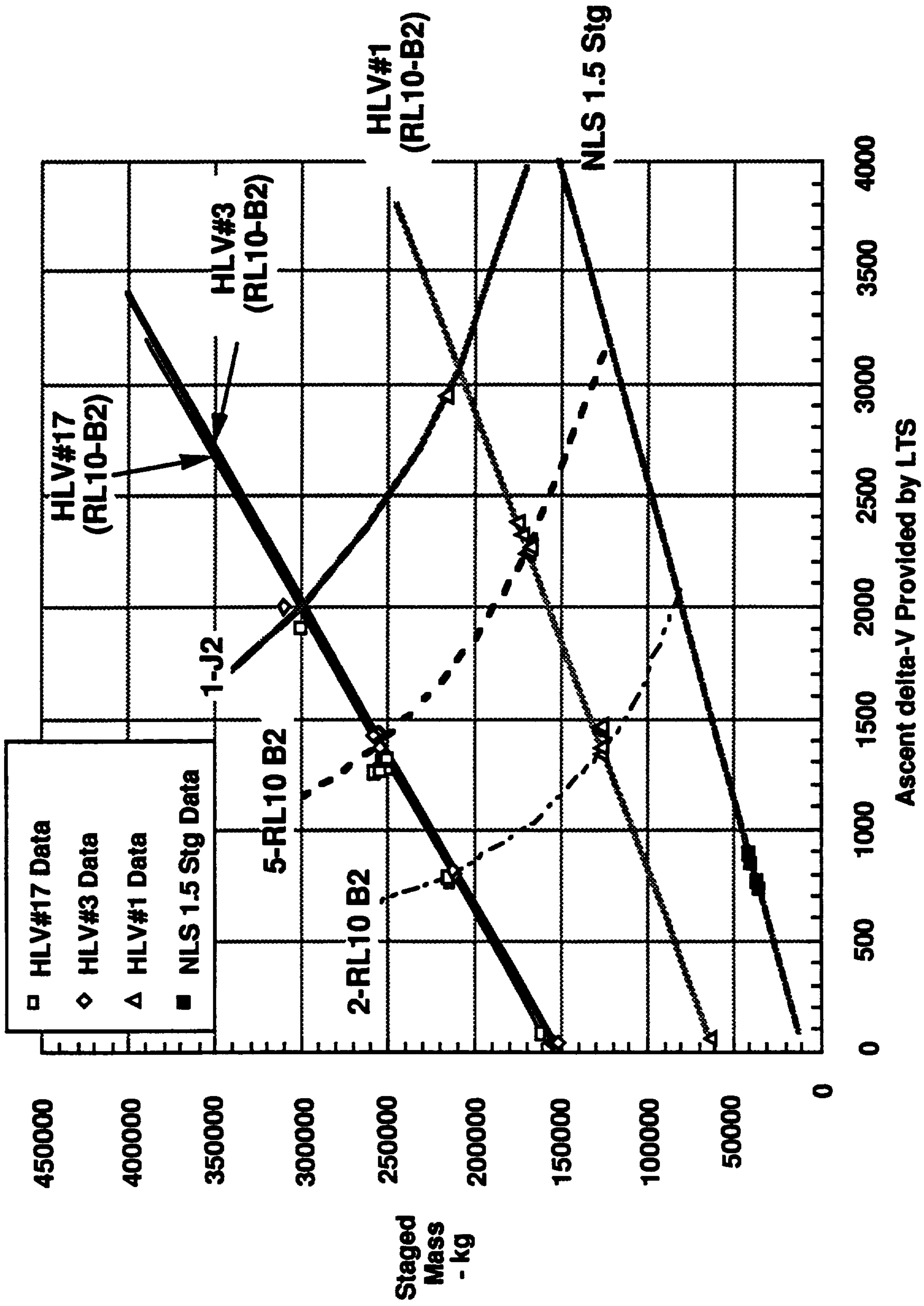


Figure 1.3.3-15 HLV Staged Mass vs. LTS Ascent Delta-V

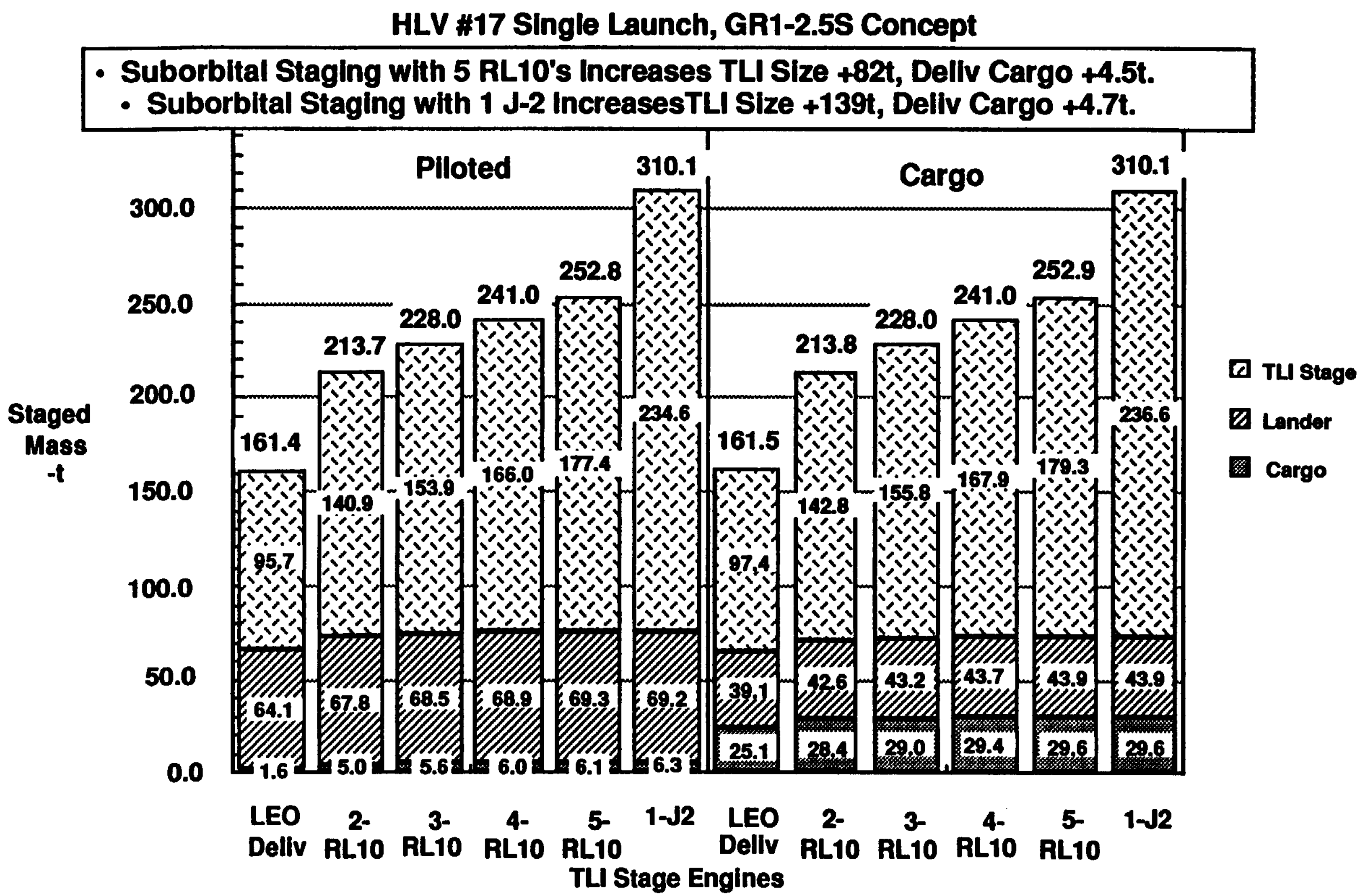


Figure 1.3.3-16 Suborbital Staging Impact on Cargo

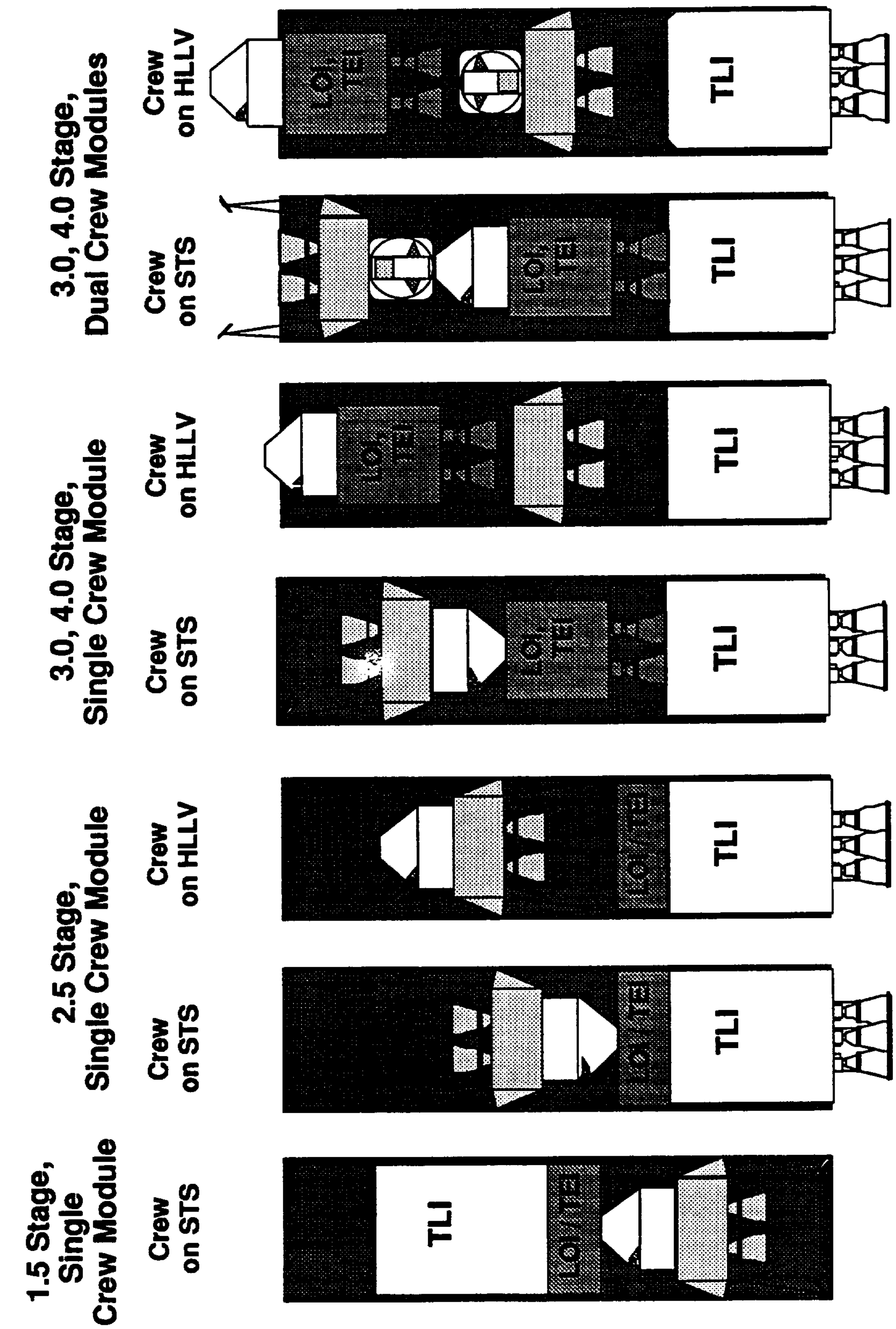


Figure 1.3.3-17 Launch Vehicle General Manifests

containers rather than a flight vehicle. Reusable systems requiring on-orbit assembly would be similar, with flight elements launched separately and assembled on orbit. Vehicles with fewer stages, less on-orbit assembly, and that don't require lunar orbit rendezvous present the fewest launch packaging problems.

1.3.4 Reusability Options

One method of reducing recurring costs and increasing vehicle performance is to reuse portions of the vehicle for subsequent missions. Key issues associated with reuse are; the refurbishment of reusable flight elements in space, and the refilling of reusable tanks in zero-g conditions.

a. Ground-Refurbishment – In this option, only the crew module is reused. It is returned to the ground, either ballistically or in the Shuttle payload bay, and is refurbished for subsequent reuse.

b. Ground and SSF refurbishment – A variation of the ground refurbishment approach has a portion of the flight vehicle returned to the Space Station and refurbished and refilled, while the labor-intensive refurbishment of the crew module takes place on the ground. In this case, the reusable stage returns to the station either by aerobraking or all-propulsively. This option reduces the amount of inert weight to be launched to orbit, and saves high-cost avionics and propulsion elements.

c. Space Station Refurbishment – In this option, both the core stage and crew module are returned to the station, refurbished, and reused. This reduces even further the inert weight launched to orbit, and again saves high-cost avionics and propulsion elements.

d. Reusability Recommendations – Because vehicle reusability corresponds to the method of launch, the findings in Phase 1 correspond to the ETO options summary shown in Figure 1.3.3-3. These results showed that ground-based refurbishment of the crew module with a two-launch rendezvous and dock scenario had the lowest cost, the least risk, and the best mission capture. For this reason, phase 2 of the study didn't deal with space-basing.

1.3.5 Number of Stages

Typically, a multi-stage system exhibits better performance than a single-stage system by dropping inert weight after propellants are expended. That inert weight may be either full stages with propulsion systems, or drop-tanks with no propulsion. The benefits of staging are evident in improved performance, but increase vehicle complexity and cost. For the design of a lunar STV, the number of stages may range from a single stage up to four stages (e.g. Apollo). For the purpose of consistent nomenclature, flight elements with both propulsion and tankage are considered full stages., and propellant drop-tanks (one or more) are considered as a half stage.

An exception to the performance benefit of increased number of stages is the case of a multi-stage lunar lander, in which the addition of an ascent stage does not necessarily increase lunar-delivered cargo, due to the significant added inert weight of the propulsion system. In this case, the use of lunar drop-tanks might be better than a separate ascent stage.

a. Tank Drop or Staging Options – One issue associated with staging is disposal of the staged flight elements. Figure 1.3.5-1 shows typical staging options in an expendable concept, stage or drop-tank disposal can be accomplished by reentry into the Earth atmosphere, by being boosted out of the Earth-Moon system, by impact on the lunar surface, or by being left on the lunar surface. In a reusable mode, stages may be left in LEO or LLO for refilling and reuse. In this case though, the propellants must still be transported.

b. Staging recommendations – For space-based missions, the best mission performance occurs with staging events following the first and second burns (TLI and LOI for options using LOR, and TLI and lunar descent for lunar-orbit-direct options). For ground-based missions, the best performance occurs with staging following TLI for both lunar-orbit-rendezvous cases and and lunar-orbit-direct cases, and following lunar descent (drop-tanks only) for the lunar-orbit-direct cases. High performance penalties occur for no staging events on lunar-orbit-direct vehicles and for lunar ascent droptanks on LLO node vehicles.

From a cost point of view, minimum cost sytems are those that minimize the number of stages, due to added development and increased recurring costs (thrown-away engines) of multi-stage vehicles. The lowest cost, best performance designs minimize the number of full stages, but still expend inert weight by using expendable drop-tanksets. A summary of the Phase 1 and Phase 2 findings concerning staging options are shown in Figures 1.3.5-2 and 1.3.5-3, respectively.

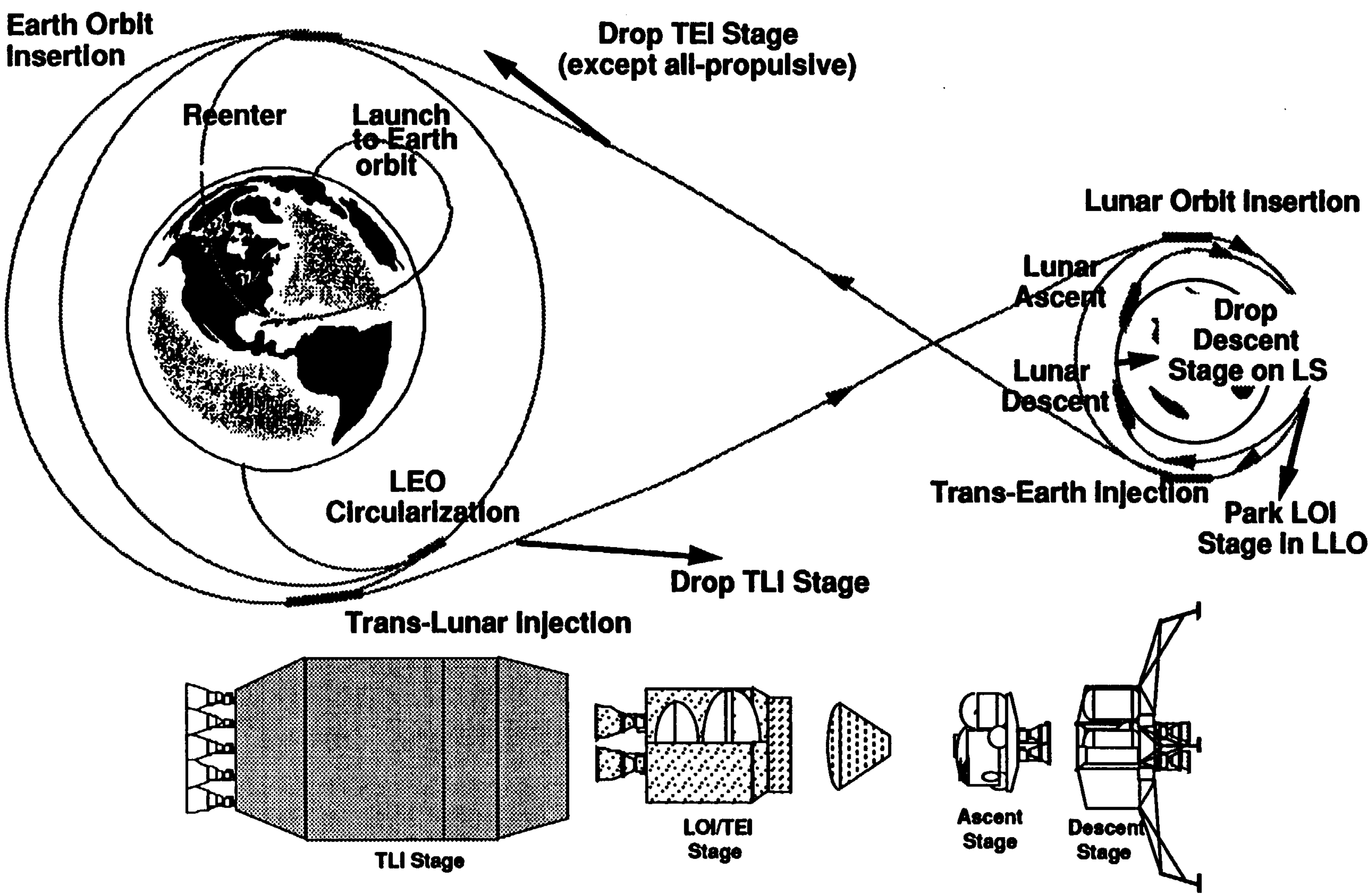


Figure 1.3.5-1 Vehicle Staging Options

Assumptions

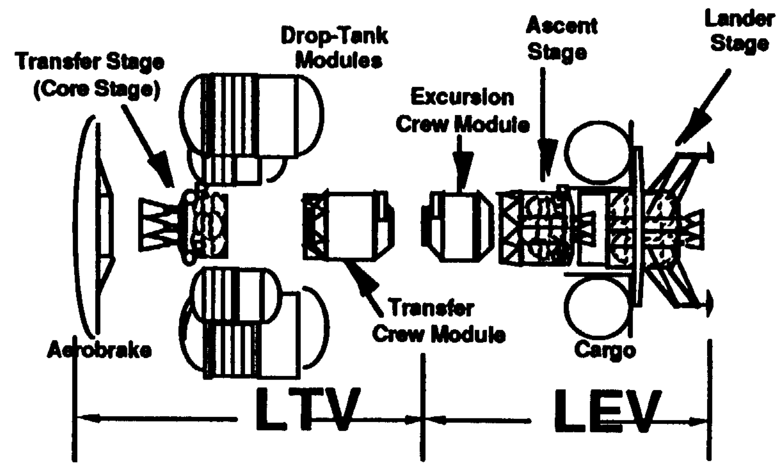
- Stage is propulsion + tanks, .5 stage is a set or sets of drop tanks
- 1.5 - 3 stages examined for direct landing cases
- 1.5 - 4 stages examined for cases using LLO node
- Trade criteria (weighting): Cost (50%), Margins & Risk (30%), Other Mission Capture (15%), and Benefits to Mars (5%)
- Trade study based on Lunar Missions only

Trade Scoring

STAGE NUMBER	COST	M&R	MSSN CAP	MARS BENE.	TOTAL
1.5	1.74	3.18	3.11	2.72	1.92
2.0	3.01	3.33	3.12	2.58	2.88
2.5	2.90	3.36	2.84	2.58	2.75
3.0	3.95	3.40	3.30	2.58	3.60
3.5	3.72	3.28	3.28	2.58	3.53
4.0	4.81	3.28	3.68	2.58	4.24

- Scoring based on 1-5 scale with 1 being best
- Total scores (.5 Cost + .3 M&R, etc.) respread to 1-5 scale
- Cost score is 70% DDT&E + 30% LCC, based on Low boost cost ($1K/kg to LEO)

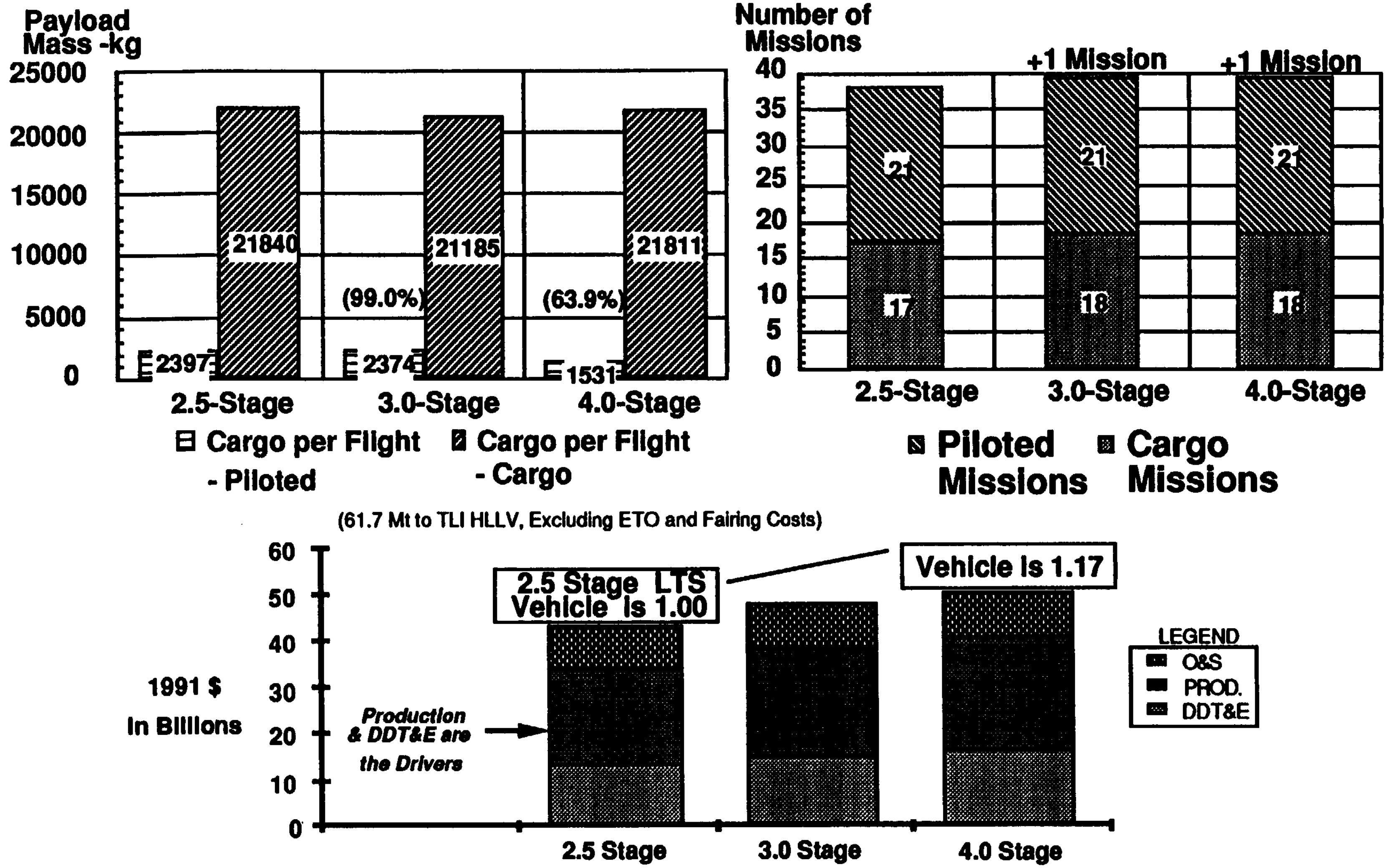
Sample 3.5 Stage Concept



Conclusions

- Cost and Margins & Risk evaluation criteria tended to favor less stages (less DDT&E and operational costs/complexity)
- Benefits to Mars criterium favored higher number of stages
- Total scoring strongly favored less stages with 1.5 stage concepts (selected) preferred

Figure 1.3.5-2 Phase 1 Stage Number Trade Findings



Aerobraked, Single Crew Module Vehicles
Launch Vehicle Option #4 with PoD Shroud (Capability to TLI - 61.7 mt)

Figure 1.3.5-3 Phase 2 No. Stages Trade Summary

1.3.6 Crew Module Design

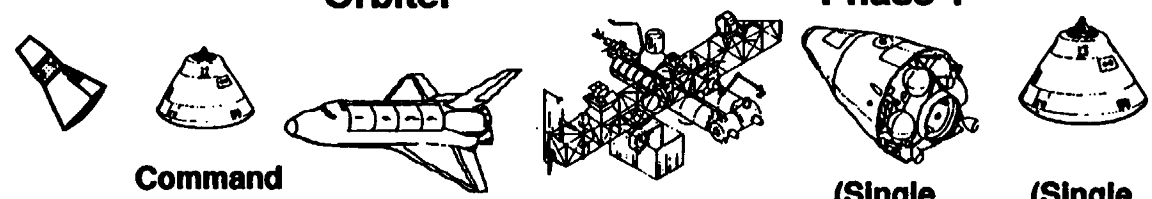
The U.S. Space program has a history of manned spaceflight extending back 30 years. Past and proposed future space missions have specific requirements that influence crew module design, as shown in Figure 1.3.6-1. The current STV study has focused on a return to the moon, with a purpose of going back to stay. With this purpose, several factors influence crew module design, including mission function, crew safety, and crew comfort.

a. Crew Module Type – Crew module function and crew safety considerations drive the type of crew module used. Options include a single crew module, a hybrid crew module, and a dual crew module. The differences in crew module functionality are shown in Figure 1.3.6-2.

With a single crew module, the crew remains in one crew module throughout the mission. That crew module must perform all life support functions and ensure crew safety during all mission phases including lunar transfer as well as the lunar excursion. The hybrid crew module design includes two crew modules that are both used for lunar transfer, with only one being used for the lunar excursion portion of the mission. This option requires the use of lunar orbit rendezvous, leaving one crew module in orbit during the lunar stay. The third design case is a dual crew module system, where separate transfer and excursion modules are designed for the distinct mission phase requirements. This case also requires lunar orbit rendezvous, and only the transfer crew module is used for the return to Earth (i.e. like Apollo).

A performance and cost comparison between the crew module types is given in Figure 1.3.6-3. The recommended crew module arrangement is a single crew module design, due to the high cost of additional crew module development and the requirement for LOR with its limited abort options.

b. Crew return options – Upon return from the moon, the crew module with crew may be designed to reenter Earth atmosphere and return to the ground, or may return to LEO following an aerobraking maneuver and rendezvous with a Shuttle or PLS to return the crew to the ground.
In the ground-return case, the objective is to return the crew module as near to the refurbishment facility as possible and with the least amount of damage to reduce



	Gemini	Apollo	STS Orbiter	SSF	STV Phase 1	LTS
		Command Module			(Single Module)	(Single Module)
Destination:	LEO	LLO	LEO	LEO	Lunar Surface	Lunar Surface
Crew Size:	2	3 / 1	7	8	4	4
Crew Module Life:	14 days	14 days	20 years	20 years	5 years	5 years
Mission Duration:	14 days	14 days	7-10 days	180 days	192 days	192 days
Crew Habitation:	14 days	14 days	7-10 days	180 days	15 days	15 days
Habitable Volume	80 ft3 (2.3 m3)	208 ft3 (5.9 m3)	2625 ft3 (74.3 m3)	11620+ ft3 (35.3 m3)	720 ft3 (20.4 m3)	400 ft3 (11.3 m3)
Hab. Volume/person:	40 ft3 (1.1 m3)	69 ft3 (2.0 m3)	375 ft3 (10.6 m3)	1452+ ft3 (41.1 m3)	180 ft3 (5.1 m3)	100 ft3 (2.8 m3)

Figure 1.3.6-1 Crew Module Background

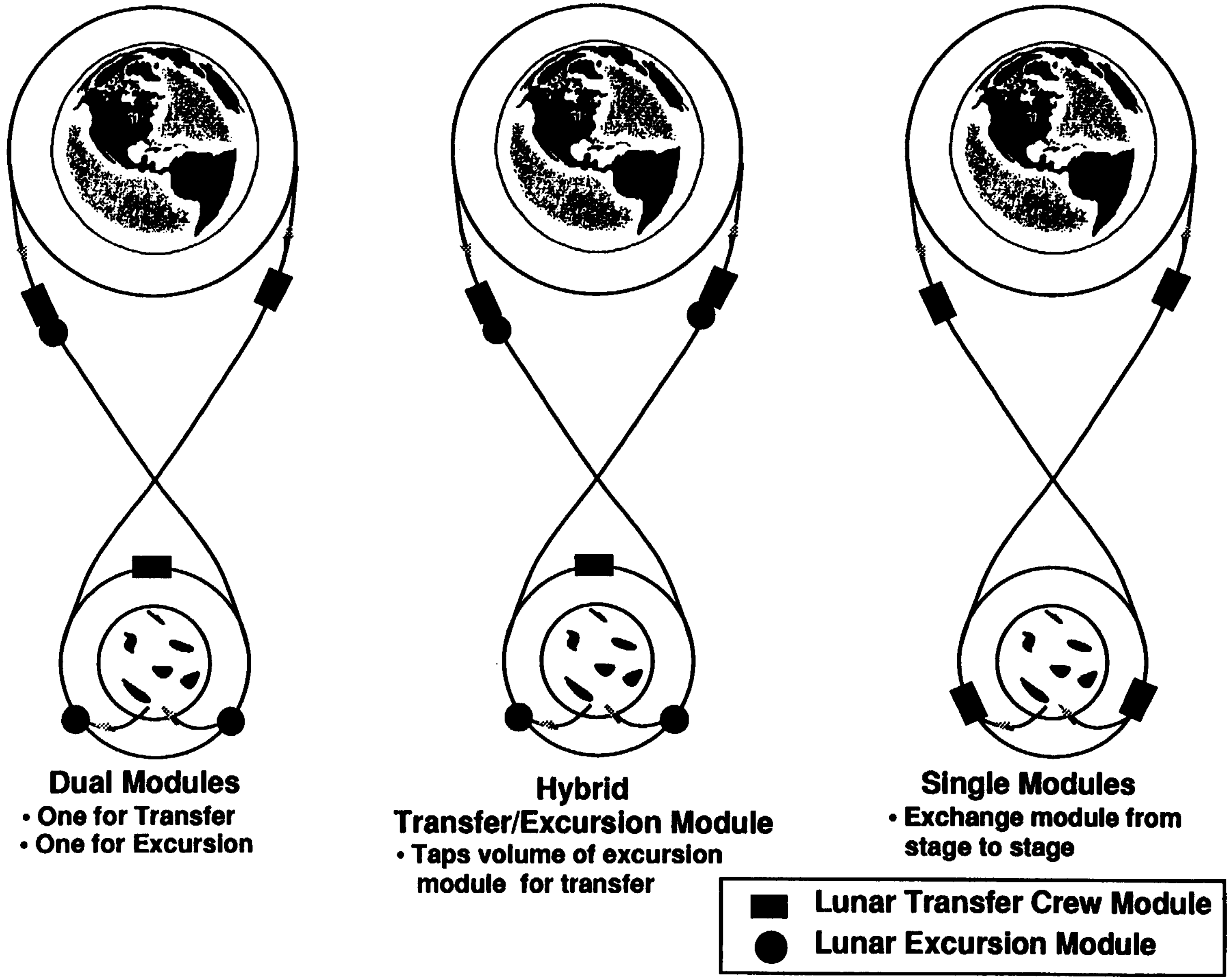


Figure 1.3.6-2 Crew Module Options

Assumptions

- Dual and Hybrid concepts are used only in conjunction with an LLO element storage node (Transfer module left in LLO during mission)
- Single concept can be used with or without use of an LLO node (i.e. tankage or A/B can be left in LLO)
- Radiation protection provided in transfer module only, or single module
- Trade criteria (weighting): Cost (50%), Margins & Risk (30%), Other Mission Capture (15%), and Benefits to Mars (5%)
- Trade study based on Lunar Missions only

Trade Scoring

CREW MODULE	COST	M&R	MSSN CAP	MARS	TOTAL
Single	2.01	3.41	2.65	2.73	2.11
Hybrid	2.31	3.01	2.93	2.66	2.21
Dual	3.30	3.56	2.87	2.58	3.12

- Scoring based on 1-5 scale with 1 being best
- Total scores (.5 Cost + .3 M&R, etc.) respread to 1-5 scale
- Cost score is 70% DDT&E + 30% LCC, based on Low boost cost ($1K/kg to LEO)

Crew Module Concepts

Dual

- Transfer cab stays in LLO during mission, returns to Earth/LEO between missions
- Excursion cab descends to surface during mission, stays in LLO between missions

Hybrid

- Transfer cab stays in LLO during mission, returns to Earth/LEO between missions-has no command functions
- Excursion cab descends to surface during mission, returns to Earth/LEO between missions

Single

- Single cab performs entire mission

- All concepts have space based or ballistic return options

Conclusions

- Cost favored single followed by hybrid. Leaving of excursion module in LLO between missions drove up cost for dual
- Hybrid had best Margins & Risk score with two pressurized volumes and no modules left in LLO between missions
- Dual module concepts were most similar to Mars mission approaches for best Mars benefits score
- Total score strongly favored Single and Hybrid and in conjunction with Lunar approach trade results (Lunar orbit direct), Single crew module was selected.

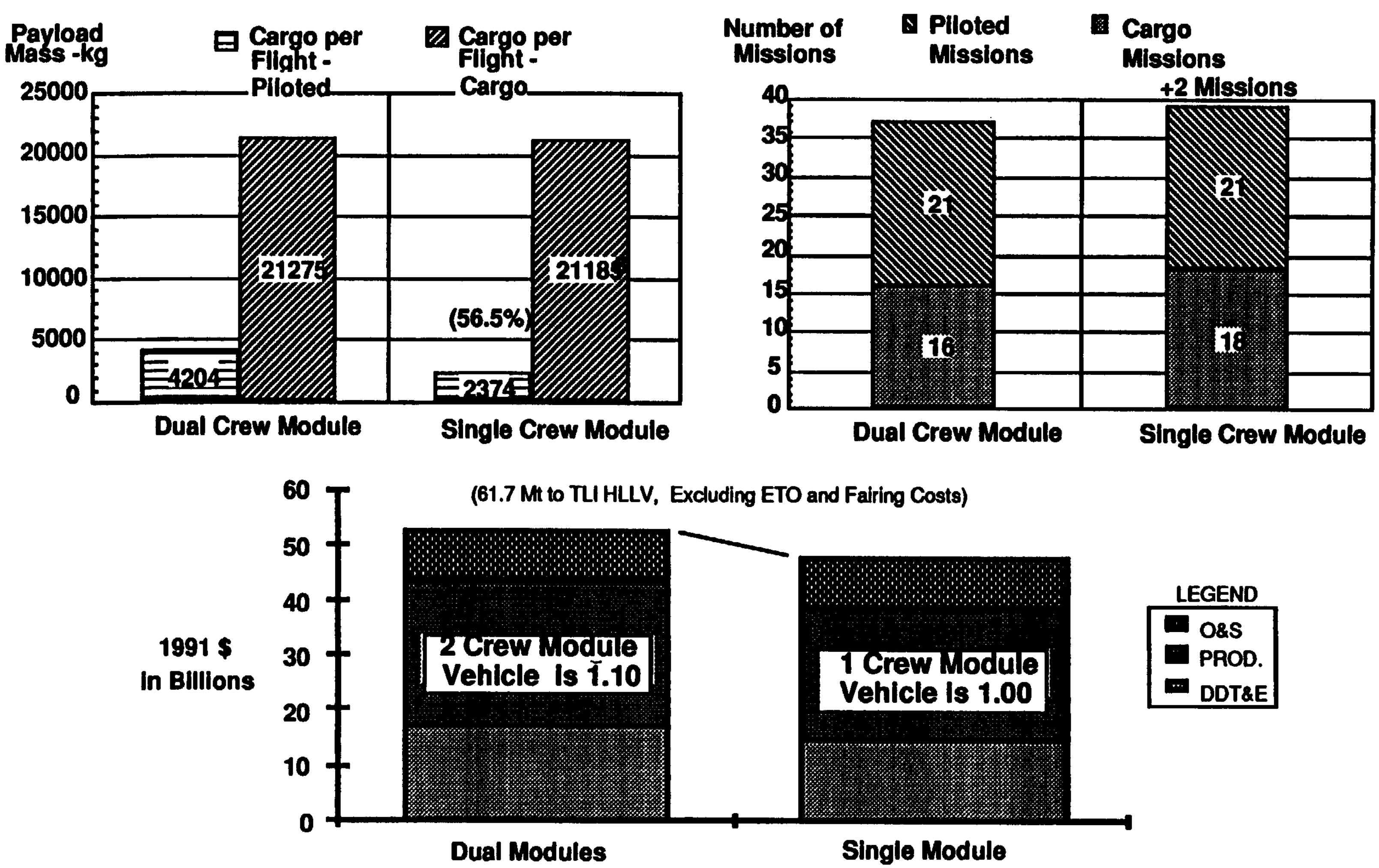
Figure 1.3.6-3 Phase 1 No. Crew Module Trade Summary recovery and refurbishment costs. This may be done by designing the module with sufficient L/D to accurately target a land landing site and providing the vehicle with sufficient impact attenuation to reduce damage. Two shapes studied to provide L/D include a modified Apollo-shape and a biconic shape.

For the LEO-return option, the crew module must have rendezvous and dock capability to dock with the Shuttle or SSF, but does not require the reentry TPS. However, the crew module will need to carry extra life support consumables for a contingency LEO stay, in case they return early due to an emergency or theShuttle launch is delayed.

c. Crew launch – The crew for a lunar STV mission may be launched either aboard the STV launch vehicle or aboard the Shuttle or PLS and then transferred to the lunar vehicle. The performance and cost impacts associated with launching the crew aboard the launch vehicle are given in Figure 1.3.6-4. The cost trade is essentially even if it is assumed the STS launch is paid for by another user and the STS drops off the lunar crew before of after their scheduled mission. However, we thought it more likely a scheduled STS launch would be required at least half the time and assessed cost penalties accordingly. In the case of a manned launch of the heavy-lift vehicle, provisions must be made for crew safety in the case of a launch abort. A launch escape system similar to that used on Apollo could be provided for launch aborts up to 400,000 ft, after which the upper stage could provide sufficient escape propulsion. The crew module must also be designed to withstand water impact loads following a launch abort.

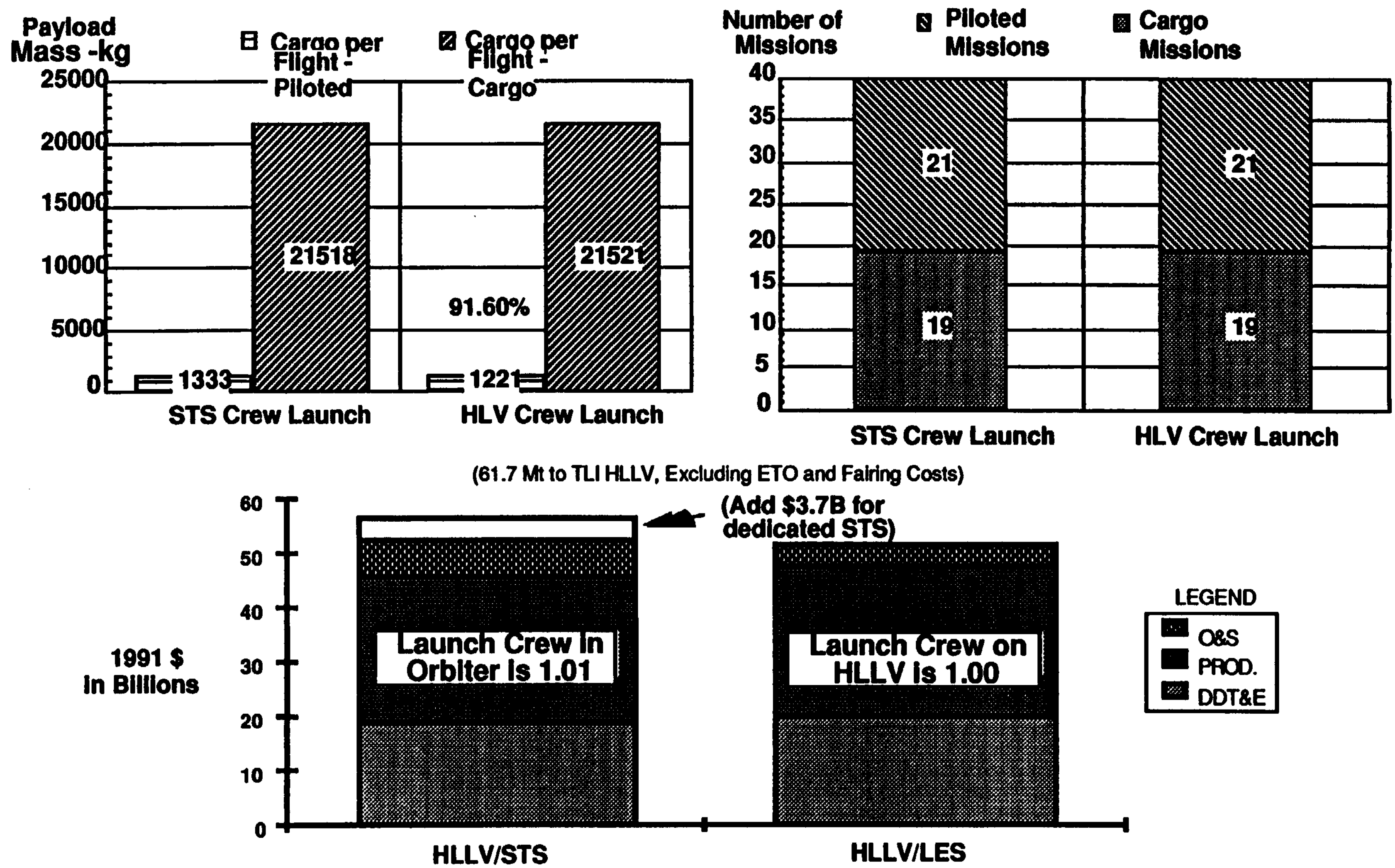
d. Habitability – Crew module habitability is driven by both crew comfort and mission functions, based on mission duration and crew size. In general, as the duration of a mission increases, the accessible volume of the crew module must also increase to provide additional functions, as shown in Figure 1.3.6-5. A historical perspective of crew module volume per person as a function of mission duration is given in Figure 1.3.6-6, as well as curves representing what is felt to be tolerable and optimum values. For lunar missions in which the crew lives in a separate lunar outpost, the transfer and excursion crew modules may have volumes similar to the Apollo crew modules.

If the crew is to live in the excursion crew module for an extended period of time, the crew module may have to be increased in size to meet habitability limits. Variations in



Aerobraked, 3.0-Stage Vehicles
Launch Vehicle Option #4 with PoD Shroud (Capability to TLI - 61.7 mt)

Figure 1.3.6-3 Phase 2 No. Crew Module Trade Summary



Ground-return, 3.0-stage, Dual Crew Module Vehicle
Launch Vehicle Option #4 with Updated Shroud Configurations

Figure 1.3.6-4 Crew Launch Trade Summary

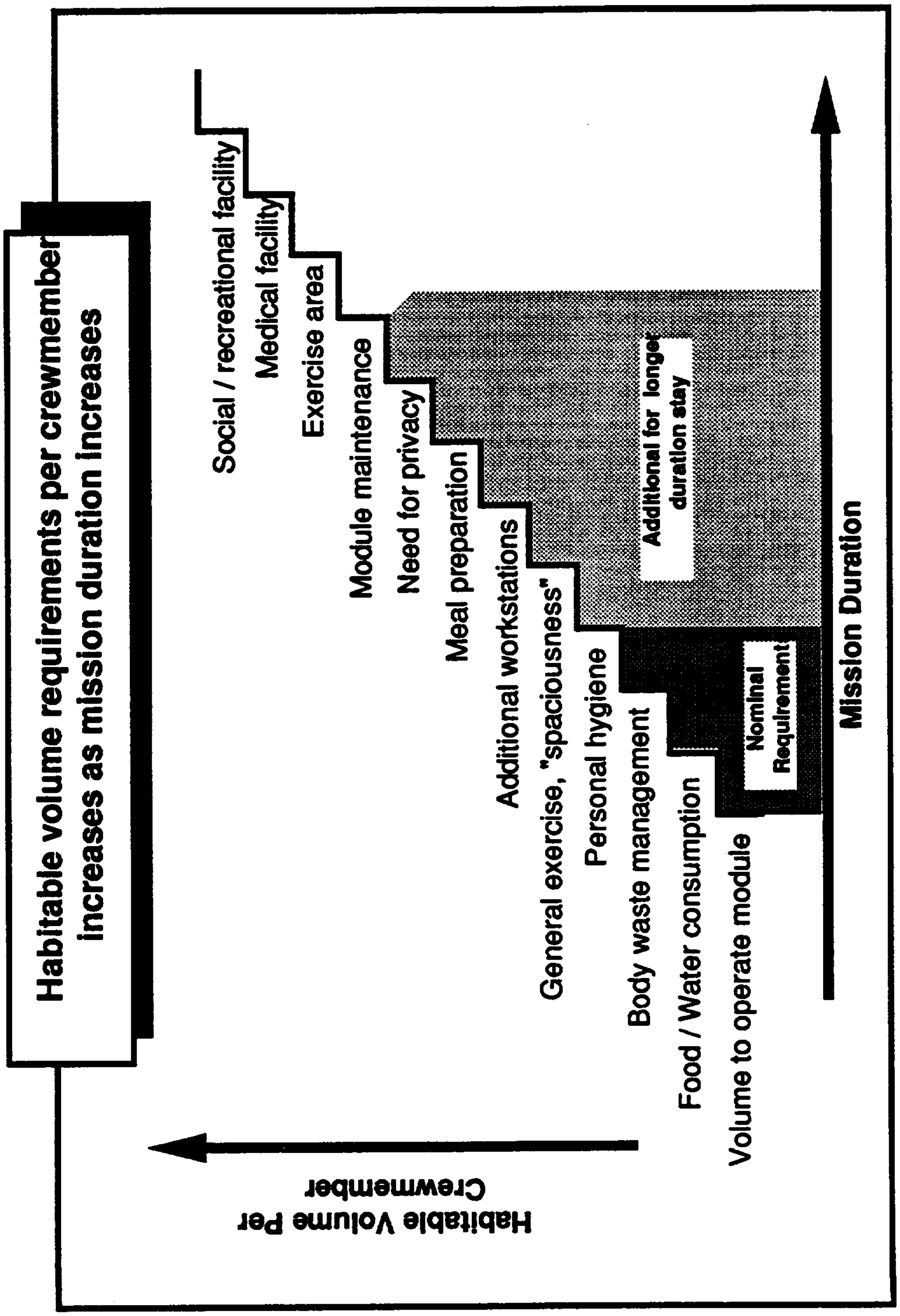


Figure 1.3.6-5 Crew Module Habitability Requirements

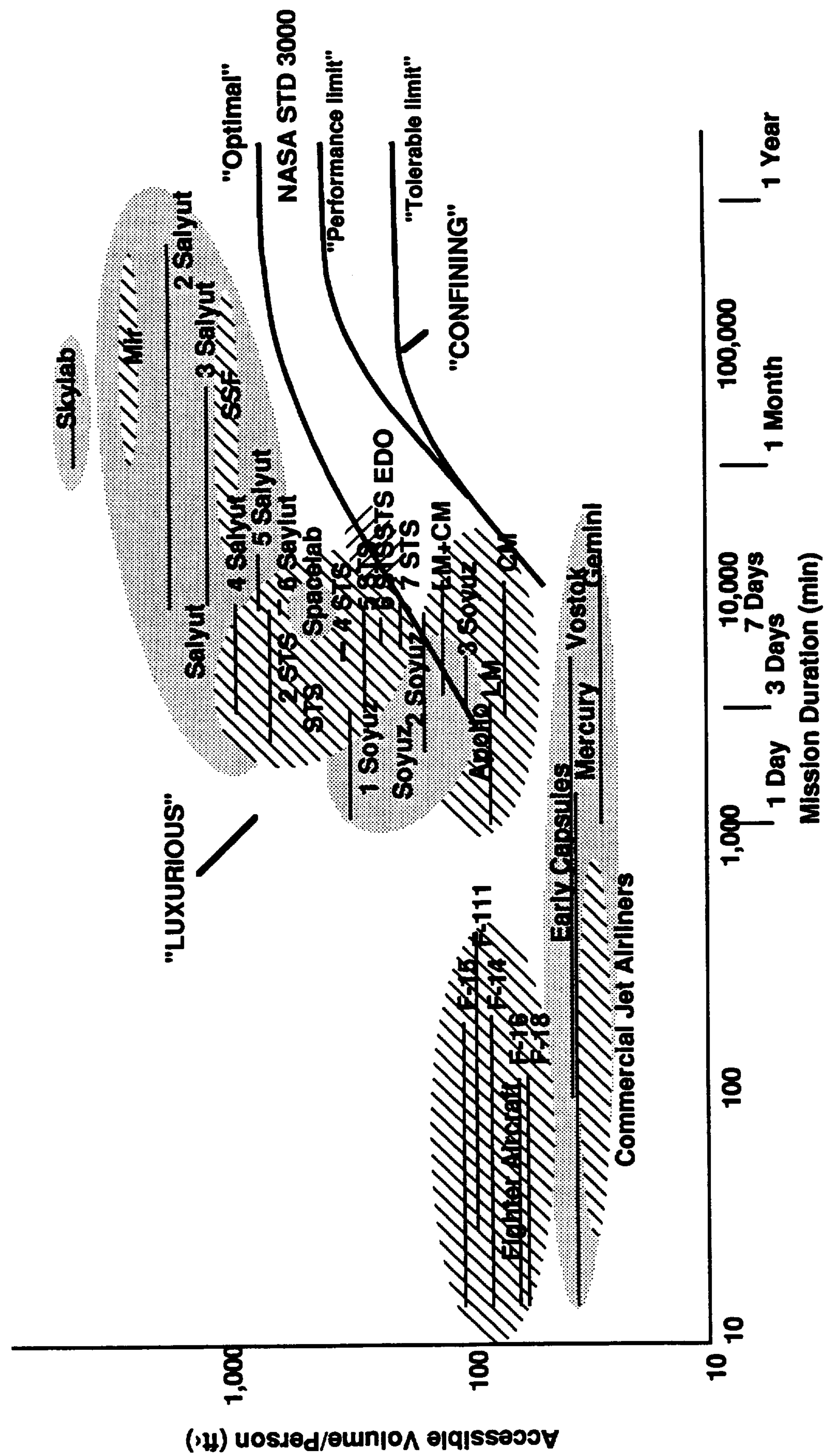


Figure 1.3.6-6 Habitable Volume Increase for Extended Duration

Assumptions:

- No Lunar Services Available (Power, Hab, etc)
- Power System: Fuel cells included on baseline system

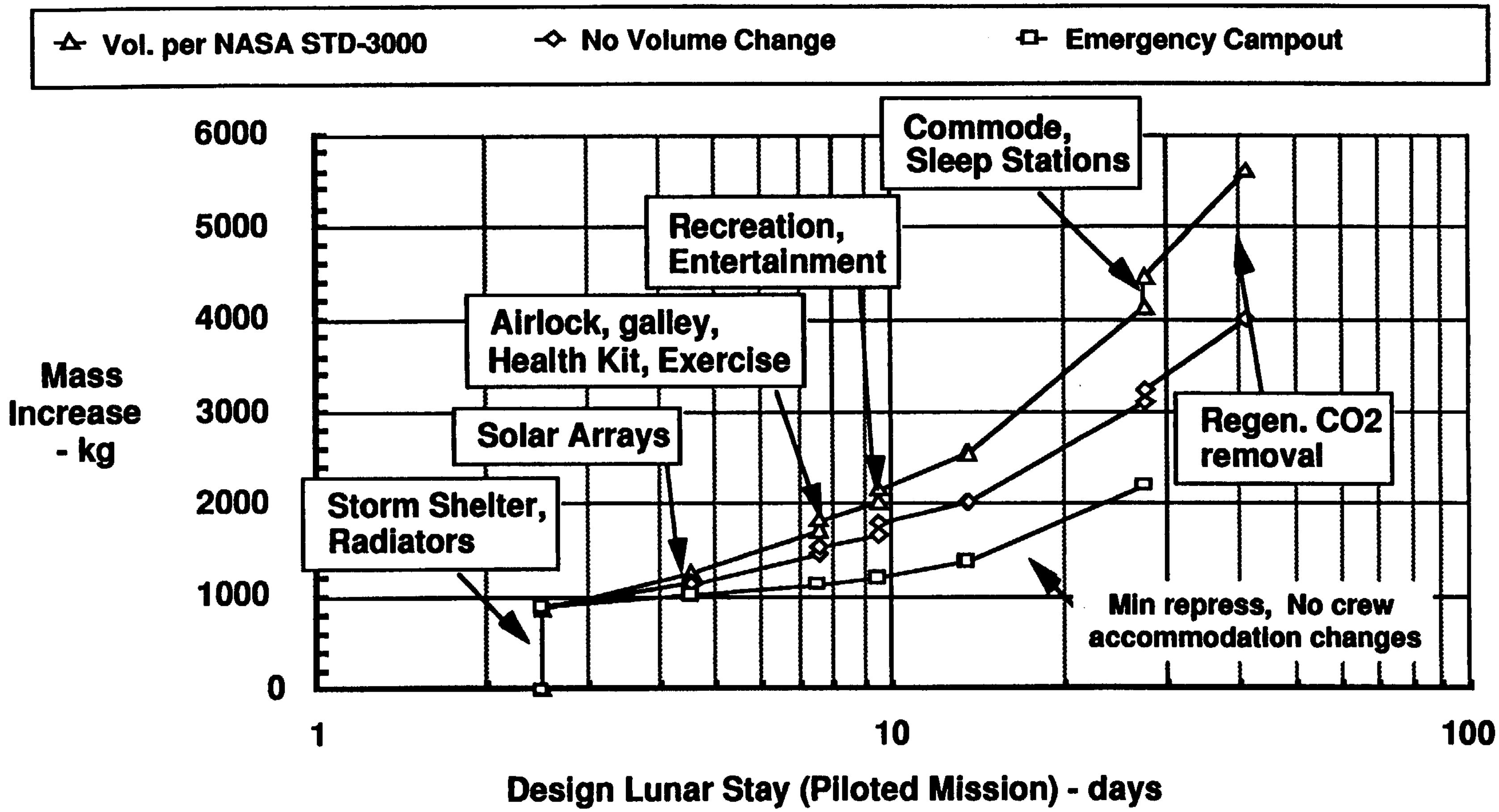


Figure 1.3.6-7 Mass Increase for Extended Crew Support

lunar crew module mass with increased inhabited duration is shown in Figure 1.3.6-7 for three cases: Case one is an emergency-only stay in which the crew module volume and equipment stays the same as the transfer stay module, Case two is where the volume remains the same and the equipment only changes , and Case three is where both equipment and volume change to meet NASA STD-3000.

1.3.7 Lunar Base Support

a. Propellant Boiloff – One of the disadvantages of high-performance cryogenic propellants is the associated boiloff during long-duration missions, especially on the lunar surface. In the absence of a lunar surface support system, these losses can significantly impact the performance of the vehicle. Several options to minimize the impact of propellant boiloff were evaluated including; on-board cryogenic refrigerators, improved thermodynamic vent systems and vapor-cooled shields, reduced heat leak paths, additional propellant tank insulation, a reflective lunar surface 'tarp' to minimize reflected heat from the lunar surface, and the use of storable propellants on a separate ascent/return stage.

A comparison of system mass for several types of LH_2 refrigerators in Figure 1.3.7-1 shows that even the lightest system requires a significant mass to reliquefy hydrogen and would be prohibitive if carried on board the vehicle. LO2 refrigeration may be more likely, as the peak cooling requirement for LO2 is around 30W, and refrigerator masses can be more than an order of magnitude smaller. The reference system for the Phase I lunar base was a surface-based refrigeration system of the Vullimier type.

Combinations of MLI, TVS, vapor-cooled shields, refrigeration, and reflective surface cover have been compared with the lunar surface support case to arrive at a minimum mass solution to minimize surface boiloff in the absence of lunar surface support. The various boiloff-control options are shown plotted in Figure 1.3.7-2 as cargo mass impact versus time on the lunar surface, compared to the reference. The minimum mass system from 1 to 65 days includes 80-layer MLI and optimized tank-support struts. At 65 days, the payload degradation is approximately -755 kg. From 65 to 180 days, the minimum mass system includes 80-layer MLI, optimized struts, vapor-cooled shields, and an on-board LO2 refrigerator. At 180 days, the payload degradation is -1154 kg. The storable ascent stage option represents a payload degradation of -7900 kg.

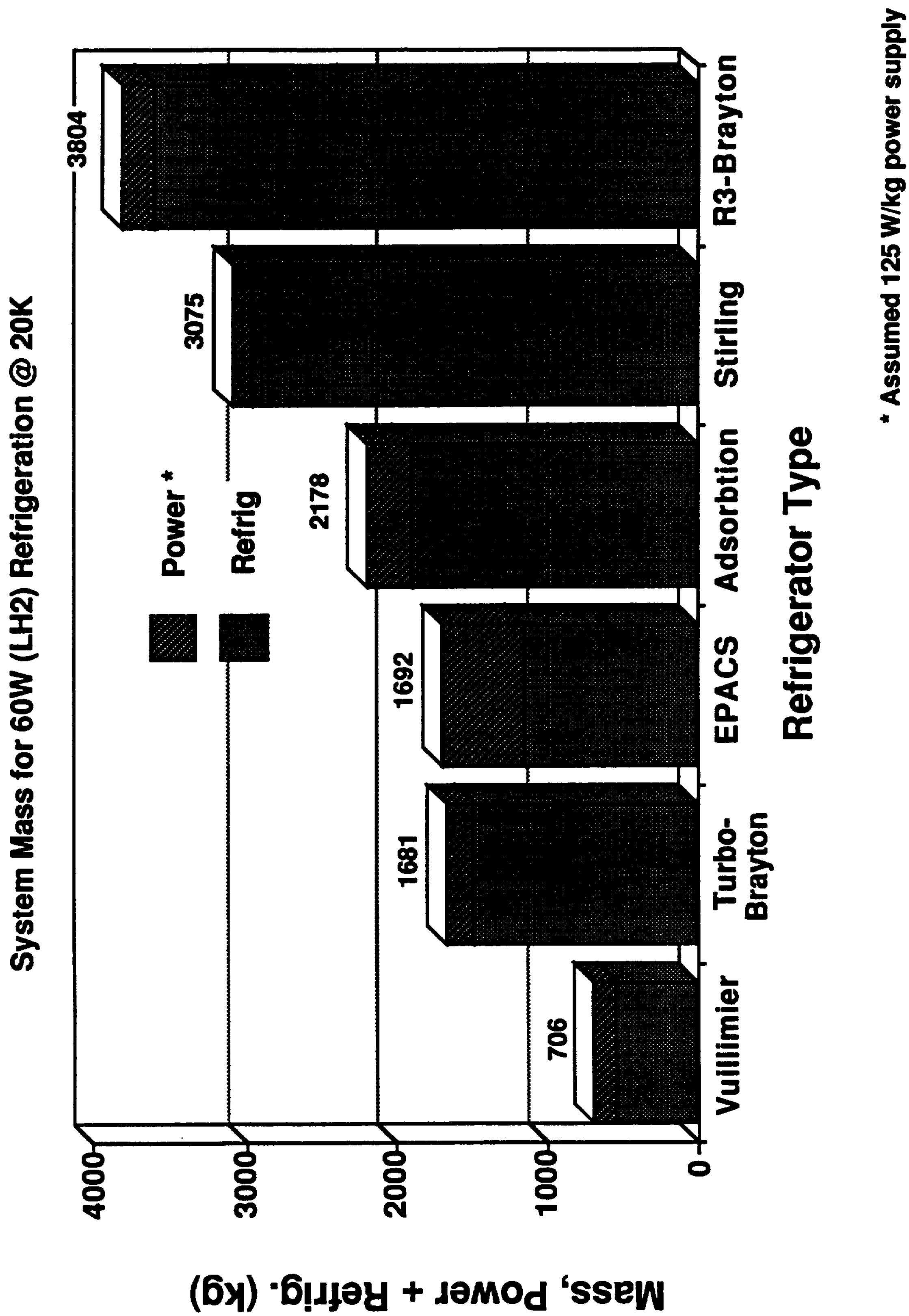


Figure 1.3.7-1 Refrigerator Mass Comparison

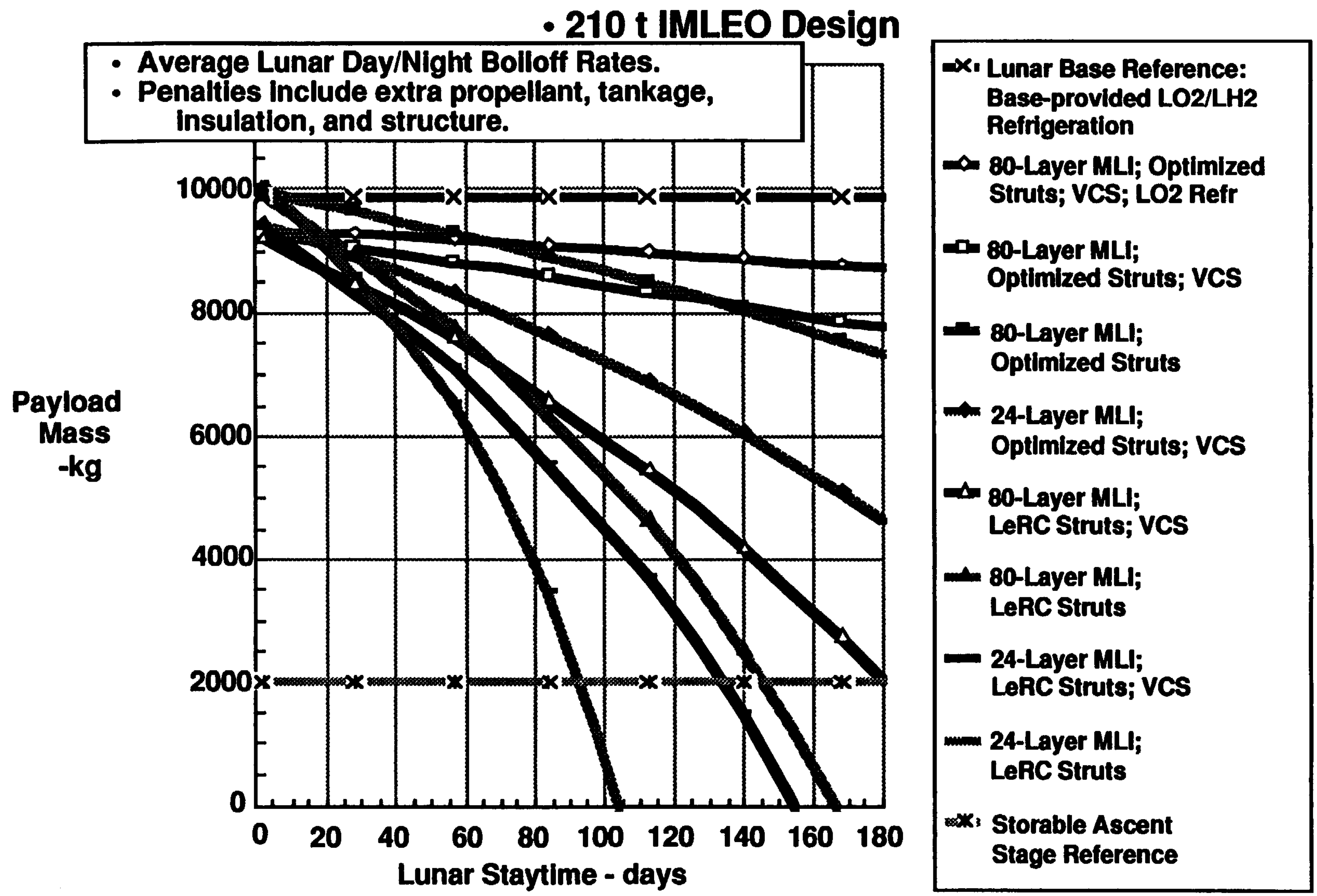


Figure 1.3.7-2 Lunar Boiloff: Piloted Payload Degradation

1.4 STV Study Results and Observations

This section summarizes the study results and compares those results with the "conventional wisdom" prevalent before the study was completed. In many cases the results are somewhat unexpected.

1.4.1 Trade Study Results

a. Mission Performance

(1) Expectations – In any space transportation system, elimination of vehicle inert weight or improvement in engine performance results in increased vehicle performance (measured as the ratio of payload capability to initial total mass). In the STV study phase 1 and phase 2 subsystem and system-level trades, any option that reduced overall inert weight or increased engine performance was expected to result in significant increases in mission performance. The use of higher Isp engines, multiple staging, lunar orbit rendezvous (LOR), and on-orbit vehicle reusability were all thought to contribute to mission performance improvement.

(2) Results – To some extent, the above expectations were proven correct, but with some notable exceptions. Many of the trade options produce only marginal improvements in performance. For instance, a comparison of lunar performance (21 piloted missions, 418 t total lunar cargo) for the phase 2 cost-optimum concept (ground-based, suborbital-staged, 2.5 stages, lunar-orbit direct, single crew module), the best performing concept (Space-based, LEO-assembled, 2.5-stage, lunar-orbit rendezvous, dual crew module), and a poor-performing concept (ground-based, suborbital-staged, 3.0-stages, lunar orbit direct, single crew module, storable ascent propellant) is shown in Figure 1.4.1-1. Also shown is the reference concept with a reduced piloted flight model (10 piloted missions, 418 t cargo), and a reduced cargo model (21 piloted missions, 200 t cargo). The greatest discriminator in overall system performance is the mission model change (25% change in performance), rather than the change in system design (15% change in performance).

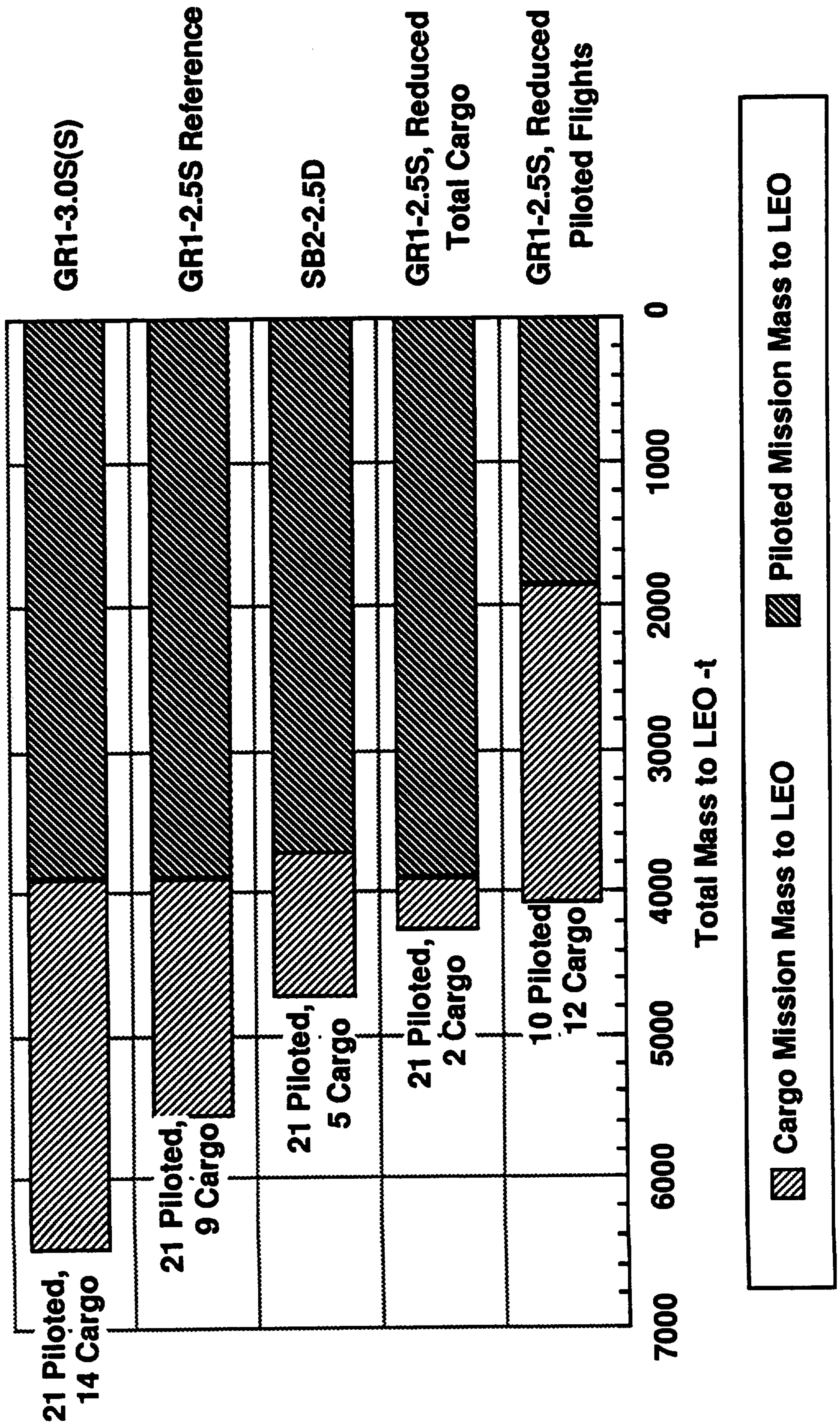


Figure 1.4.1-1: Mission Performance Results

b. Cost Trades

(1) Expectations – The normal expectation of most engineers and managers is that changes in STV candidate design performance and weight would significantly increase or decrease the relative life cycle cost scoring. This perceived notion comes from the knowledge and observation that the majority of hardware descriptive inputs to parametric cost models are generated from mass properties estimates of weight and volume (this is no longer true for avionics or primary propulsion engines; in general, vendor planning estimate throughputs are more accurate.)

(2) Results – The independent STV system cost trade results from the phase I and II cost analysis efforts performed by both STV study contractors (Boeing and Martin Marietta) proved this expectation to be incorrect. Peformance and weight changes for the minimum STV functions (to a consistent set of NASA-specified lunar mission requirements) do not directly equal cost, but changes in technology maturity level selections and hardware part count (flight element and test quantities counts) do significantly change vehicle development cost estimates.

The pie charts in figure 1.4.1-2 depict STV vehicle cost estimate results, in relative constant-year dollars, for two competing STV designs. The subsystems hardware for avionics, life support, primary electrical power, and basic fuel supply tankage (usually the tanks are resized by small changes to diameter or barrel length) normally do not change their function significantly (see pie charts comparison.) The small changes and reallocations of functions to different stage flight elements do not influence the overall life cycle cost (in realtive dollars) more than 5 to 15 percent (most planning estimates of large aerospace programs like STV are only accurate to plus or minus 25 percent.)

Performance limits (and the resultant cost estimate range impacts) are, in reality, dictated more by the mission profiles, mission payload delivery requirements (number and type of sorties), and launch booster Earth-to-orbit capabilities assumed for STV lunar (or other) mission accomplishment than

These two STV designs have similar LCC totals and the same functional requirements, but each phase of the life cycle is different due to system operational changes at the *system interfaces & mission design levels ...*

(Given: Same ETO booster and same total tonnage to the moon)

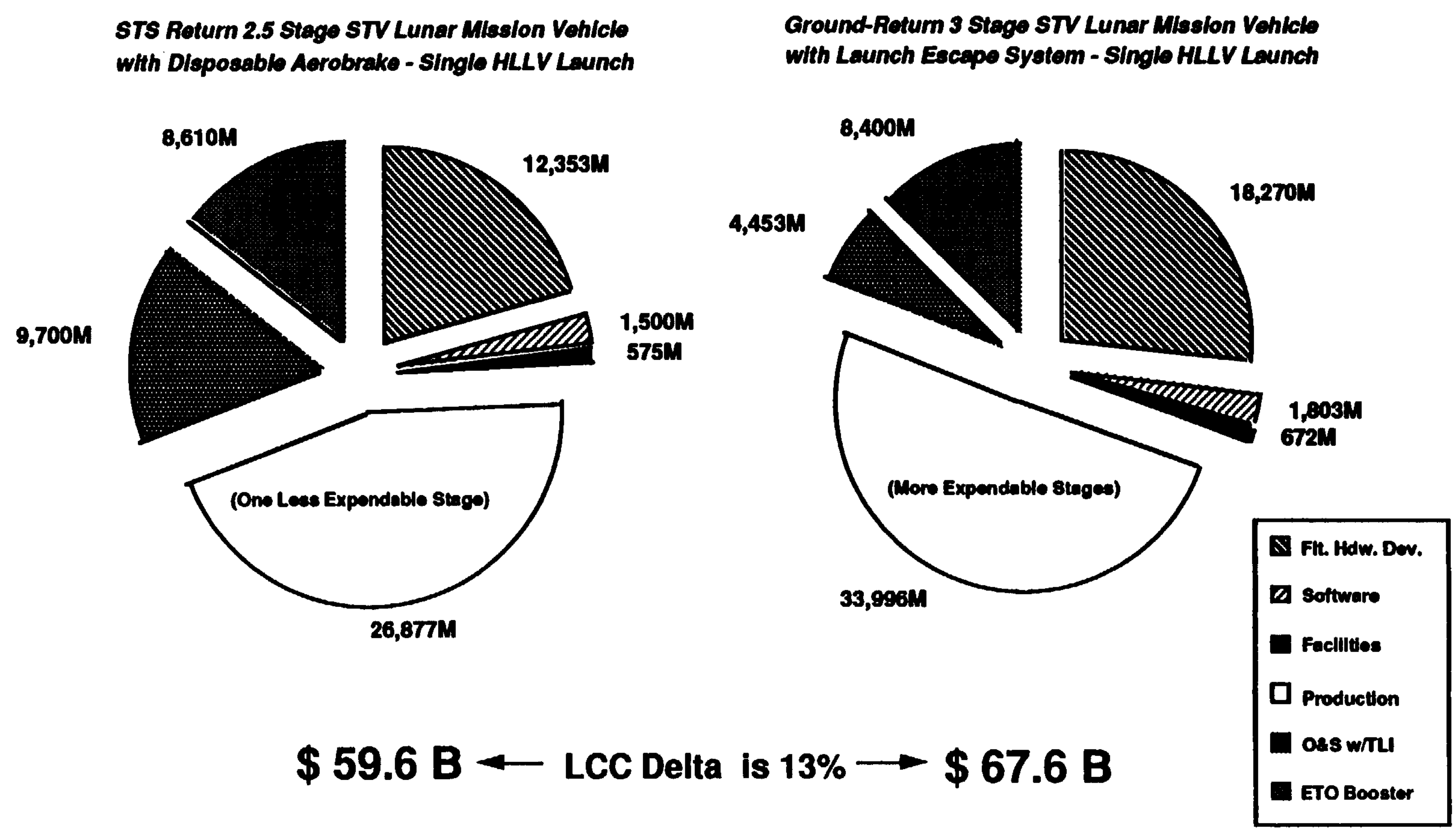


Figure 1.4.1-2: Cost trade results for two competing STV designs (91$)

by varying STV vehicle design characteristics. In the STV study, we found that the system life cycle cost is significantly influenced by basing changes (space or ground), ETO booster changes (number of flights, mix, and size), hardware reusability/expendability, and number of crew module system design decisions (in that order.)

c. Infrastructure

(1) Expectations – The availability of a new large launch vehicle capable of performing a lunar mission in a single launch was expected to provide major operational cost savings to the lunar exploration program. The reductions were thought to come from reduced hardware expended and the reductions in mission planning and operations.

(2) Results – The $ 7B delta system acquisition cost penalty incurred with the 220 mt class booster overwhelmed the operational cost savings for the relatively few missions in the current lunar exploration scenario. The 220 mt booster can be justified relative to the reference NLS booster only if the Mars exploration missions are included in the costing. It probably makes more sense to develop a new low launch cost vehicle which can support SSF with a single launch, the lunar exploration mission with two launches, and leave the Mars mission for the next generation launch system.

d. Risk / Safety

(1) Expectations – Having already flown men to the moon and returned them safely to earth it was expected that risk and safety would not generate major design changes in the Apollo-like STV/LTS concepts.

(2)Results – The need for solar-storm radiation protection and "anytime" abort-capability increased the total weight in LEO by roughly forty tons relative to Apollo, despite improvements in propulsion and structures.

e. Technology

(1) Expectations – A mission as difficult as roundtrips to the moon would surely benefit from performance enhancing technologies such as aerobraking and advanced high pressure engines.

(2) Results – Neither aerobraking nor the Advanced Space Engine (ASE) proved to be cost effective for the relative few missions in the lunar exploration model. The performance gains could not buy back the large DDT&E expenditures over the small number of flights. It would be even worse with discounted dollars.

1.4.2 Evaluation Process Lessons Learned

In the current operating environment, cost will be the primary design driver. Design-to-cost is difficult, but possible, within the context of this type of study. For instance, we found that areas of significant cost change do occur when less hardware is expended in the mission (lower production costs) and the costs of space-basing refurbishment at Space Station Freedom are deleted (saves facilities front end costs and in-space operations labor costs.) Therefore, if the delivery mass and flight rates stay constant for lunar missions, vehicle launch integration and number of element docking/fuel transfer functions increase cost more than optimizing the vehicle subsystems (decreasing engine quantities per vehicle, resizing tanks, reducing redundancy, etc.)

2. STV Phase 1 Summary

Phase 1 of the STV Concepts and Requirements study included STV concepts designed for specific lunar missions but capable of performing other Civil Needs Data Base (CNDB) missions, with few constraints on required ETO capability. This phase of study had two distinct study segments. During the first, a 90-day study, support was provided to NASA in defining a point-of-departure STV. The resulting STV concept was performance optimized with a two-stage LTV/LEV configuration.

After the March 1990 Interim Review (IR#2), the effort was expanded to perform a full architectural trade study with the intent of developing a decision database to support STV system decisions in response to changing SEI infrastructure concepts. Several of the architecture trade studies were combined in a System Architecture Trade Study. In addition to this trade, system optimization and definition trades and analyses were completed and some special topics were addressed. The following summarizes the Phase 1 findings, specifically the relationships between requirements and design characteristics of four reference concepts:

1) 90-day study reference Multi-launch, Space-based, 2.0 stage vehicle.

2) Multi-launch, Space-based, 1.5 stage vehicle

3) Dual-launch, Ground-based 1.5 stage vehicle,

4) Single-launch, Ground-based 1.5 stage vehicle

This section also summarizes the architecture study methodology and trade resuolts.

2.1 Design Driver Assumptions

2.1.1 Available Funding Resources

Funding requirements were not constrained for this period of the study, so life cycle cost became a discriminator between concepts.

2.1.2 Space Program Infrastructure

The infrastructure required by the STV concepts studied during this phase of study includes ground facilities, launch vehicles, and LEO facilities.

a. Ground Facilities – All STV concepts willl use the facilities of and launch from the Kennedy Space Center. Any differences in ground facilities will result from the size of flight elements launched. In all cases, propellant tanks, crew modules, lunar payloads, and vehicle stages must be assembled, checked out, and integrated into the launch stack. Space-assembled equipment must be preintegrated on the ground for fit and function, then disassembled and launched, requiring a unique facility.

b. Launch Vehicles – All STV concepts will require some type of HLLV. The size of the launch vehicle required may vary from a 71 metric ton booster for a multiple-launch, LEO-assembled vehicle to a 250 metric ton booster for a single-launch vehicle. In all of these HLLV configurations, the STV will require propellant fill, drain, and vent (cryogenic hydrogen and oxygen), ground power and thermal conditioning, and telemetry and command feedthrough. Possible launch vehicle options in this phase of study included the Shuttle-C, ALS, and ALS Heavy-lift derivatives, shown in Figure 2.1.2-1.

c. LEO Interfaces – The requirements for LEO infrastructure are the most significant differences between vehicle concepts studied. Space-based vehicles have extensive LEO interface requirements, including a node that will have the capability to assemble the vehicle, such as Space Station Freedom. As such, the node will have to be able to provide propellant handling and conditioning capability as well as extensive vehicle refurbishment capability. A decription of the proposed SSF facilities functions required is shown in Figure 2.1.2-2.

2.1.3 Available Technology

a. Initial STV Technology Survey – The technologies applicable to STV lunar mission applications were researched and the survey results were initially presented at the phase I STV study Interim Review (IR) #2 briefing. The survey included military (U.S. Department of Defense) hardware and software technology data sources, as well as European and NASA technology data sources. The emphasis at IR#2, and

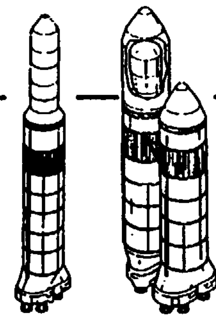
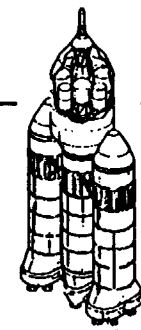
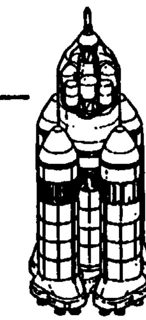
Launch Vehicle Program Element	SB  BASELINE ALS	GO  ALS / SEI GROWTH OPTIONS	GB 
ALS Vehicle DDT&E		• Core Tank Strengthened • Second Booster added	• Core tank redesign: Size & strength • Boosters stretched & strengthened
Manufacturing		• Increase Tank manufacturing capacity • Increase P/A refurb capacity	• Increase tank manufacturing capacity • Increase P/A refurb capacity
Stack & Assembly		• Add 1 vehicle integration cell • Add 1 cargo integration cell	• Design & build a larger vehicle integration cell: • Add 1 cargo integration cell
Transport		• Design & build 2 man rated MLPs with access tower	• Design & build 2 large man rated MLPs with access tower
Launch Pad			• Design & build new pad: Vehicle size exceeds current requirements

Figure 2.1.2-1 Launch Vehicle Options

Mission Description

- Space Transfer Vehicle based at SSF
- Five launches / Lunar mission
- 71t HLLV ETO capability
- Wet droptanks stored at SSF
- Aerobrake assembly/vehicle integration at SSF
- Crew delivered to SSF on STS
- Space tug transfers payloads to SSF
- Propellant depot at SSF for core tanks
- Other CNDB/DoD missions conducted from SSF

Space Station Freedom

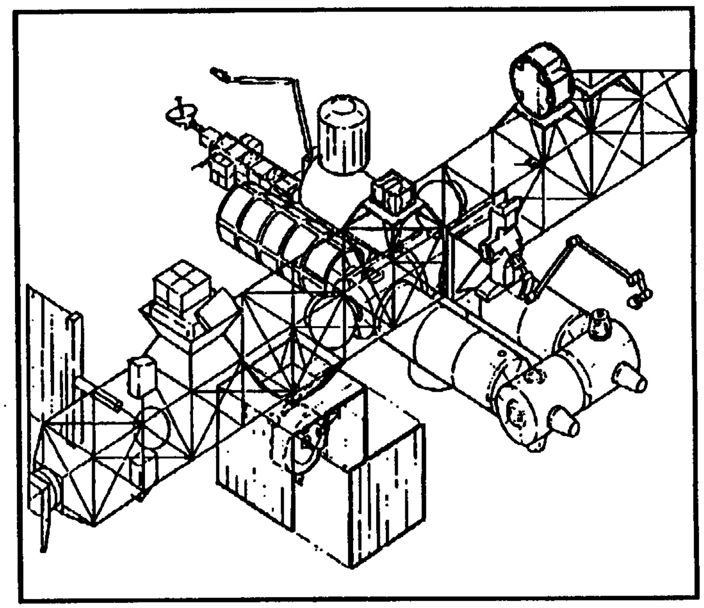


Figure 2.1.2-2 SSF Facilities

subsequent IR briefings, was to select technology areas of innovation which improved LTS vehicle survivability, stay time, fluid and docking capabilities, vehicle operation autonomy (flight and housekeeping/test), and also improved crew comfort and safety.

b. STV Technology Summaries – Later STV study technology presentations and studies attempted to provide the customer with technology development and application schedule plans. The Boeing, Martin, NASA study team representatives went even further at the end of phase I to refine the STV/LTS/Mars Transportation System technologies requirements subject matter into an overall technology plan for MSFC and the NASA level II Space Exploration Initiative (SEI) Office of Technology at Johnson Space Center, Houston, Texas. Mr. Fred Huffacker of NAS MSFC managed the joint technologies summary activities in concert with the STV study office at MSFC.

A sample of these standardized forms for the level II office input from the space transportation system studies completed in 1990 is presented in Figures 2.1.3-1 through 2.1.3-6. These forms were completed by the cooperative NASA/contractors team for all critical STV subsystems design and technology application areas. The real bonus is that they provide the space transportation technical community with a good consensus-generated summary of forecasted technology requirements. The technology data was generated by consensus from four STV-related (and yet independent) NASA study sources and design preference points of view.

2.1.4 Lunar and Non-SEI Mission Requirements

Specific mission requirements including SEI requirements are given in Figure 2.1.4-1, and include piloted and unpiloted missions drawn from the 1989 Civil Needs Database (CNDB). Of the 10 mission types, 3 are piloted and must return the crew.

SEI / Space Transportation Systems Planning

Program Title: **<u>Space Transportation Advanced Development</u>**

<u>WBS: 9.2.1.4 - Vehicle Avionics</u>

<u>Tasks:</u>

(1) Analysis of system requirements and space electronics system standards.

(2) Develop and manage SEI Avionics and Software interface requirements & specifications database.

(3) Incorporate new, autonomous GN&C functions.

(4) Establish approach for major subsystem elements.

(5) Test and demonstrate critical components and subassemblies to readiness level 6 before STV Preliminary Design Review (PDR): January, 1998.

(6) Perform system simulations, ground testing, and in-space test (if relevant) for operational proof.

(7) Establish SEI Avionics/Software National Testbed.

Figure 2.1.3-1: Overall Space Exploration Initiative Plan Includes STV Requirements

Vehicle Avionics Technology/Advanced Development

Description: The vehicle avionics architecture selected for the Lunar Transfer Vehicle is based on photonics technologies with advanced software and advanced sensors. Current state-of-the art alternatives use a multi-bus wire network with conventional wire connectors, Large Scale Integration (LSI) avionic circuits, and custom integrated circuit chips. The photonics architecture utilizes ASIC chips, Silicon-on-Saphire (SoS) chips, fiber optic components, lower power systems, pooled spares, and fault-tolerant components.

Application/Benefit: The application of advanced photonics and advanced sensors will improve space transportation system safety, reliability, availability, and improve vehicle turnaround time. All of these improvements relate to a lower life cycle cost and better performance. The benefit requires ample development funding and a national commitment from the funding source. Avionics refurbishment is a high value cost item in the vehicle operations and support phase.

Current Status: The technology readiness level (TRL) on a 1 to 9 scale is currently at level 4. The photonics hardware has been developed for advanced U.S. military vehicles. Current proprietary programs at various contractors are under way to convert the military technology advancements to space transportation system product development projects. The proprietary programs are attempting to reach a level 5 TRL for space transportation vehicles by 1995, with new software and critical fiber optics connector components.

Technology Needs/Plan: The following pages contain a five-year STS avionics plan for development of the photonics-based architecture with advanced navigation instruments, application-specific integrated circuits (ASIC's), fault-tolerant subsystems, standardized network interfaces, neural networks, and sensor fusion developments. The total dollars include $ 42.5M for the establishment of U.S. national testbeds.

Advancement in computer technology is needed to meet the autonomous navigation control and in-flight operational maintenance requirements for Lunar and Mars STS missions. The component development lead times (30-36 months) for enabling ASIC , fiberoptic components and embedded software technologies will require architecture design requirements to be solidified by the year 1993. ASIC and photonics hardware will be needed for a credible Lunar Transportation System test program within the late 1990's time frame (assuming an operable LTS by the year 2002.)

Figure 2.1.3-2: Specific LTS Avionics Technology Status is Described

Vehicle Avionics Technology/Advanced Development

SEI / STS Technology Category	Current Tech.	*TRL	Selected Tech.	Rationale for Selection	Priority
1. Application Specific IC's	GaAs VLSI	5	GaAs/SoS Wafer	Radiation, integration flexibility; high rel.	Higher
2. Fiber Optic Sensors	50% of Qty.	4	100% Photonics	Lower EMI; high rel.	Higher
3. Neural Networks	None	3	Multi-path Netw.	Autonomy; safety	High
4. Navigation Instruments & Autonomous GN&C	Hexad RLG R/R R. Radar	5 3 6	Fiber Opt. Gyro, Sensor Fusion Syn. Aper. Radar	High rel.; lower LCC; more fault tolerant; improved landing	Highest
5. Digital Data Buses	Med. Speed	6	High Speed	High perf.; safety	Higher
6. Sensor Networks	Multi-bus	4	Network I/F Unit	Reduced paths/LCC	Highest
7. Standard Interfaces	Wiring	2	IR/Fiber Opt. I/F	Reduced complexity; less EMI, corrosion.	Highest
8. Vehicle Health Management	50% Appl.	4	100% Coverage	Lower LCC, refurb.	Higher
9. Fault Tolerant Avionics	Quad. String	4	Voting System, Photonics	High rel.; lower LCC; parallel processing; radiation resistant	Highest
10. Communication & Tracking	TDRSS	6 ,7	ATDRSS, Laser Communication	Radiation tolerant; high speed; secure	High

* Note: Overall mean (average) "Technology Readiness Level" (TRL) is about 4.5, on a scale from1 to 9.

Figure 2.1.3-3: LTS Avionics Technologies Listing is Documented and Prioritized

C-2

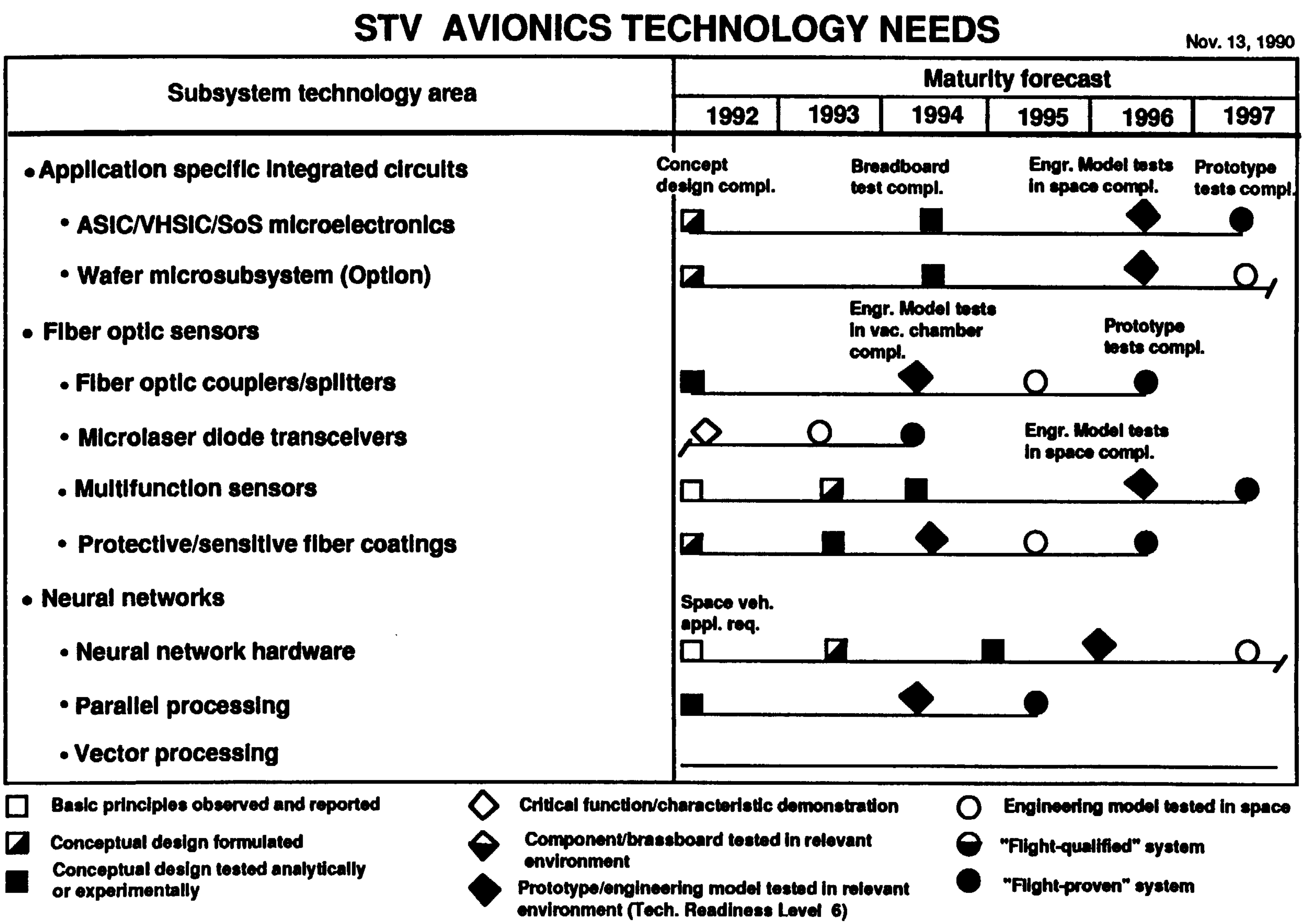


Figure 2.1.3-4: LTS Avionics Technology Development Schedules are Postulated

- **Preliminary requirements:**
  - **(1) Low-Earth orbit communications to Earth and Moon bases**
  - **(2) TDRSS/SSF/SGLS/DSN communication interface capability**
  - **(3) Secure transmission capability**
  - **(4) Emergency band capability during Sun storms**
  - **(5) Backup systems capability**
  - **(6) Redundant strings for reliability**
  - **(7) EVA/FTS/SSF secure communication links**
  - **(8) Compatibility with GPS and Russian systems**
  - **(9) Omnidirectional capability for T&C during anomalous state**

Figure 2.1.3-5: Preliminary LTS Communication Requirements Help Define Need Areas

1995 – 2000 Projections:	Forecast		
	Engineering models	Space prototype	Fully qualified and operational
Common pressure vessel NiH_2 backup batteries	1990	1993	1995
Microwave filter technology	1992	1995	1997
Solid state power amplifiers (GaAs FETs*)	1989	1992	1995
Laser diode array without telescope	1988	1992	1994
Phased array beam, packet switching	1994	1997	2000
On-board processing communication (low to hi-bit converter)	1995	1998	2001
Multibeam connectivity	1992	1995	1998
EHF interface systems (MILSTAR)	1987	1992	1996
UHF follow-on system interface	1990	1992	1993
Printed circuit radiator technology (MMIC**)	1988	1994	1997
Improved demodulation systems	1991	1994	1996
On-board switching – TST switch (ACTS)	1986	1992	1995
Speech/communication bit rate conversion	1992	1995	1999
Increased orbital design life – 15 years	1994	1997	2000

* Field effect transistor
** Monolithic microwave integrated circuit

(Reference: Proceeding of the IEEE; July, 1990 Special Issue)

Figure 2.1.3-6: Specific Space Communication Technology Forecasts are Documented

Assumptions:

- All unpiloted missions, except polar platform servicing (S1) for space-based, are delivery-only.
- 1989 Civil Needs Data Base, excluding Mars missions, as follows:

		Delta V (m/s)		Deliv.	Return
	Mission	Delivery	Return	Cargo (kg)	Cargo (kg)
N1	Nuclear Debris Disposal	4175	0	25000	0
G1	GEO Delivery	4207	0	10000	0
P1	Planetary Delivery	4451	0	16000	0
D1	Molniya Delivery	4499	0	6800	0
S1	Polar Platform Servicing (GB)	6356	0	4500	0
S1	Polar Platform Servicing (SB)	6356	3428	4500	4500
L4	Lunar Cargo Delivery	6390	0	34000	0
C1	Capsule Recovery - Piloted	2696	40	0 **	500 **
G2	GEO Servicing - Piloted	4202	1862	4000 **	4000 **
L3	Lunar Excursion - Piloted	6390	2913	12500 **	500 **

** Excluding Mass of Crew Module

Figure 2.1.4-1 Mission Requirements

2.2 90-Day Study LTV/LEV Reference Concept

The 90-day study reference concept consists of two major elements, as shown in Figure 2.2-1. One portion, a reusable cryogenic transfer vehicle with a reusable aerobrake, crew module, and two pairs of expendable drop-tanksets, is based and refurbished at the LEO node. The other portion, a reusable lunar excursion vehicle, with crew module, is based in low lunar orbit (LLO). Both transfer vehicle and excursion vehicle have four main engines, allowing engine-out capability during all mission phases.

2.2.1 System Design and Operation

For initial piloted missions, the transfer vehicle, excursion vehicle, aerobrake, and crew modules are launched to the space station or LEO node aboard several heavy-lift launch vehicles, assembled, and the transfer vehicle is fueled from a propellant depot. The drop-tanks are launched fully loaded aboard two heavy-lift launch vehicles, integrated with the transfer vehicle, and then the crew and cargo are launched aboard a shuttle to the completed stage. The transfer vehicle returns to the LEO node after each mission, where it can be used for subsequent lunar missions or for other non-lunar missions.

The lunar mission sequential configuration of the vehicle is depicted in Figure 2.2.1-1. In a steady-state mode, after leaving the Space Station, the TLI drop-tanks are jettisoned following the TLI burn, and the vehicle performs a lunar-orbit insertion (LOI) burn to circularize into a circular lunar orbit. The transfer vehicle then performs a rendezvous with the lunar excursion vehicle and refills the LEV tanks from the second set of drop-tanksets on board. The crew enters the excursion crew module, the excursion vehicle separates from the transfer vehicle, and descends to the lunar surface.

Upon arrival on the lunar surface, the cargo is unloaded, the vehicle is hooked up to lunar surface support equipment, and the crew moves to the lunar habitat for the lunar stay. At the end of the lunar stay, the crew loads return cargo and boards, checks out the vehicle, then the excursion vehicle ascends and returns to LLO, where it rendezvous' with the transfer vehicle. The crew moves back into the transfer crew module, the stages separate, and the transfer vehicle jettisons the second pair of drop-tanksets and returns to Earth orbit, utilizing an aeromaneuver to insert into LEO and

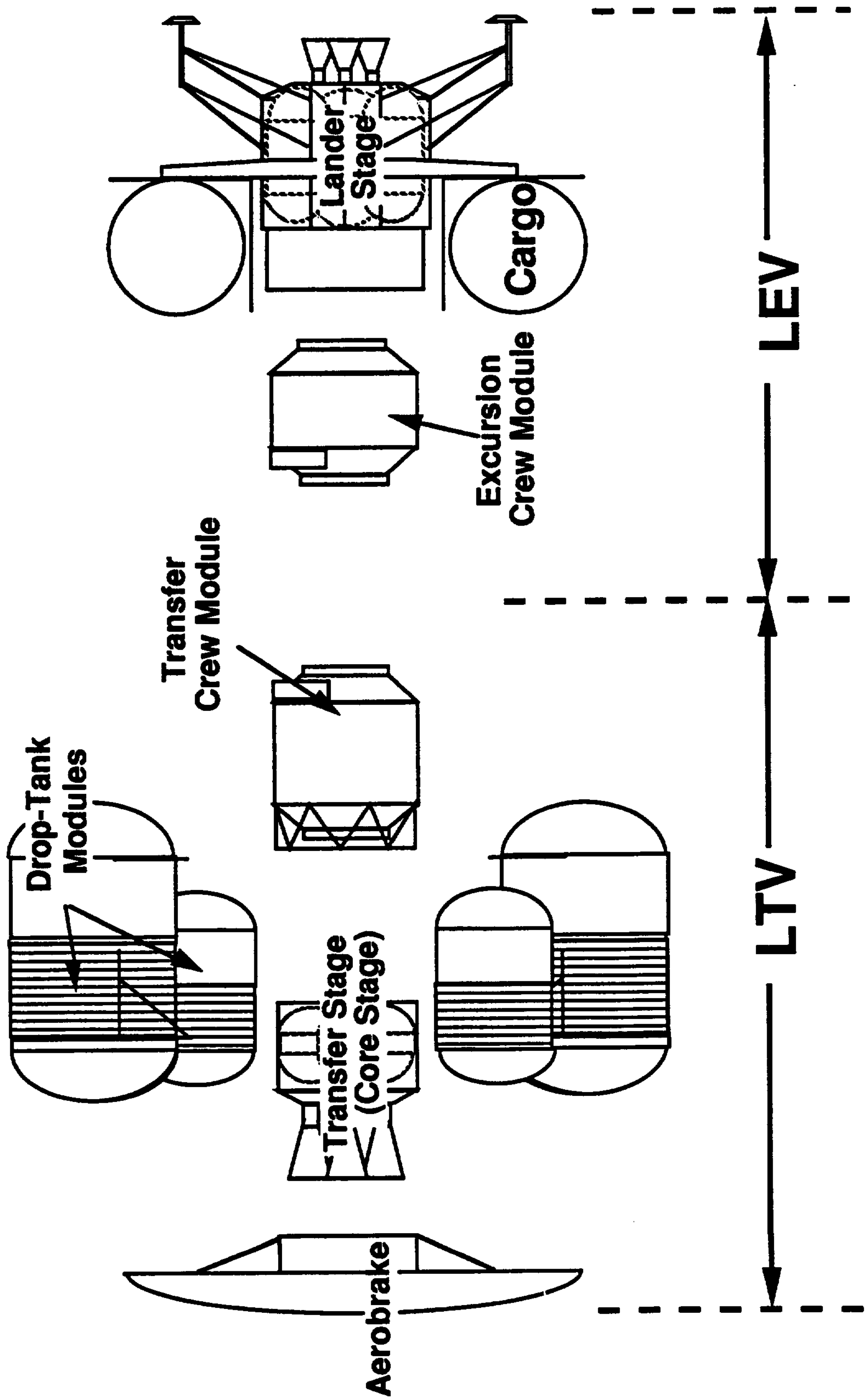
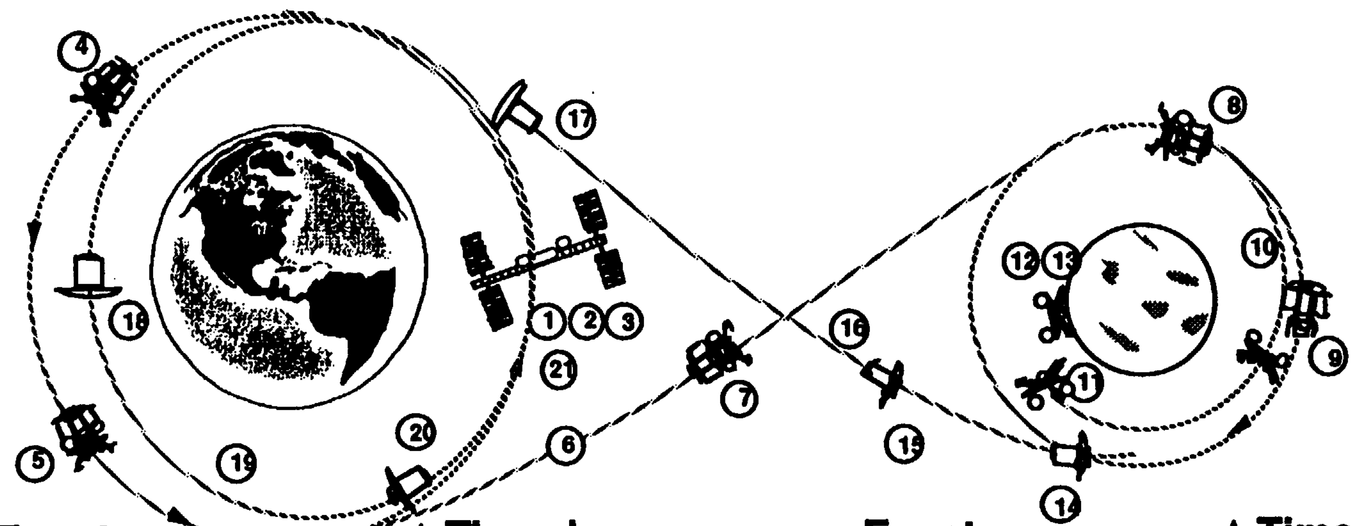


Figure 2.2-1 Reference 90 Day Study Configuration



Event	Δ Time, hours	Event	Δ Time, hours
1) Vehicle assembly	4 weeks	12) Landing	0.0
2) Payload Integration	2 weeks	13) Cargo offload	1.0
3) Separate from SSF	0.0	14) LTV TEI burn	0.1
4) Coast	0.8	15) Trajectory correction maneuvers	0.1
5) TLI burn	0.3	16) Transit/coast	84.0
6) Trajectory correction maneuvers	84.0	17) Pre aero correction maneuver	0.1
7) Transit/coast	0.1	18) Aeromaneuver	0.1
8) LOI burn	0.1	19) Post aero correction maneuver	0.1
9) LEV/payload separation	0.0	20) Earth orbit circularization	0.1
10) LEV deorbit burn	0.05	21) Rendezvous/docking with SSF	1.0
11) Descend to Lunar surface	2.0		

Figure 2.2.1-1 Lunar Mission Sequence

rendezvous with the Space Station, where it is inspected and refurbished for the next flight.

For the unpiloted mission the transfer vehicle with an expendable excursion vehicle is flown without crew module and aerobrake on a one-way delivery, and the excursion vehicle is left on the lunar surface with the descent tanksets after landing.

2.2.2 Flight Element Description

a. Lunar Transfer Vehicle – The transfer stage has an external load-bearing body structure with structural interfaces to the crew module, aerobrake, cargo, and drop-tanks. The body structure is a cylindrical graphite/epoxy honeycomb structure with internal stabilizing rings which forms the backbone of the structure. An aft thrust structure of graphite / epoxy design distributes thrust loads from the main engines to the vehicle and resists lateral engine gimbal loads. The thrust structure has four engine mounting pads and associated thrust vector actuator supports and includes struts for lateral load stabilization.
The transfer stage includes one LO2 tank and one LH2 tank with elliptical end-domes and associated propellant aquisition devices.

(1) Drop-tanksets – The current space-based tanksets include Aluminum-Lithium main tanks with composite honeycomb sandwich intertanks that distribute launch loads from an aft launch vehicle interface ring and graphite / epoxy transfer vehicle interface trusses that permit on-orbit transfer vehicle integration and provide structural support during the mission. Both TLI and LOI drop-tanksets are integrated about the sides of the transfer vehicle below the aerobrake and require graphite / epoxy support trusses with titanium fittings, sized for transfer and orbit insertion loads, as well as deployment and release fittings for expending the empty tanks.

(2) Aerobrake – The space-based aerobrake is a rigid space-assembled shell structure of high-temperature graphite/polyimide sandwich panels affixed to a system of graphite/polyimide longerons and frames and covered with high-temperature thermal protection ceramic tiles. The aerobrake is

launched in a folded position, as shown in Figure 2.2.2-1 and assembled and integrated at the Space Station.

The graphite polyimide construction allows the aerobrake structure to run hotter than would be allowable with an aluminum structure (650° F for GR/PI vs. 350° F for aluminum). Along with this higher temperature capability, the thermal expansion of the graphite polyimide can be tailored to match that of the overlying ceramic TPS, eliminating the need for a strain-isolation pad under the tiles.

(3) Crew Module – The transfer crew module consists of a pressurized primary shell with internal bulkheads and partitions, windows for docking maneuvers, and two hatches for EVA and crew transfer. The crew module design makes use of SSF technology and design, and is 4.45 m diameter, 3.96 m long, and includes 37.6 m^3 free volume.

b. Lunar Excursion Vehicle

(1) Lander – The excursion stage has an external load-bearing structure with structural interfaces to the crew module and cargo. The body structure includes twelve major longerons, a series of interior stabilizing struts, forward and aft stabilizing struts, and exterior closeout panels. The graphite/ epoxy longerons transfer primary loads and form the backbone of the structure. Eight of the longerons include interface fittings for the landing gear and cargo attachment The stabilizing struts are graphite / epoxy struts of varying lengths and sizes with titanium end fittings. The exterior and lower closeout panels are sandwich panels with honeycomb core and graphite / epoxy face sheets. These panels provide structural stiffness as well as shielding for the excursion vehicle tanks.

The thrust structure is of graphite / epoxy design, with four engine mounting pads and associated thrust vector actuator supports. It also includes struts for lateral load stabilization. Lunar landing gear includes four sets of landing gear that are deployed during initial replacement flight lunar transit and left deployed while the vehicle is stored in LLO, ready for the next mission.

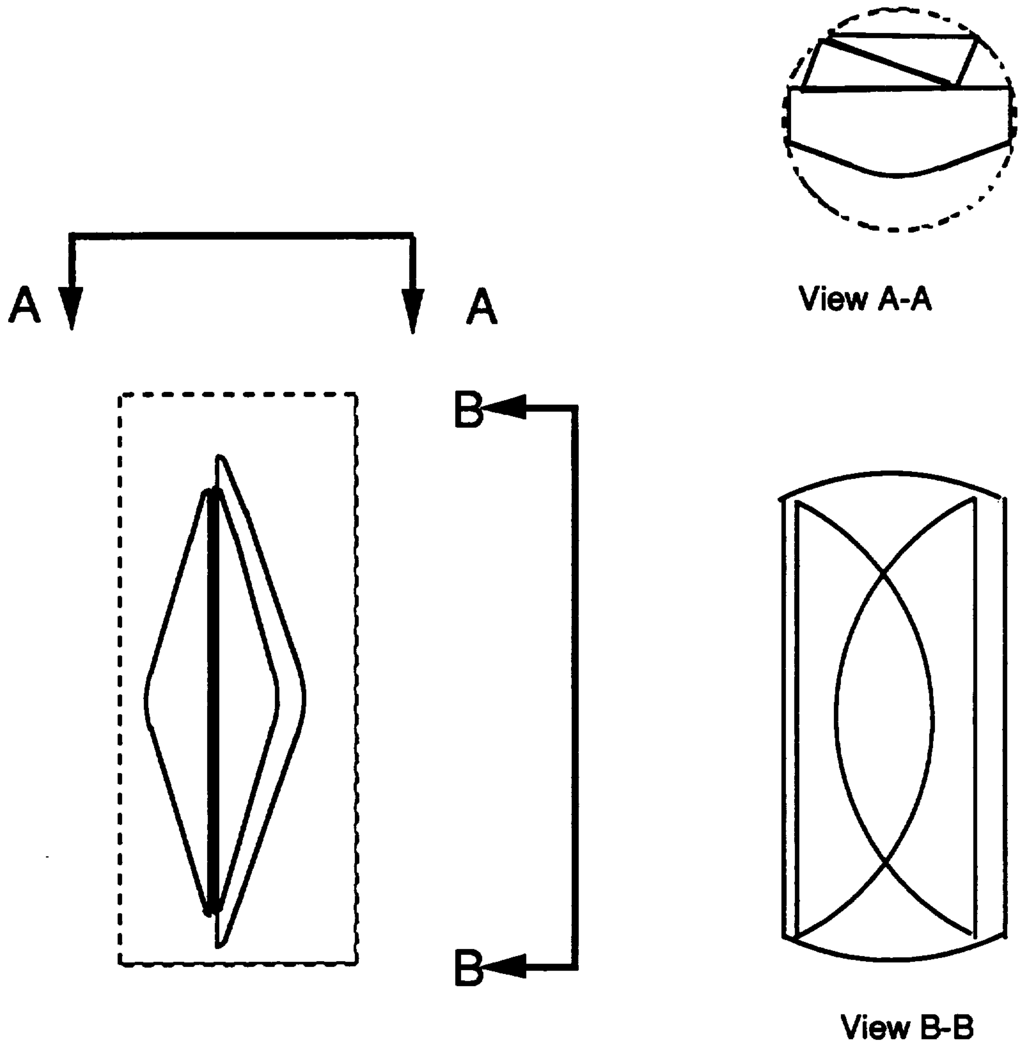


Figure 2.2.2-1 Aerobrake Packaging for Launch

This page intentionally left blank

The excursion vehicle tanks are made of Aluminum-Lithium and include a single LO2 tank in the center, and four LH2 tanks around the periphery, each with associated propellant aquisistion devices. The tanks contain enough propellant for the lunar landing and lunar ascent to LLO.

(2) Crew Module – The excursion crew module consists of a pressurized primary shell with internal bulkheads and partitions, windows for landing and docking maneuvers, and two hatches for EVA and crew transfer. The design of the crew module makes use of SSF technology and design, is 4.45 m diameter, 2.8 m long, and includes 22.8 m^3 free volume.

2.2.3 Subsystem Description

a. Main Propulsion – The selected space-based STV main propulsion system is a LO2/LH2 system and uses four advanced expander-cycle engines with a vacuum thrust of 15,000 lb per engine, and an assumed specific impulse of 481 seconds. It includes the engines with electromechanical actuation, as well propellant delivery, pressurization, fill, and vent systems.

b. Reaction Control – The reaction control system is a gaseous O2 / gaseous H2 system with an assumed specific impulse of 410 seconds. It includes four GO2/GH2 thruster modules on each stage and associated accumulators, pressurization, and control.

c. Electrical Power – The electrical power system features redundant O2/H2 fuel cells fed from accumulators filled from the vehicle main propellant tanks, as well as distribution and control units and associated wire harnesses.

Fuel cell reactants are drawn from accumulators included in the Reaction Control Subsystem. The redundant accumulators are sized to provide oxygen and hydrogen reactants for both RCS and EPS functions for a period of time needed to fill the other accumulators. Once filled, the reactants are isolated and heated to supercritical pressure. Reactants are then drawn off to supply the fuel cells through a system of CRES manifolds.

For peak power loads during main engine actuation, three rechargeable Nickel-Hydrogen batteries are included in the power supply to supplement fuel cell power. The batteries are sized to provide a total of 5.0 kilowatt-hours of power to the main engine actuators.

The power distribution system consists of power distribution and control assemblies, inverters, and remote switching devices that interface with other vehicle subsystems and external power supplies.

d. Avionics –

(1) Guidance and Navigation – Provisions for lunar mission operations, including rendezvous, docking, and lunar landing, with built-in redundancy for piloted operations.

(2) Communication and Data Handling – Provisions for communication, vehicle health maintenance, and data handling, with audio/video interfaces for piloted operations and instrumentation for drop-tank monitoring and control.

(3) Displays and Controls – Provisions on the crew modules for limited crew control and status monitoring of the vehicle during critical phases of the mission.

e. Environmental Control / Life Support – Environmental Control and Life Support (ECLSS) includes provisions on the crew module for atmosphere supply and control, internal equipment cooling, as well as metabolic and equipment heat rejection. Figure 2.2.3-1 shows a life support hardware schematic similar to the Orbiter system that meets the requirements of all STV configurations. The schematic reflects the fault tolerance levels required for critical equipment, with triple critical system components rather than separate triple systems.

The system is an open loop life support system, with no regeneration of either atmosphere or water. ECLSS functions, as well as the relationship to other vehicle functions is given in Figure 2.2.3-2. Since an adequate supply of water is provided as

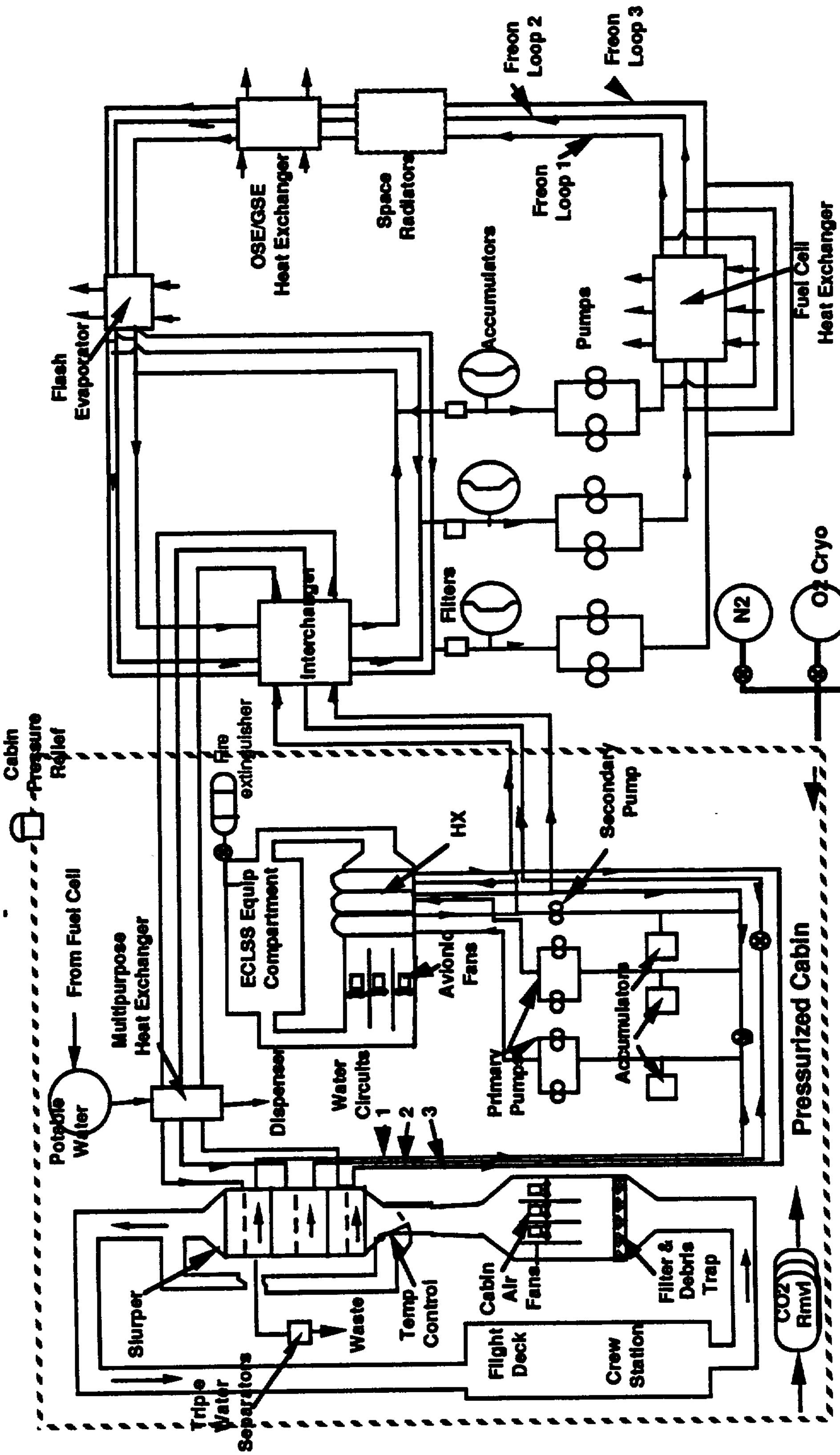


Figure 2.2.3-1 ECLSS Hardware Schematic

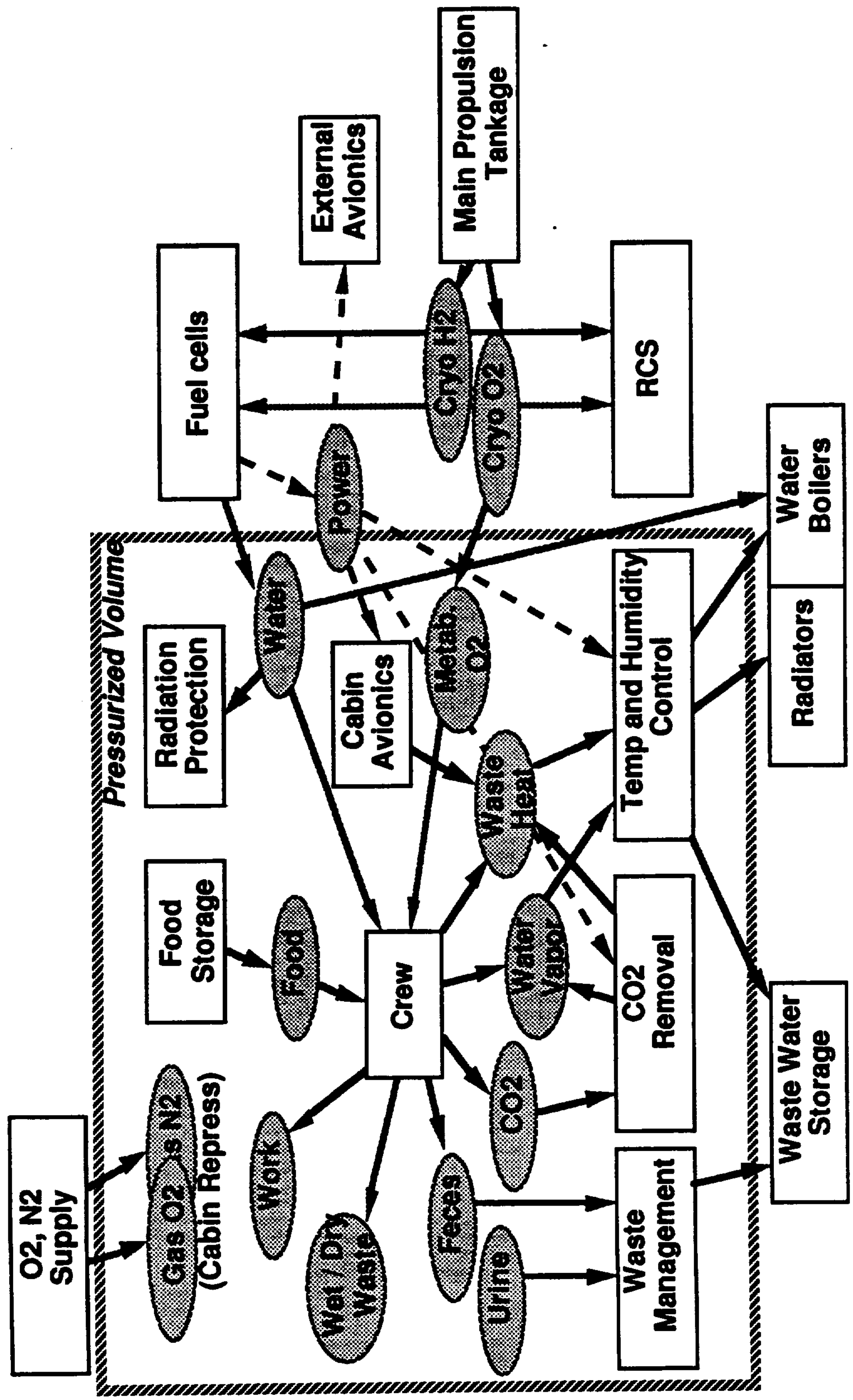


Figure 2.2.3-2 Crew Module Subsystem Interaction

by-product of the fuel cell power supply system, only minimal water stores and supply tanks are required for STV, and recovery of cabin humidity condensate is not required. Atmospheric gases are supplied from storage and from the fuel cell reactant supply accumulators, and carbon dioxide is removed from cabin air by replaceable LiOH canisters.

An active thermal control (ATC) loop is incorporated into the environmental control system, with coldplates for electronic equipment cooling, a cooling water loop for cabin thermal control, a Freon loop to cool vehicle heat loads, various equipment heat exchangers, and a variety of heat rejection devices designed for specific mission phases. Heat rejection devices include ground support equipment (GSE) heat exchangers, water flash evaporators, and space radiators. Prior to launch, heat is rejected through a GSE heat exchanger. During launch, a passive thermal sink for initial liftoff and a water spray boiler for above 140,000 ft are employed until the vehicle separates from the launch vehicle, after which triple-loop metallic radiators are deployed to reject heat. The water spray boilers may also be used to supplement the radiators during peak in-space heat load periods.

f. Personnel Provisions – Personnel provisions include food, water, and waste management systems, as well as fire detection and crew furnishings. The food management system provides for the storage, preparation, and preservation of food for the crew. The food is shelf-stabilized and is prepared using warm water and heated in a convection oven, similar to the shuttle.

The water management system provides for potable water during the mission duration, and includes a water storage tank with water drawn from the fuel cell by-products, water dispenser, as well as tanks with a contingency water supply. The waste management system for both space- and ground-based vehicles includes a partitioned zero-g commode / hygiene station with waste storage tank and pre-moistened wipes for personal hygiene.

Crew furnishings include flight seats, emergency medical / health provisions, and personal equipment storage provisions. The flight seats, similar to those on the STS Orbiter, provide restraint and impact attenuation for all phases of flight and can be removed and stowed during flight .

2.2.4 Performance

The reference vehicle concept can either deliver 13,800 kg cargo to the lunar surface in a steady-state piloted mode or 42,300 kg in a cargo delivery mode. With this cargo split, a total of 418 tonnes of cargo is delivered to the lunar surface over 21 piloted and 4 cargo-only missions. The TLI drop-tanksets used in the cargo-delivery mode are slightly larger than those for the piloted mode to take advantage of extra launch vehicle capability.

Misson performance of the two stage 90-Day study reference vehicle in both a cargo-delivery and steady-state piloted mode is given in figure 2.2.4-1.

	REFERENCE PILOTED STEADY-STATE		REFERENCE CARGO DELIVERY	
<u>SUMMARY MASS</u>	<u>179302</u>		<u>190888</u>	
CREW / CARGO		14568		42300
CREW / SUITS	768		0	
LUNAR CARGO	13800		42300	
TANKSETS		141988		142024
TANKSET #1, #2	3808		3733	
LOI PROPELLANT	17509		14	
LEV PROPELLANT	23039		37260	
TEI PROPELLANT	5225		0	
TANKSET #3,#4	6033		6110	
TLI PROPELLANT	86374		94907	
LTV (REUSABLE)		13368		0
AEROBRAKE	2823		0	
TRANSFER CAB	5824		0	
PROPULSION MODULE	4721		0	
LEV		9378		6564
LANDER	5797		6564	
EXCURSION CAB	3581		0	
<u>SEQUENTIAL MASS</u>				
LAUNCHED MASS	156556		190888	
ASSEMBLED MASS IN LEO, LTV	169924		190888	
LLO ARRIVAL, LTV	61296		190888	
START LUNAR DESCENT, LEV	46985		86124	
LUNAR LANDING, LEV	30230		49600	
LUNAR TAKE-OFF, LEV	16803		N/A	
START TEI, LTV	19972		N/A	
AEROMANEUVER, LTV	15747		N/A	
EOM: LTV AT SSF	14710		N/A	
EOM: LEV AT LLO	9378		N/A	

Figure 2.2.4-1 Reference Vehicle Performance

2.2.5 Program Cost Estimates

Preliminary STV study estimates of the two stage 90-Day Study reference vehicle were based on conceptual design definition information contained in figure 2.2.5-1.

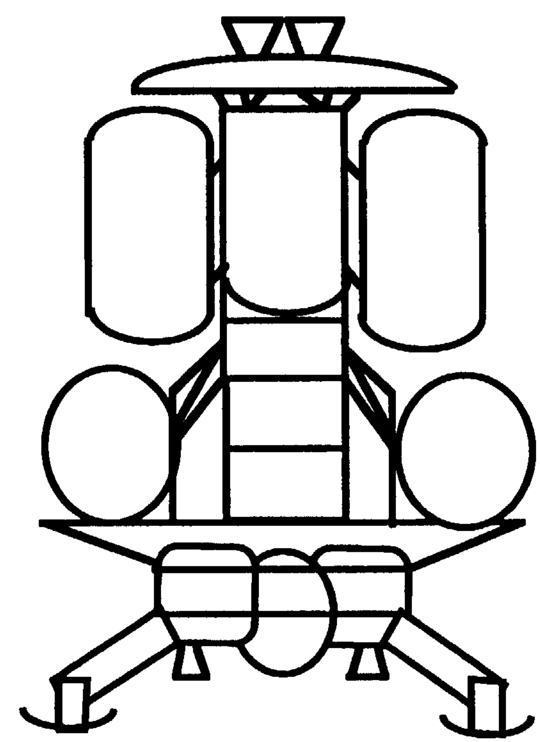
A Space Exploration Initiative level work breakdown structure (WBS) was provided to the study contractors in order to develop a complete, indentured WBS dictionary for STV Lunar Transportation System (LTS) mission and other STV mission life cycle cost estimates. The WBS dictionary (ref.: Boeing STV Phase I Final Report, Volume III, Book 2) was tailored to be expandable to handle STV/LTS designs of one or more stages. Figure 2.2.5-2 illustrates the summary, initiative level WBS provided by NASA used to develop LCC estimates.

Figure 2.2.5-3 represents the only specific summary cost data on the reference vehicle presented during the study. The figure pie chart shows the areas of technology emphasis based on the ratio of each subsystem area to total hardware development cost. Later trade studies provide MSFC with a summary LCC estimate for the reference vehicle. The summary LCC for the reference vehicle was presented by the Boeing team at the IR#3 briefing (ref.: Volume II, page F-14) in June of 1990. The Figure 2.2.5-4 Crew Module Trade comparison chart contains the reference LCC summary. The reference vehicle LCC summary is the second bar to the right of the origin. The LTS 90-Day Study design reference vehicle system for STV was coded **SB2-2.5D** because it is: *space-based* (**SB**); uses *lunar orbit rendezvous* (**2**); has *two stages - transfer and excursion* (**2**), with *drop tanks* (**.5**); and has reusable *dual crew modules* (**D**).

Technology & Programmatics - Reference Hardware Design Review

WBS Summary Items	Weight in Metric
Structures, Tanks, Landing Gear	4,869 Kg
Aerobrake (incl.Thermal Shield)	2,881
Separation & Docking Equip.	103
Thermal Control (Core/Tanks)	1,763
Propulsion (2 Types, 8 Engines)	4,031
Reaction Control (2 Types)	513
Power (Primary Electrical)	716
Guidance Nav. & Ctrl. Avionics	376
Comm. & Data Hdlg. (incl. VHS)	320
Wiring & Elec. Interface Equip.	555
Subtotal -	16,127 Kg
Weight Growth Margin (@ 15%)	2,417
Estimated Total Dry Weight	18,544 Kg

Vehicle Set Illustration



AFP- IR 1-7

3-22-90

Figure 2.2.5-1: Cost Model Weights Input for LTS Mission - 90 Day Study Configuration

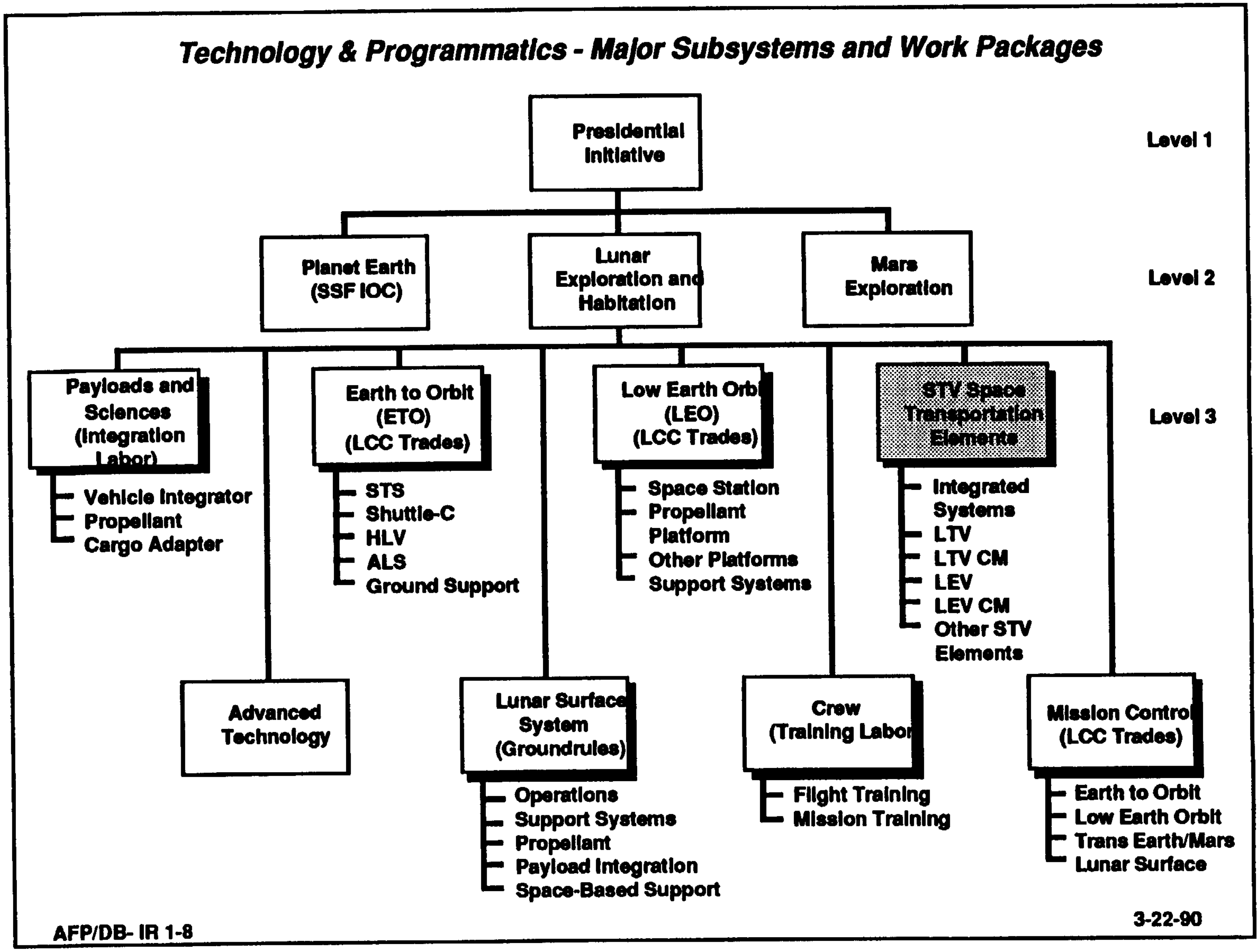


Figure 2.2.5-2: Work Breakdown Structure Tree at the Initiative Level is Defined First

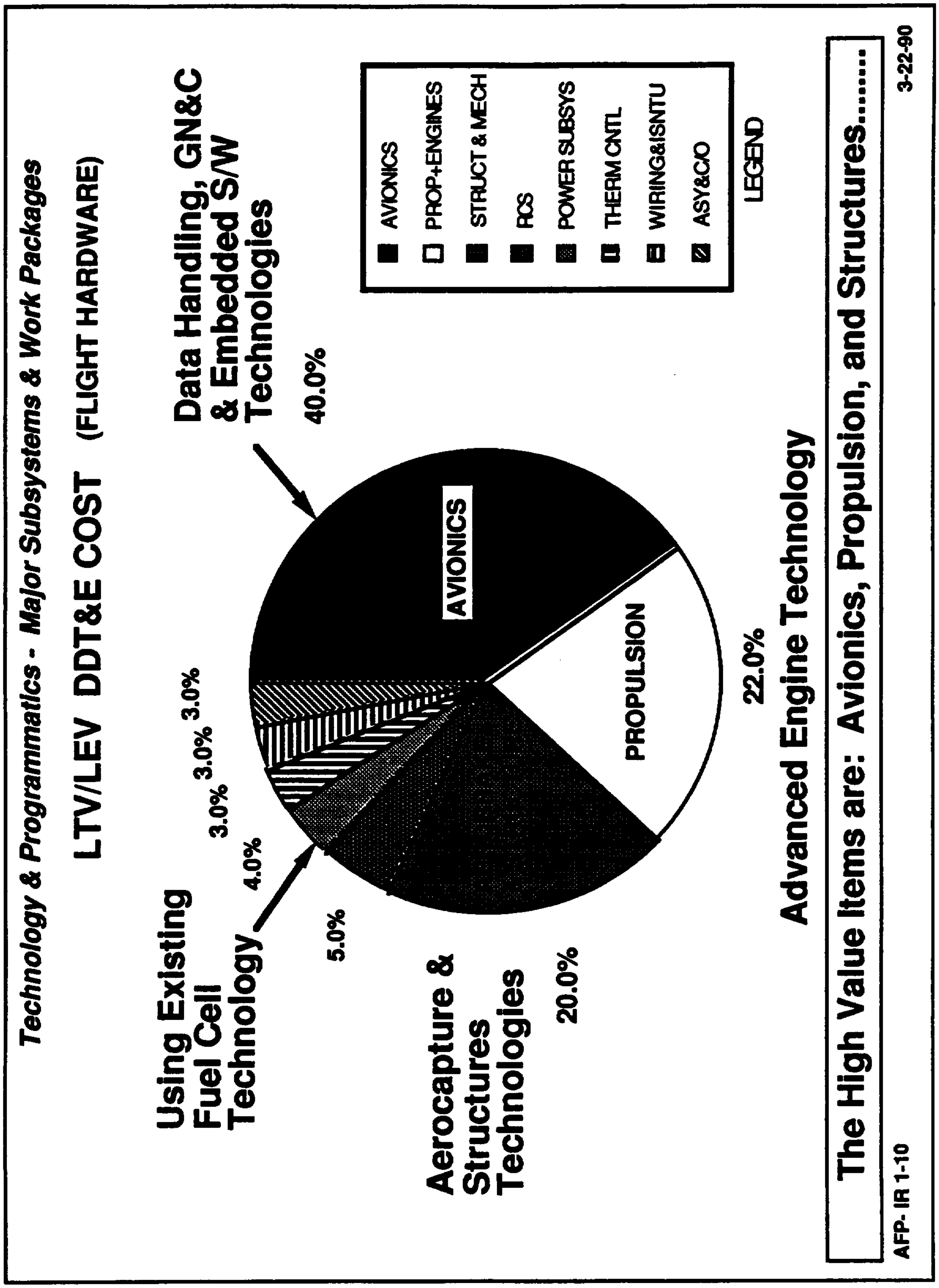
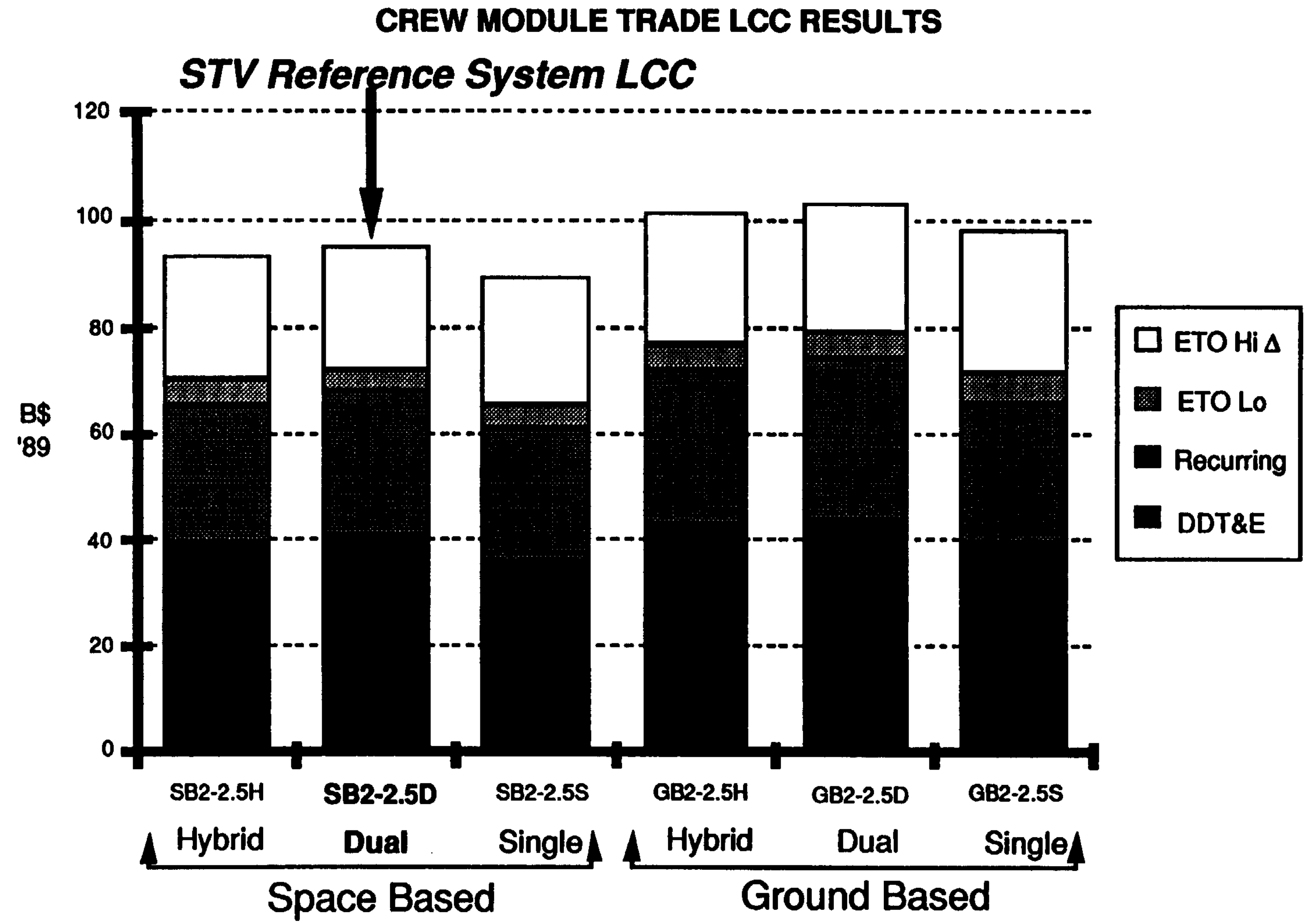


Figure 2.2.5-3: A Preliminary Development Estimate Breakout for the 90 Day Study Vehicle



Single module shows slight cost benefit over hybrid (3-4%) and dual (5-6%)

Figure 2.2.5-4: The 90 Day Study Vehicle is Compared to Other LTS Mission Designs

2.3 Space-Based Multi-launch Concept

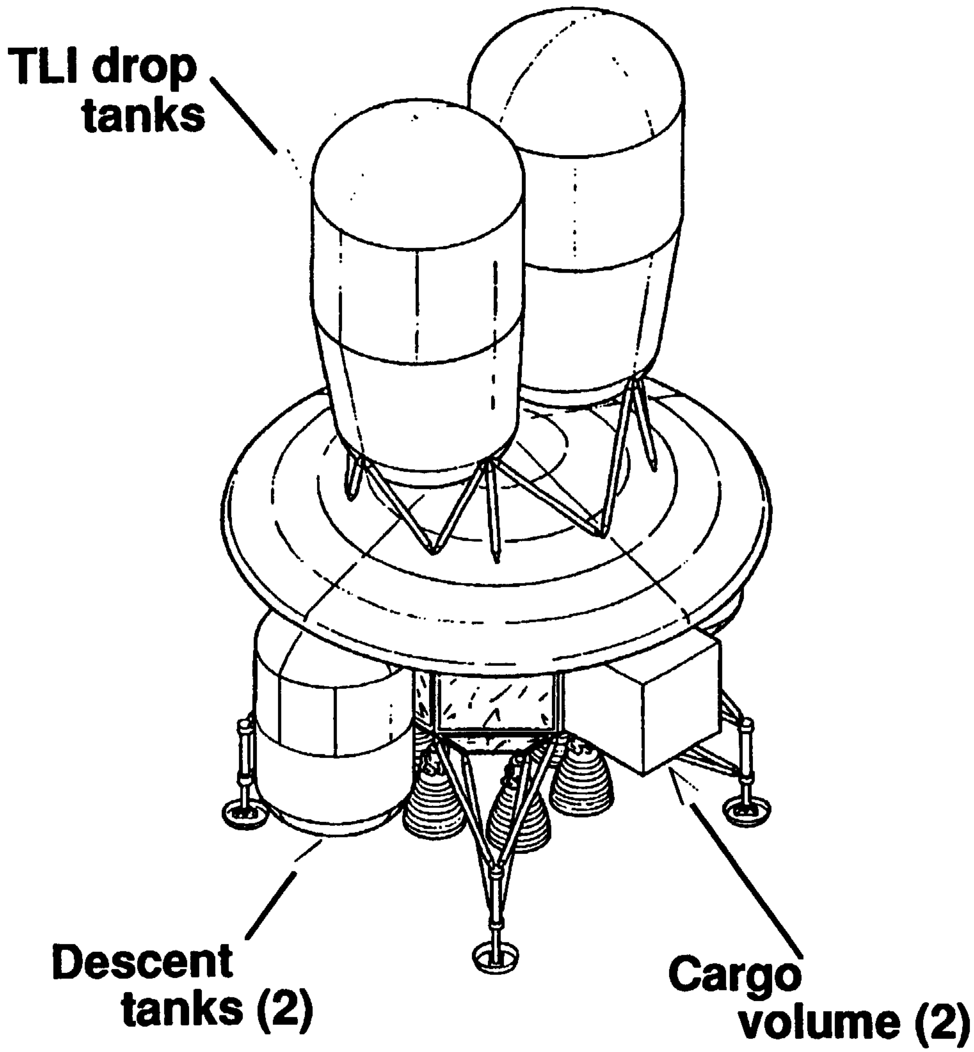
The selected space-based concept is based at the Space Station Freedom or other LEO node and is a cryogenic vehicle with a reusable core stage and two pairs of expendable drop-tanks, as shown in Figure 2.3-1. For piloted lunar missions, the core stage is flown with landing gear, a crew module, and a rigid, space-assembled aerobrake. For unpiloted lunar cargo-delivery missions, the core stage is flown in an expendable mode without the crew module and aerobrake. The drop-tanks for both missions include a pair of tanksets holding trans-lunar injection propellant and a pair of tanksets holding lunar-descent propellant. The vehicle has six main engines, allowing two engine-out capability during all mission phases.

2.3.1 System Design and Operation

For initial piloted missions, the core stage, crew module, and aerobrake are launched empty to the space station or LEO node aboard a heavy-lift launch vehicle, assembled, and then fueled from a propellant depot. The drop-tanks are launched fully loaded aboard three heavy-lift launch vehicles, integrated with the core stage, and then the crew and cargo are launched aboard a shuttle to the completed stage. The core stage returns to the LEO node after each mission, where it can be used for subsequent lunar missions or for other non-lunar missions.

The lunar mission sequential configuration of the vehicle is depicted in Figure 2.3.1-1. The aerobrake must be launched in sections to fit in the launch shroud and must be assembled on-orbit and then attached to the core vehicle with the crew module. The crew module is offset from the vehicle centerline to provide lunar landing visibility and cg offset for the aeromanuever, as shown in Figure 2.3.1-2.

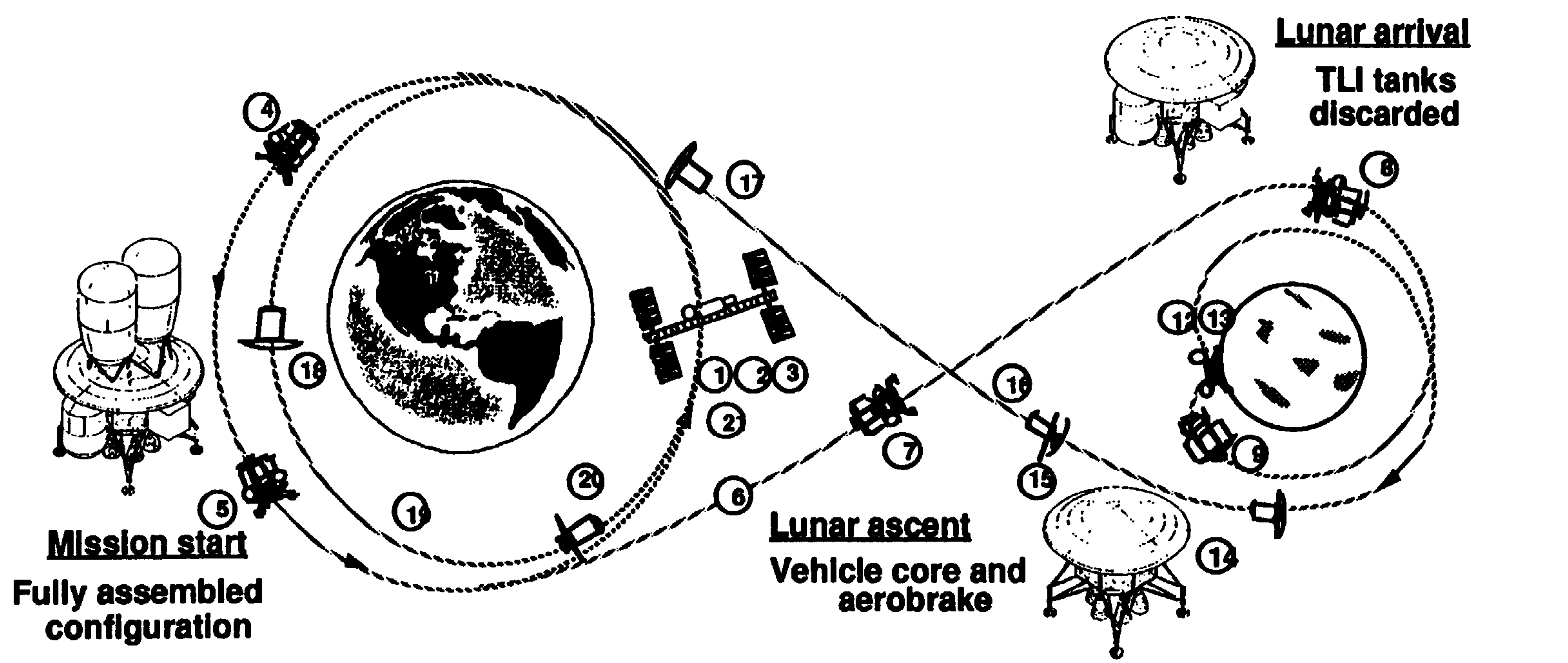
During the mission the TLI tanks are dropped after the TLI burn, and the vehicle descends to the lunar surface following lunar injection. During descent, the core ascent tanks remain full, balancing the cg to the centerline during the critical descent. For landing, the crew can view two landing pads and the horizon over the top of the cargo pallet. Upon arrival, the descent tanks are removed and the cargo is unloaded, the vehicle is hooked up to lunar surface support equipment, and the crew moves to the lunar habitat for the lunar stay. Due to the aerobrake overhang, cargo must be unloaded from the side of the core and moved to the base, either with built-in provisions or utilizing a lunar 'flatbed trailer', as shown in Figure 2.3.1-3. At the end of



Features

- 50 ft dia aerobrake
- Largest element 65 mt (TLI tankset)
- Reentry L/D >.2
- Asymmetric vehicle (offset crew module)
- Launchable in 30 ft shroud
- 15 ft x ∞ cargo envelope (expendable missions)
- Recovery to SSF
- Reuse all high value elements
- Crew module fits in Shuttle cargo bay
- Self unloadable

Figure 2.3-1 Space-based Vehicle Concept



Event		Δ Time, hours
1)	Vehicle assembly	4 weeks
2)	Payload integration	2 weeks
3)	Separate from SSF	0.0
4)	Coast	0.8
5)	TLI burn	0.3
6)	Trajectory correction maneuvers	84.0
7)	Transit/coast	0.1
8)	LOI burn	0.1
9)	Descend to Lunar surface	2.0

Event		Δ Time, hours
12)	Landing	0.0
13)	Cargo offload	1.0
14)	LTV TEI burn	0.1
15)	Trajectory correction maneuvers	0.1
16)	Transit/coast	84.0
17)	Pre aero correction maneuver	0.1
18)	Aeromaneuver	0.1
19)	Post aero correction maneuver	0.1
20)	Earth orbit circularization	0.1
21)	Rendezvous/docking with SSF	1.0

Figure 2.3.1-1 Lunar Mission Sequence

The returning vehicle is asymmetric and has an offset crew cab to provide the cg offset required for aerobraking.

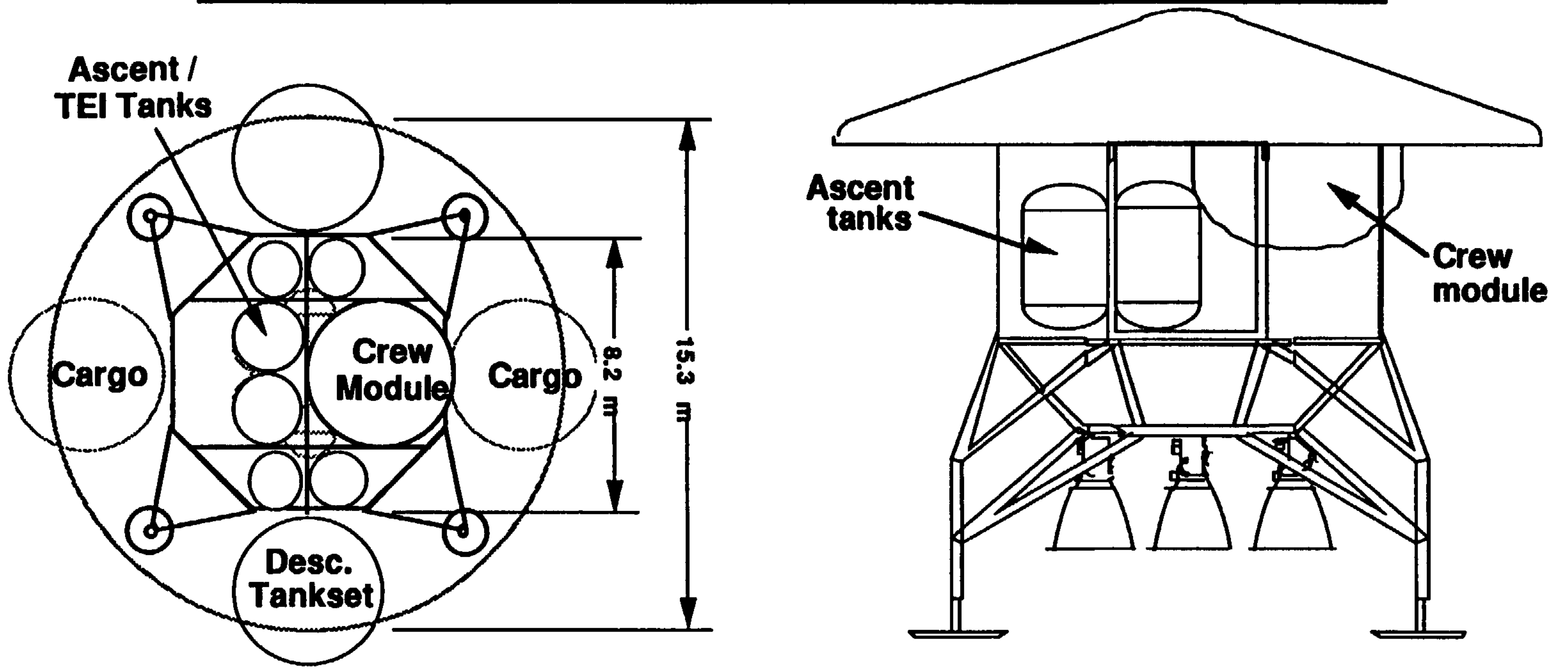


Figure 2.3.1-2 Crew Module Offset

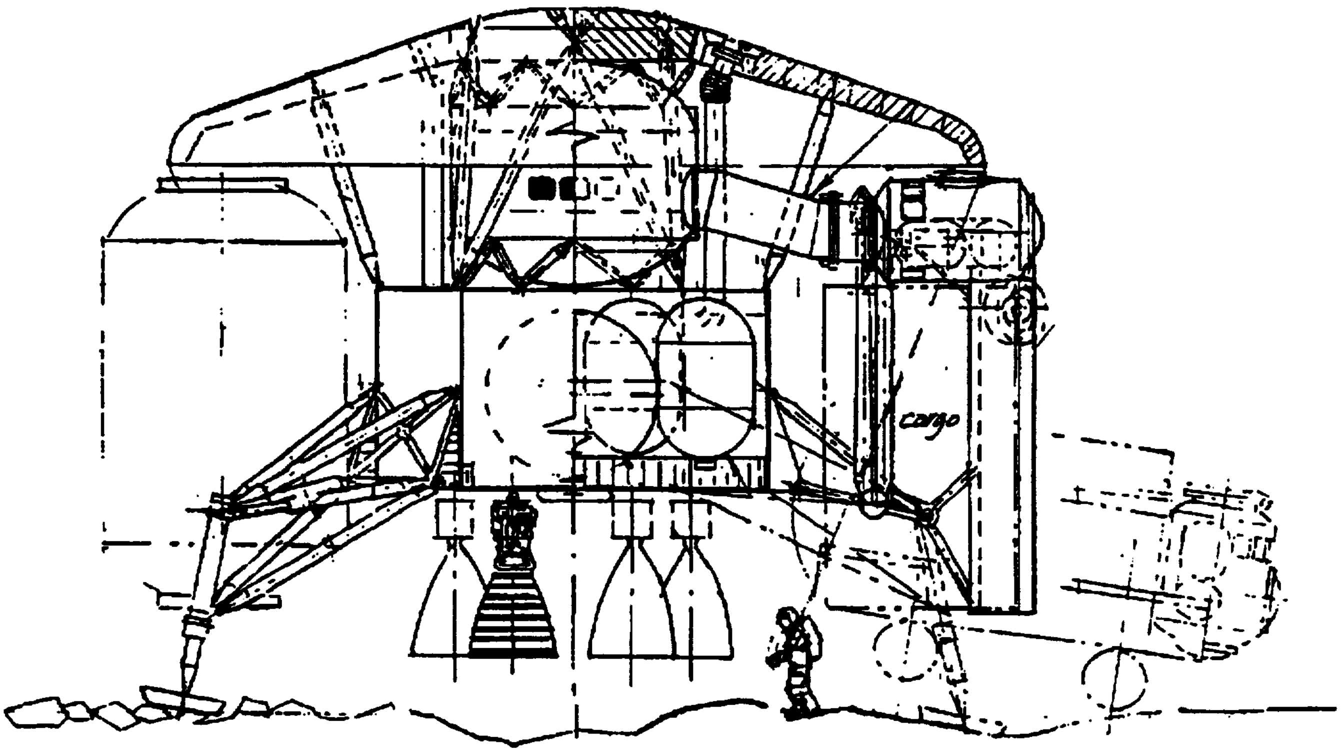


Figure 2.3.1-3 Payload Unloading Scheme

the lunar stay, the crew loads return cargo and boards, checks out the vehicle, then the core vehicle ascends and returns to the LEO node, utilizing an aeromaneuver, where it is inspected and refurbished for the next flight.

For the unpiloted mission, the core stage is flown without crew module and aerobrake, and is left on the lunar surface with the descent tanksets after landing.

Mass summaries for the space-based STV concept are given in Figures 2.3.1-4 and 2.3.1-5 for the piloted and unpiloted lunar missions, respectively. A weight growth margin of 15 percent was added to the estimated dry weight of each flight element to cover effects of design changes required to meet specifications at the time of delivery.

2.3.2 Flight Element Description

a. Core Stage – The space-based core stage has an external load-bearing body structure with structural interfaces to the crew module, aerobrake, cargo, and drop-tanks. The body structure includes twelve major longerons, a series of interior stabilizing struts, forward and aft stabilizing struts, and exterior closeout panels. The twelve 15.0 ft long, graphite / epoxy longerons transfer primary loads and form the backbone of the structure upon which the rest of the structure is supported. Eight of the longerons include interface fittings for the landing gear attachment, cargo attachment, and descent drop-tank attachment. The stabilizing struts are graphite / epoxy struts of varying lengths and sizes with titanium end fittings. The exterior and lower closeout panels are sandwich panels with honeycomb core and graphite / epoxy face sheets. These panels provide structural stiffness as well as shielding for the core tanks and crew module.

The octagonal thrust structure distributes thrust loads from the main engines to the vehicle and resists lateral engine gimbal loads. The thrust structure is of graphite / epoxy design, with six engine mounting pads and associated thrust vector actuator supports. It also includes struts for lateral load stabilization.

Lunar landing gear includes four sets of landing gear on the core stage that are deployed during lunar transit and stowed during Earth return, then reused for the next mission.

	Lunar Piloted Mission							
	Core Stage			TLI Drop-Tankset		LD Drop-Tankset		Lunar Surface Cargo
	Lander	Crew Module	Aero-Brake	Module #1	Module #2	Module #1	Module #2	
Structure and Mechanisms	1935	1496	1976	884	884	474	474	
Tankage - Main	583	-	-	1089	1089	597	597	
Protection	437	509	1583	642	642	385	385	
Propulsion - Main	2126	-	-	393	393	257	257	
Propulsion - Reaction Control	347	-	-	-	-	-	-	
Power Source	374	-	-	-	-	-	-	
Wiring & Electrical Interface	433	272	-	-	-	-	-	
Guidance, Navigation & Control	464	-	-	23	23	23	23	
Communication & Data Handling	422	124	15	35	35	39	39	
Displays & Controls	-	108	-	-	-	-	-	
Environmental Control	-	762	-	-	-	-	-	
Personnel Provisions	-	635	-	-	-	-	-	
Weight Growth Margin	1068	586	536	460	460	266	266	0
Total Dry Mass	**8189**	**4492**	**4110**	**3526**	**3526**	**2041**	**2041**	**9870**
Crew, with Suits	-	800	-	-	-			
Non-Propellant Consumables	-	291	-	-	-			
Non-Cargo Items - Residuals	332	-	-	781	781	374	374	
Inert Mass	**8521**	**5583**	**4110**	**4307**	**4307**	**2415**	**2415**	**9870**
MPS Usable Propellants	20967	-	-	63452	63452	27500	27500	
RCS Usable Propellants	137	-	-	102	102	152	152	
EPS Usable Reactants	242	-	-	5	5	391	391	
Other - losses, etc	107	1800	-	3168	3168	343	343	
Total LEO-Assembled Mass	29974	7383	4110	71034	71034	30801	30801	
	41467			142068		61603		9870
	255007							

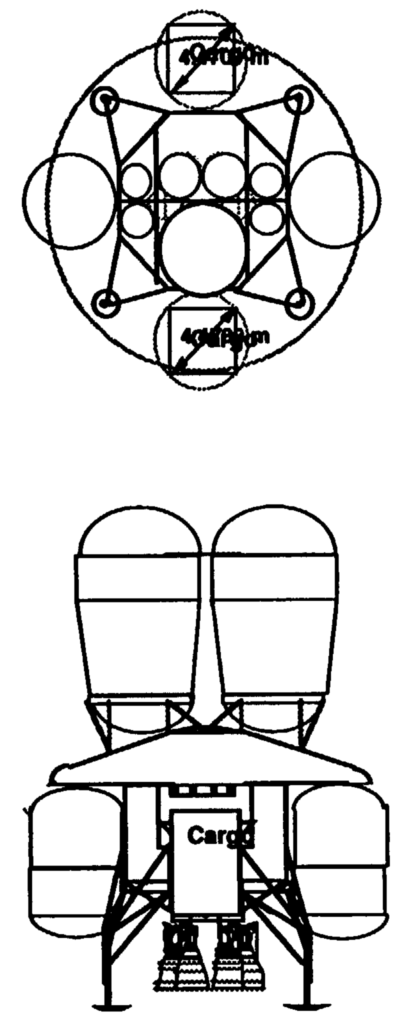
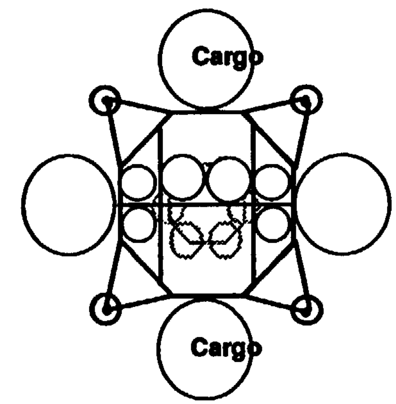


Figure 2.3.1-4 Space-Based STV Mass Summary - Lunar Piloted

	Core	TLI Drop-Tankset		LD Drop-Tankset		Lunar Surface Cargo
	Lander	Module #1	Module #2	Module #1	Module #2	
Structure and Mechanisms	1935	884	884	474	474	
Tankage - Main	583	1089	1089	597	597	
Protection	437	642	642	385	385	
Propulsion - Main	2126	393	393	257	257	
Propulsion - Reaction Control	347	-	-	-	-	
Power Source	374	-	-	-	-	
Wiring & Electrical Interface	433	-	-	-	-	
Guidance, Navigation & Control	464	23	23	23	23	
Communication & Data Handling	422	35	35	39	39	
Displays & Controls	-	-	-	-	-	
Environmental Control	-	-	-	-	-	
Personnel Provisions	-	-	-	-	-	
Weight Growth Margin	1068	460	460	266	266	0
Total Dry Mass	**8189**	**3526**	**3526**	**2041**	**2041**	**52683**
Crew, with Suits	-	-	-			
Non-Propellant Consumables	-	-	-			
Non-Cargo Items - Residuals	332	781	781	374	374	
Inert Mass	**8521**	**4307**	**4307**	**2415**	**2415**	**52683**
MPS Usable Propellants	21327	64887	64887	28212	28212	
RCS Usable Propellants	126	116	116	106	106	
EPS Usable Reactants	0	0	0	0	0	
Other - losses, etc	0	2624	2624	68	68	
Total LEO-Assembled Mass	**29974**	**71934**	**71934**	**30801**	**30801**	**52683**
		143868		**61603**		
	288127					



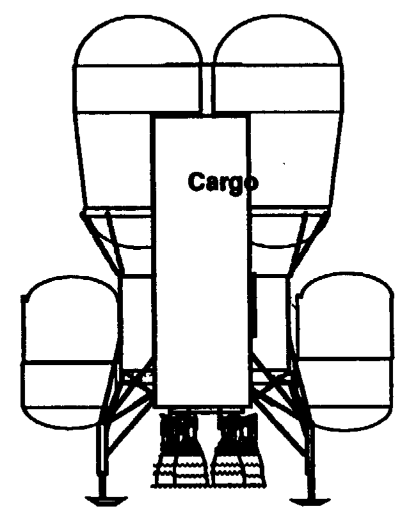


Figure 2.3.1-5 Space-Based STV Mass Summary - Lunar Cargo

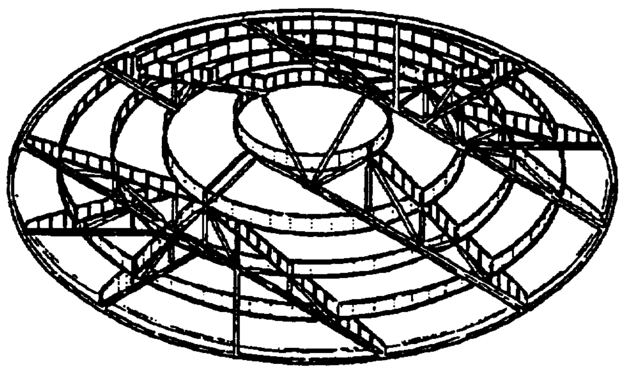
The core stage tanks are made of Aluminum-Lithium and include two cylindrical LO2 tanks and two cylindrical LH2 tanks with associated propellant aquisition devices. The tanks contain enough propellant for lunar ascent and Trans-earth injection.

b. Drop-tanksets – The current space-based tanksets include a single LO2 tank and a single LH2 tank with associated slosh baffles and propellant aquisition devices, composite honeycomb sandwich intertanks that distribute launch loads from an aft launch vehicle interface ring, and graphite / epoxy core vehicle interface trusses that permit on-orbit core vehicle integration and provide structural support during the mission. The TLI drop-tanks are integrated with the core vehicle above the aerobrake and require an aerobrake interface ring with both compression and tension interface fittings, and graphite / epoxy tankset support struts with titanium end fittings sized for TLI burn loads. The descent drop-tanks are integrated on the sides of the core vehicle and require graphite / epoxy support trusses with titanium fittings, sized for lunar descent and landing loads, as well as deployment and release fittings for dropping the empty tanks on the lunar surface.

c. Aerobrake – The space-based aerobrake is a rigid space-assembled shell structure of high-temperature graphite/polyimide sandwich panels affixed to a system of graphite/polyimide longerons and frames. The longerons in this structure are arranged in a series of concentric rings and feed the loads from the honeycomb panels into the truss members. The truss structure which carries the load into the vehicle core structure consists of two open-truss primary beams which are offset from the aerobrake centerline and span the width of the brake. These two trusses also provide structural attachment for the aerobrake side panels which are attached during the aerobrake's assembly. On these side panels, three secondary trusses spread radially from the core structure attachment points as shown in Figure 2.3.2-1.

The graphite polyimide allows the aerobrake structure to run hotter than would be allowable with an aluminum structure (650° F for GR/PI vs. 350° F for aluminum). Along with this higher temperature capability, the thermal expansion of the graphite polyimide can be tailored to match that of the overlying ceramic TPS. The impact of this CTE match is that if Shuttle tiles are used, the underlying strain isolation pad (SIP) can be left out and the tiles would then be bonded directly to the underlying panels.

d. Crew Module – The space-based crew module consists of a pressurized primary shell with internal bulkheads and partitions, windows for landing and docking maneuvers, and two hatches for EVA and crew transfer. The crew module has 27.8 m^3 pressurized volume.



Rigid Deployable
15.2 m (50 ft) diameter
9.2 m (30 ft) shroud diameter
(W/CdA =71.4 kg/m2
= 14.6 psf)
70 deg cone
3 m (10 ft) spherical radius nose
.3 m (1 ft) edge radius

TPS Definition
Zirconia fiberous ceramic
mechanically attachable to GR/PI
BlackGlas™ overlayment

Structural Definition
Graphite/polyimide honeycomb panels
Graphite/polyimide support beams
High temperature seals

Figure 2.3.2-1 Aerobrake Definition

2.3.3 Subsystem Description

A breakdown of the Space-based vehicle subsystems is shown in Figure 2.3.3-1. Descriptions of the major subsystems are as follows:

a. Main Propulsion – The selected space-based STV main propulsion system is a LO2/LH2 system and uses six advanced expander-cycle engines with a vacuum thrust of 15,000 lb per engine, and an assumed specific impulse of 481 seconds. It includes the engines with electromechanical actuation, as well as propellant delivery, pressurization, fill, and vent systems.

b. Reaction Control – The reaction control system is a gaseous O2 / gaseous H2 system with an assumed specific impulse of 410 seconds. It ncludes four GO2/GH2 thruster modules and associated accumulators, pressurization, and control.

c. Electrical Power – The electrical power system features redundant O2/H2 fuel cells fed from accumulators filled from the vehicle main propellant tanks, as well as distribution and control units and associated wire harnesses.

Fuel cell reactants are drawn from accumulators included in the Reaction Control Subsystem. The redundant accumulators are sized to provide oxygen and hydrogen reactants for both RCS and EPS functions for a period of time needed to fill the other accumulators. Once filled, the reactants are isolated and heated to supercritical pressure. Reactants are then drawn off to supply the fuel cells through a system of CRES manifolds.

For peak power loads during main engine actuation, three rechargeable Nickel-Hydrogen batteries are included in the power supply to supplement fuel cell power. The batteries are sized to provide a total of 5.0 kilowatt-hours of power to the main engine actuators.

The power distribution system consists of power distribution and control assemblies, inverters, and remote switching devices that interface with other vehicle subsystems and external power supplies.

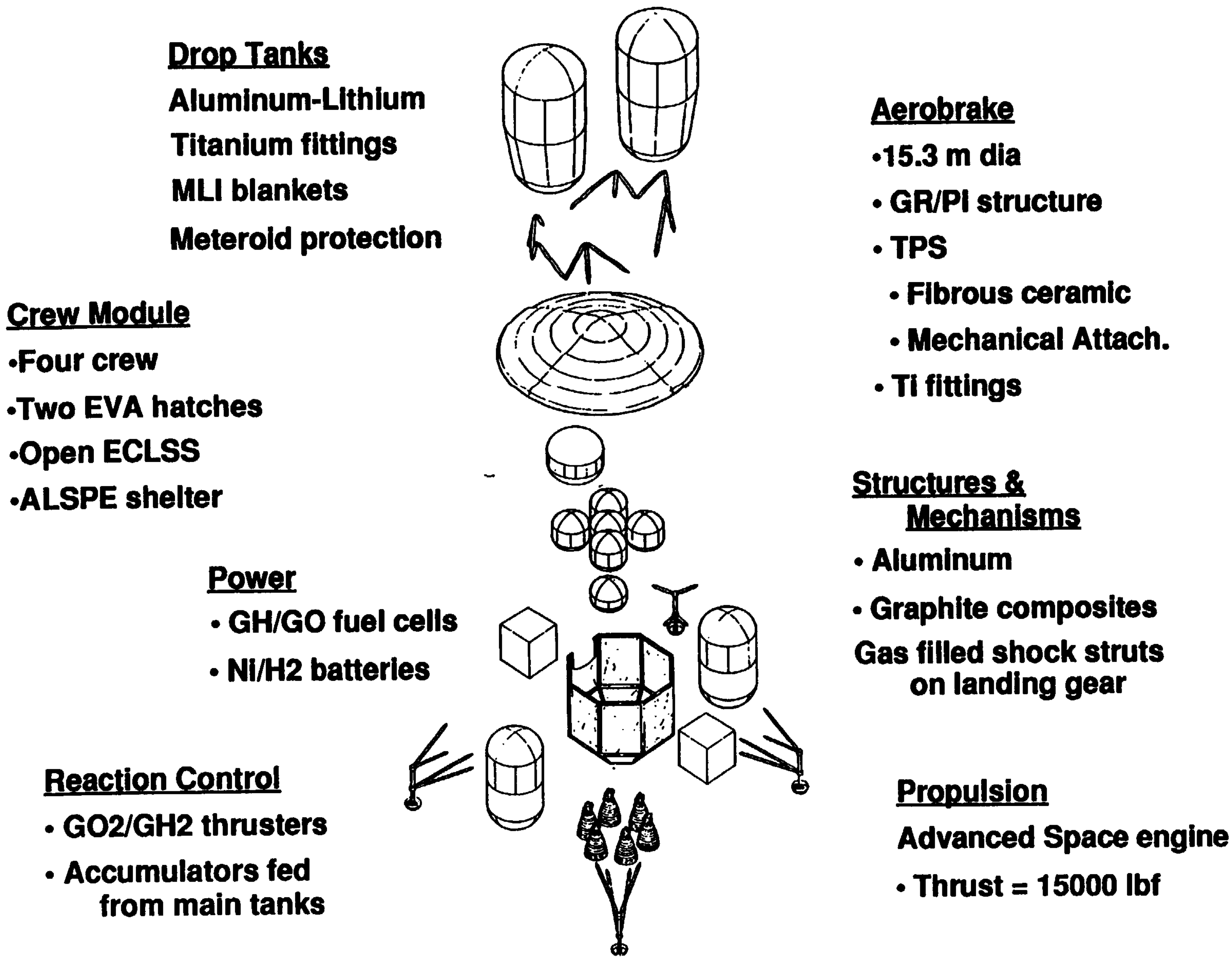


Figure 2.3.3-1 Space-based Subsystem Overview

d. Avionics – A schematic of the avionics subsystem equipment is given in Figure 2.3.3-2.

(1) Guidance and Navigation – Provisions for lunar mission operations, including rendezvous, docking, and lunar landing, with built-in redundancy for piloted operations.

(2) Communication and Data Handling – Provisions for communication, vehicle health maintenance, and data handling, with audio/video interfaces for piloted operations and instrumentation for drop-tank monitoring and control.

(3) Displays and Controls – Provisions on the crew modules for limited crew control and status monitoring of the vehicle during critical phases of the mission.

e. Environmental Control / Life Support – Environmental Control and Life Support (ECLSS) includes provisions on the crew module for atmosphere supply and control, internal equipment cooling, as well as metabolic and equipment heat rejection, similar to the 90-day reference concept. In this case, though, all life support functions are contained in a single crew module.

The system is an open loop life support system, with no regeneration of either atmosphere or water. Atmospheric gases are supplied from storage and from the fuel cell reactant supply accumulators, and carbon dioxide is removed from cabin air by replaceable LiOH canisters.

An active thermal control (ATC) loop is incorporated into the environmental control system, with coldplates for electronic equipment cooling, a cooling water loop for cabin thermal control, a Freon loop to cool vehicle heat loads, various equipment heat exchangers, and a variety of heat rejection devices designed for specific mission phases. Heat rejection devices include ground support equipment (GSE) heat exchangers, water flash evaporators, and space radiators.

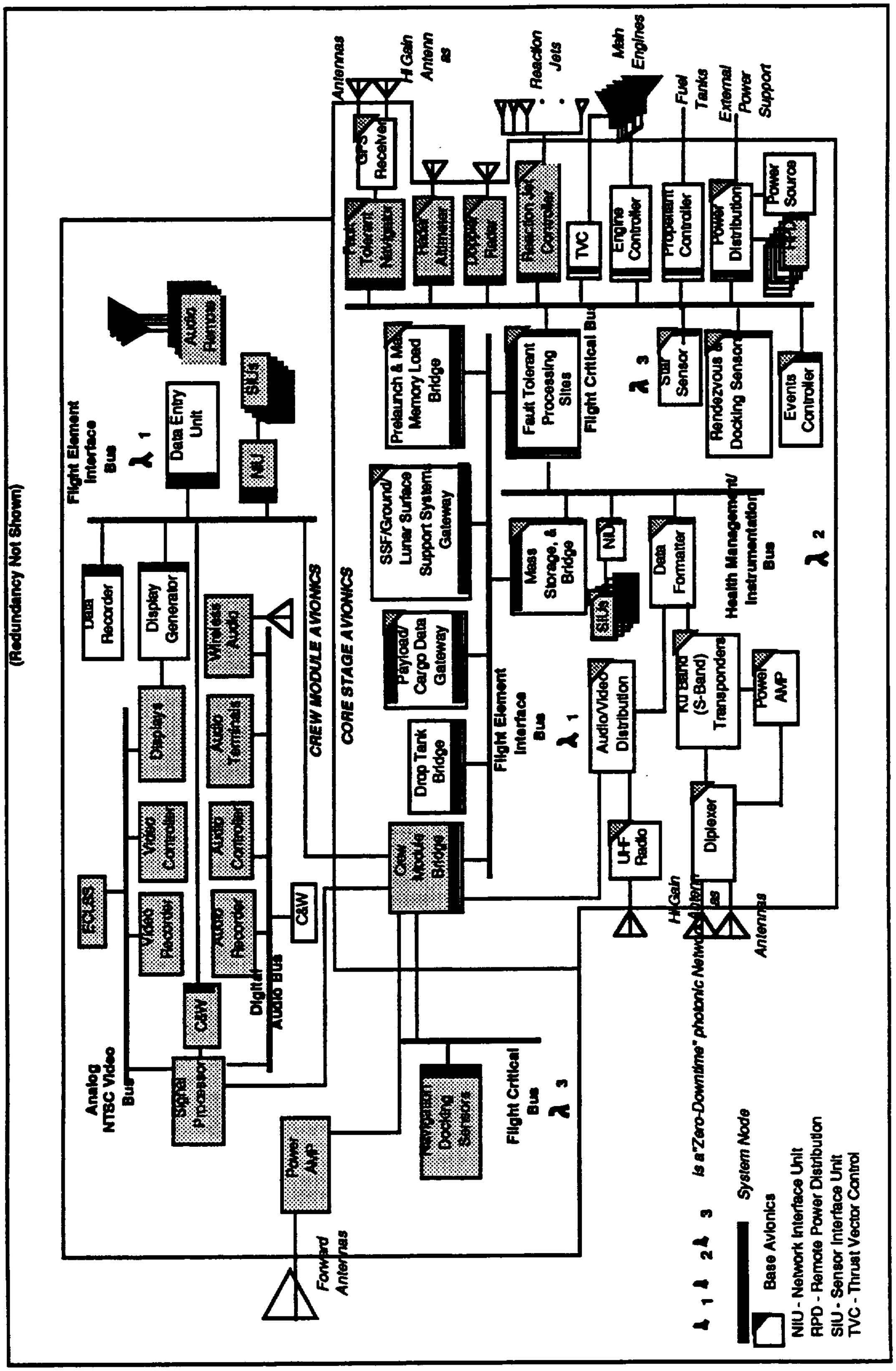


Figure 2.3.3-2 Avionics Block Diagram

f. Personnel Provisions

Personnel provisions include food, water, and waste management systems, as well as fire detection and crew furnishings, similar to the 90-day study reference. The food management system provides for the storage, preparation, and preservation of food for the crew. The food is shelf-stabilized and is prepared using warm water and heated in a convection oven, similar to the shuttle.

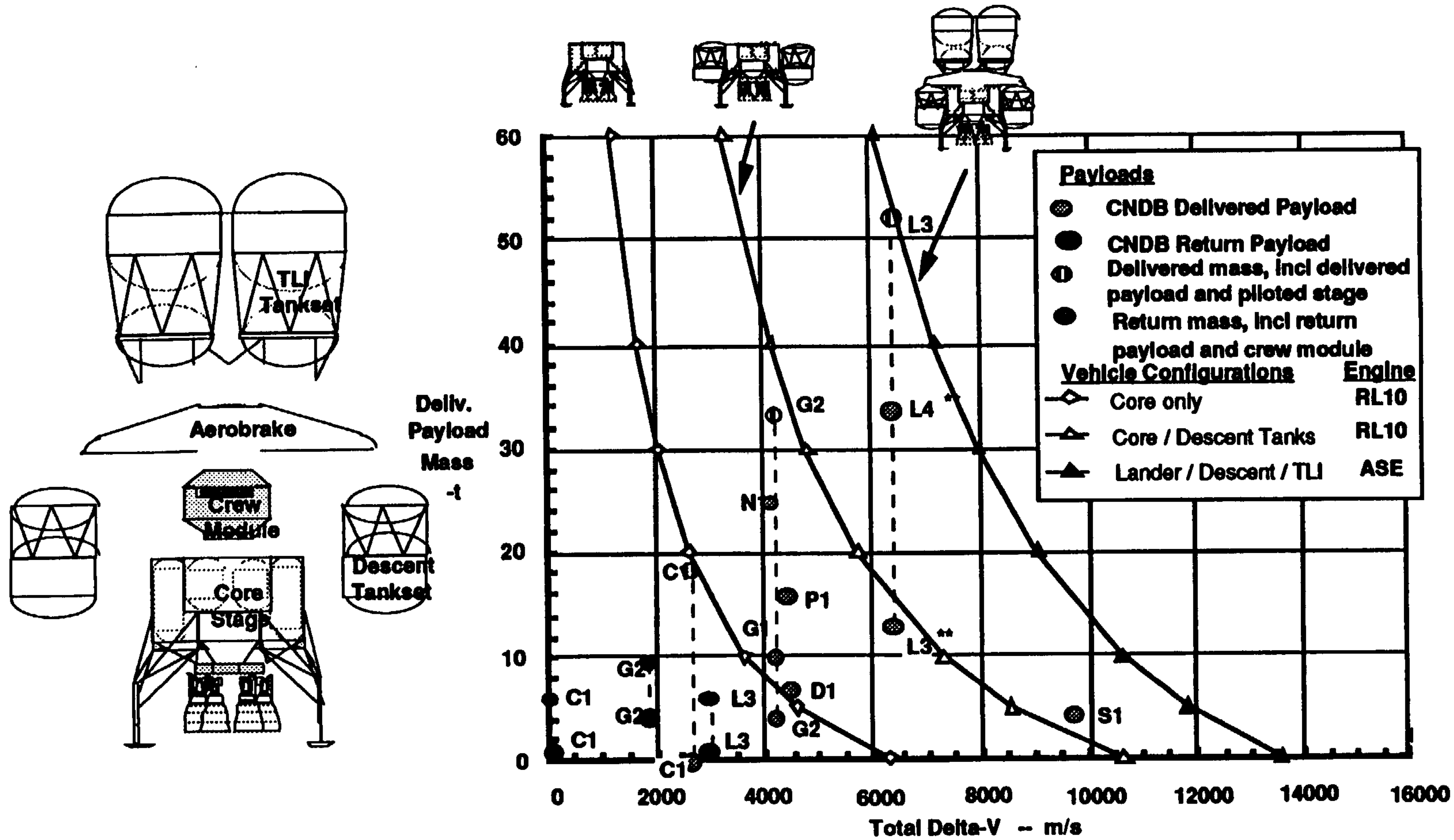
The water management system provides for potable water during the mission duration, and includes a water storage tank with water drawn from the fuel cell by-products, water dispenser, as well as tanks with a contingency water supply. The waste management system for both space- and ground-based vehicles includes a partitioned zero-g commode / hygiene station with waste storage tank and pre-moistened wipes for personal hygiene.

Crew furnishings include flight seats, emergency medical / health provisions, and personal equipment storage provisions. The flight seats, similar to those on the STS Orbiter, provide restraint and impact attenuation for all phases of flight and can be removed and stowed during flight .

2.3.4 Performance

The current space-based vehicle concept can either deliver 9870 kg cargo to the lunar surface in a piloted mode or 52,683 kg in a cargo delivery mode. With this cargo split, a total of 418 tonnes of cargo is delivered to the lunar surface over 21 piloted and 4 cargo-only missions, and the size of the vehicle is common to both piloted and cargo-only missions.

Different configurations of the space-based STV flight elements can be used to capture other non-SEI missions, as shown in Figure 2.3.4-1 as well as CNDB mission payloads and delta-V's. Un-piloted delivery missions are shown as single points on the chart and are captured by the core stage with RL10's and descent tanksets, except for the lunar cargo delivery mission (L4) and recoverable polar platform servicing mission (S1), which require additional tanksets. Piloted missions are shown with dashed lines, connecting delivered mass (including return stage, crew module, and delivered payload) to the delivered payload, and return mass (including crew module and return payload) to return payload quantities. The sample return mission (C1) is



** Optimized cargo split is 52.7 t / 9.9 t

Figure 2.3.4-1 Space-based Vehicle Capabilities captured completely by the core stage only. This stage is also adequate for both the lunar (L3) return and GEO servicing (G2) return. To deliver the core stage, crew module, and payload for the G2 mission, descent tanksets must be added for the delivery leg. To deliver the lunar core stage and lunar cargo, the full lunar vehicle is required.

2.3.5 Space-Based LTS Cost Estimates - 1.5 Stage

The space-based single stage estimate for the STV Lunar Transportation System (LTS) mission is generated from many sources of description and cost data. Besides the 90-Day Study vehicle (a two stage configuration) experience, the Boeing STV study team collected data from prior Orbital Transfer Vehicle (OTV) studies and companion space transportation studies accomplished for NASA by General Dynamics (GD) and Martin Marietta (MMC). Data for space operations and provisioning descriptions was extracted from several GD and MMC studies of space-based systems operating from a space station node.

a. Aerobrake Estimate – Prior Boeing engineering work on aerobrakes and hot structure spaceplane materials (like Dynasoar structures, Single Stage to Orbit structures, advanced fighter structures, and the National Aerospace Plane special sealants) was obtained and reviewed by the Boeing team. Aeroassist Flight Experiment advanced tile design improvements CAD/CAM process data was also obtained from NASA JSC project engineers. This background data helped the configurator and cost analysts to identify aerobrake subsystems technology applications, from a designers point of view, for the aerobrake cost estimate inputs and the development of an in-depth aerobrake development plan.

b. Drop Tank Estimates – The estimates for drop tanks were compared with Boeing Saturn 1C and public domain Shuttle external tank acquisition cost and hours actual data to check the reasonableness of the parametrically-derived cost estimates. Data was collected from Kennedy Space Center for the NSTS Shuttle external tank operations and it was used to develop the drop tank operations flows for STV operation and support cost estimating.

Design of the thermal protection and fluid control of the drop tanks was supplemented by cost analysis of the COLDSAT test system (accomplished earlier in the study as an additional special study task on the STV contract.) The COLDSAT cost estimate results were reviewed jointly by the NASA, Boeing, MMC, and GD participants. The COLDSAT contractors cost estimates information provided the teams with valuable calibration data for the space-based vehicle cost estimate.

c. Crew Module Estimates – The crew module estimate was derived from an expanded hardware definition list generated in cooperation with Boeing Seattle and Huntsville designers. The crew module and stage avionics functions were balanced between the two flight elements to ensure access and modularity for lower refurbishment hours and cost. In some cases, NSTS Shuttle Orbiter actual operations cost data for avionics and life support subsystems was used to develop the operations flow estimates for cost estimating inputs. Special operations factors were added to the Orbiter experience for advanced avionics reductions in maintenance (credit) and increased time for in-situ space refurbishment (debit).

d. In-space Operations – Cost factors and relationships were derived from data generated for Space Station Freedom in 1988 and 1989. Figure 2.3.5-1 contains the in-space cost estimating factors used to generate the operation and support costs for this preferred space-based LTS configuration. The in-space operations estimates included special costs for the maintenance supplies and booster Earth-to-orbit (ETO) costs associated with the continuous operation of the reusable LTS vehicle.

e. Development Test Quantity – From a design development stand point, special groundrules and matrices were developed for the test hardware quantity requirements. Ground and flight test hardware quantities are cost drivers in the development estimate. Figure 2.3.5-2 contains an example of a matrix approach to identifying STV transfer stage test hardware requirements and usage in the development phase. A matrix like the one shown was developed for every configuration flight element. The final "equivalent" quantity of test units (some subsystems require more units, some less), was a total of five units for the development phase.

f. Design for Cost – Information was generated to establish a cost conscious design attitude for the space-based system configuration. Figure 2.3.5-3 contains a pie chart breakout of the core stage cost estimate which was used to identify development high value items. The development cost risk analysis results shown in figure 2.3.5-4 were used in concert with the estimate percentage allocation pie charts to optimize the space-based design and development cost estimates.

(Source of base dollar parameters is the MSFC/Boeing SSF contract)

SERVICE OR LABOR TASK, ON-ORBIT OR IN-SPACE	UNIT OF MEASURE	UNIT ESTIMATE (1991 DOLLARS)
Extravehicular Activity (EVA)	Crew (2) + IVA Obsvr. Hour	$ 135,500 /hr.
Intravehicular Activity (IVA)	Astronaut (1) Labor Hour	21,000 /hr.
SSF/Free Flyer Services:		
SSF Service Facility (+Equip.)	30x30m Hanger Cost *	$ 550 M
SSF Operations Module/Equip.	6.8x4.5m Maint. Shop*	200 M
Unpressurized Logistics Pallet	2.5x4.6m Carrier Assy.(Rec)	30 M ea.
SSF Logistics Pallet Service	Per Pound of Cargo	4,000 /lb.
SSF Logistics Module Use	Per Pound of Equip.	6,000 /lb.
SSF Airlock Services	Per Egress & Ingress Event	150,000 /ea.
Manipulator Arm Service	Per Operations Hour	46,000 /hr.
Electric Power from SSF	Per Kilowatt Hour	250 /hr.
Propulsion (On-orbit moves)	Sq. Ft. Cross Sec. Area/Day	5 /unit
Data Management Services	Per Channel Ops. Hour	7,300 /hr.
Software Support Services	Per Line of Code (HOL)	350 /line
Communication Services	Per Channel Ops. Hour	2,800 /hr.
Space Tug Refurbishment	Per STS Delivery (14 Flts./Yr.)	58 M

(Note: * These estimates include development & one production unit costs.)

Figure 2.3.5-1: In-Space Operations Factors for STV Space-Based Systems LCC Estimating

Preflight Test Hardware

Article: / Function:	Structures tests/ loads verification	Thermal tests/ loads verification	Aerobraking tests	Power systems tests	Avionics/C&D test/ verification/development	ECLSS tests	Impact attenuation tests stabilizat'n (ballistic rtn)	Interface verification	Propulsion system tests	Ground handling/recovery procedures development	Ergonometrics/ human factors	Procedures development	Crew training	Public relations	Stack integration tests	Space Operations tests	Notes:
I. Engineering Mockups			○		○		●	●		○	●	○	○	●	●	○	
IIa. Flight Simulator - Static																	-No Hdwr-
IIb. Flight Simulator - Dynamic																	-No Hdwr-
III. "Iron Bird"/SIL		○		●	●			●				○			●		
IV. Neutral Bouyancy Mock up							○	○			○	●	●			●	
Va. Structural Test Article(s)	●						●	○									
Vb. Thermal Test Article(s)		●															
Vc. Aero-Aerothermal Test Article(s)																	-No Hdwr-
Vd. Main Propul Test Article(s)								○	●								
VI. Pathfinder								●		●		●			●	●	
VII. Certification/Prototype					○					○	○	○			○	○	
VIII. Subsystem development hdwe.		●		●	●			○	●								

Primary use: ● Secondary Use: ○

Figure 2.3.5.-2: Test Hardware Matrices were Developed for all Flight Elements

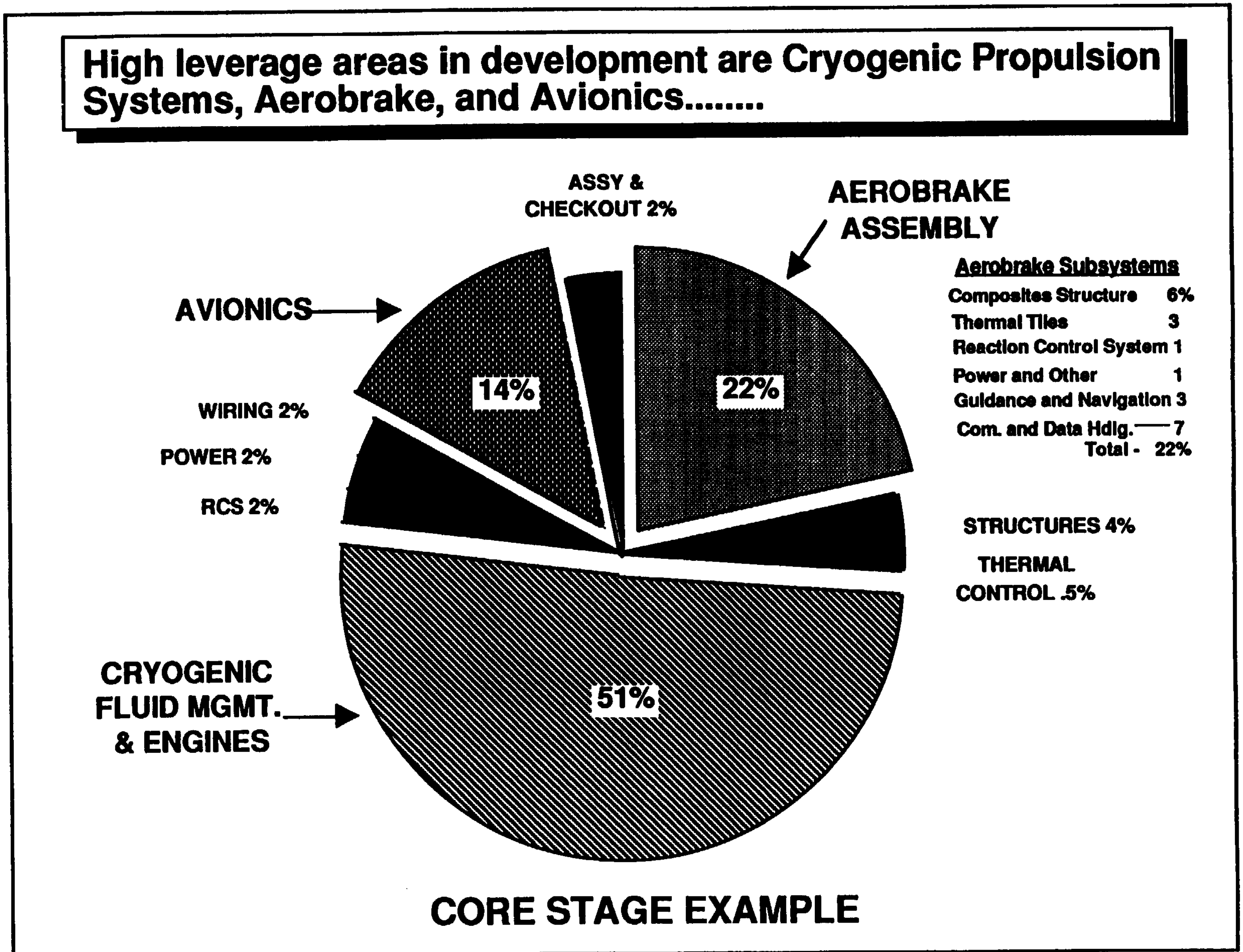


Figure 2.3.5-3: New Space-Based Vehicle High Value Items are Identified

Ranger Cost Risk Analysis
by LTS Flight Element (Before Factors Application)
(1991 Dollars in Millions)

	DDT&E Hardware Estimate*	Low	50/50	High
SPACED BASED	($ 24,594 Total)			
AEROBRAKE	1987.65	1655.08	2051.49	2459.81
CREW MODULE	4303.70	3558.59	4514.20	5366.02
TLI TANKS	390.08	323.28	402.95	484.23
LUNAR DESCENT TANKS	680.19	564.89	702.86	843.93

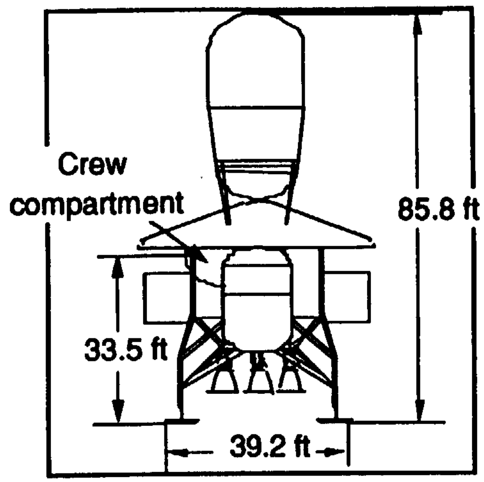
Figure 2.3.5-4: A Boeing Ranger Cost Analysis is Conducted

g. Cost Estimate Summaries – Boeing was asked to look at other uses for the STV LTS core stage in a derivative configuration with less engines. Figure 2.3.5-5 is a summary of the space-based LTS estimates presented in October, 1990. The "STV Other Production" line in the summary is the production cost estimate for those other mission hardware units.

After the final review, Boeing was directed by the customer to change the mission model for STV to only include the lunar missions plus the required space tug missions and add ETO cost estimates to the LCC total. Figure 2.3.5-6 contains the final space-based LCC summary. The net result of eliminating the "other" missions described in the NASA Civil Needs Database and adding ETO costs (dollars per pound estimtes supplied to both contractors by NASA MSFC) was about 15.5 billion dollars due to mission model change credits and ETO cost debits.

Evolutionary system requirements, which generate hardware development benefits and added program complexities, need to be considered at the subsystem *component* level to properly estimate the STV configuration costs for a mixed mission architecture. Special attention to design input descriptions and sequence of development for the related hardware subsystems can reduce (or significantly increase) the overall cost estimate. In the case of this estimate, the added other STV mission expendable hardware was a *significant LCC driver* for the space-based LTS configuration (over 50% of LCC, according to the mission model groundrules provided by the NASA customer and Boeing mission analysts.)

(Constant-Year 1991 Dollars in Millions)



- Single Stage Vehicle

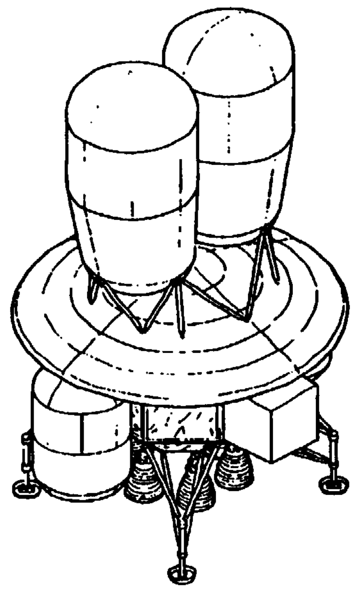
Operations:
- SSF Lunar Node for LTS and Selected Missions
- ETR Launch Site; Some Other Mision Ground Launches
- 178 Expended stages for other missions
- 8 Lunar Sortles

Development:	
Core Stage & Aerobrake	$ 6,459 M
Drop Tanks	1,536
Crew Module	2,457
Software (Flt. & Ground)	1,500
Subtotal -	11,952 M
Requirements Factor (30%)	3,586
Contractors Fees (8%)	1,243
NASA Pog. Support (15%)	2,517
Subtotal -	$ 19,298 M
GFE Adv. Engine Program	1,525
Facilities Investments	5,900
Total DDT&E and Facilities -	$ 26,723 M
Operations for 10 Years:	
LTS Production (1/Yr.)	6,793
STV Other Production (19/Yr.)	66,076
LTS O&S (8 missions)	2,543
Other CNDB O&S (178 flt.'s)	9,369
SSF, KSC, MSFC, S/W Maint.	2,423
Total Production and O&S -	$ 87,204 M
Total Life Cycle Cost Estimate -	$ 113,927 M

10-19-90

Figure 2.3.5-5: A Space-Based, 1 1/2 Stage Vehicle Preliminary Life Cycle Cost Summary

(Constant-Year 1991 Dollars in Millions)



Space-Based Vehicle

Operations:
- SSF Lunar Node for LTS and Selected Missions
- ETR Launch Site; 71 Mt HLLV Carrier (Shuttle C size)
- Includes Tug Missions
- Other Missions Excluded

Development:	
Core Stage & Aerobrake	$ 6,459 M
Drop Tanks (Revised)	1,087
Crew Module	2,457
Software (Flight & Dev. S/W)	1,500
Subtotal -	11,503 M
Requirements Factor (30%)	3,450
Contractors Fees (10%)	1,495
NASA Prog. Support (Rev. 5-15%)	1,174
Subtotal -	$ 17,622 M
GFE Adv. Engine Program	1,072
Facilities Investment	5,900
Total DDT&E and Facilities -	$ 24,594 M
LTS O&S for 27 Years (Incl. DT&E Flights):	
Full Production (1/Yr. + Tugs)	14,525
LTS and Tug Oper. & Support	4,404
ETO Launch (71 M t Booster @ $2,500/lb.)	52,434
SSF, Fac., & S/W Maint.; ILS	2,423
Total Production and O&S -	$ 73,786 M
LTS Life Cycle Cost Estimate -	$ 98,380 M

Note: Other CNDB missions were not addressed in this update.

AFP- IR 5

1-17-91

Figure 2.3.5-6: The New Reference Vehicle Design LCC is Updated to New Groundrules

2.4 Ground-based Concepts

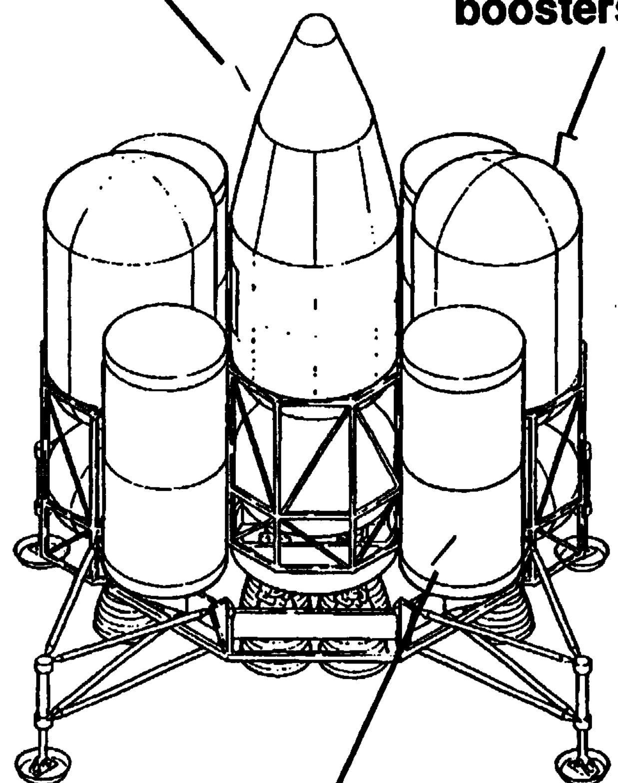
Two configurations of a ground-based STV include a single-launch concept and a multiple-launch concept. The two selected ground-based concepts are cryogenic vehicles with a reusable crew module and avionics pallet, an expendable core stage made up of a propulsion module and tankset, a pair of expendable TLI drop-tanksets, a pair of expendable delivery stages, and an expendable lunar lander platform. Figure 2.4-1 shows a single-launch concept in which all flight elements are launched full in a single HLLV launch, and Figure 2.4-2 shows a concept in which most of the LO2 is launched in a separate launch and transferred to the main vehicle in LEO. In both cases, on-orbit assembly is minimized. The vehicles each have six main engines, allowing two engine-out capability during all mission phases.

2.4.1 System Design and Operation

The ground-based vehicle can be operated in either of two launch modes. The entire vehicle with crew and cargo can be launched to orbit fully loaded aboard a very heavy-lift launch vehicle (single-launch ground-based), or it can be launched in two or more smaller launches (multiple-launch, on-orbit rendezvous). For the latter case, the first launch would include a tanker to fill the vehicle LO2 tanks and the second launch would include the vehicle with off-loaded LO2 tanks. In both cases, the only reusable element is the crew module with equipment pallet, which reenters the Earth's atmosphere and returns to the ground, where it is refurbished and reused.

The on-orbit operations of the multiple-launch vehicle are depicted in Figure 2.4.1-1. The LO2 tanker launched initially remains on-orbit until the core vehicle launch. The core vehicle is launched with a crew module escape structure that includes a docking mechanism and tank fill provisions. It docks with the tanker, fills its LO2 tanks, then jettisons the tanker, escape structure, and LO2 fill plumbing. From that point, both ground-based concepts are similar in mission configuration.

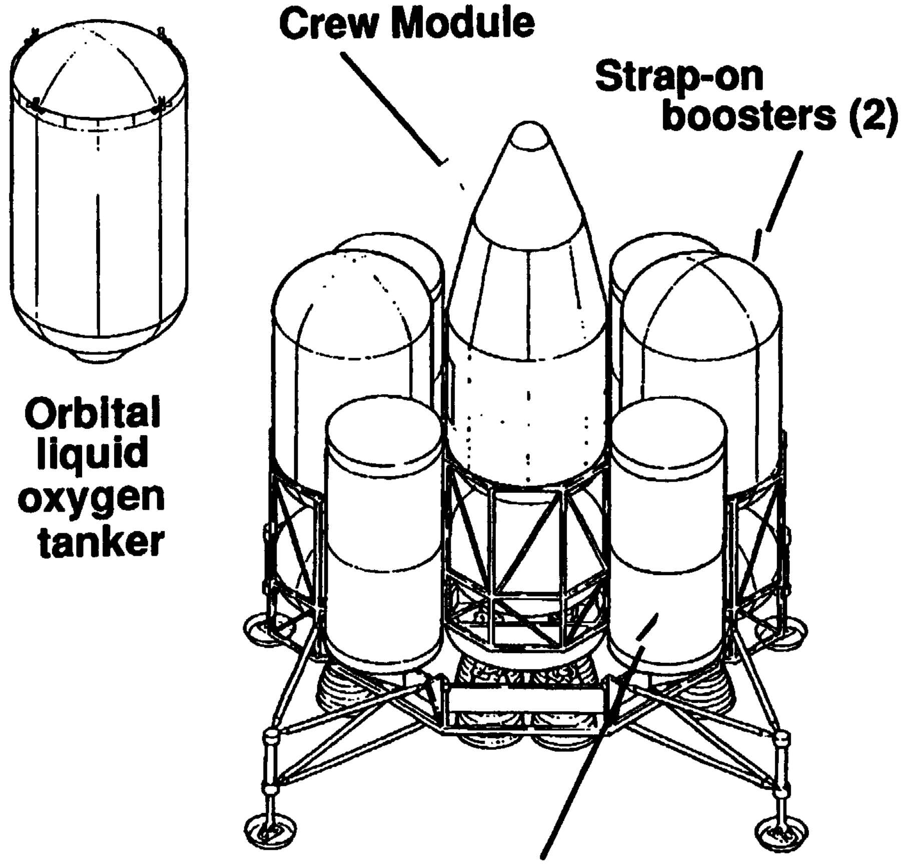
The common configuration sequence of the ground-based STV is shown in Figure 2.4.1-2. The TLI tanks are dropped after the TLI burn, and the vehicle descends to the lunar surface following lunar injection with lander, core stage, delivery stages, and cargo. During landing, the crew can view two landing pads and the horizon over the top of the cargo pallets. Upon arrival, the cargo is unloaded and the delivery stages, with one engine each, are either removed or tilted aside. The vehicle is hooked up to



Features

- Single launch with 260t booster, no rendezvous
- 13.7 m dia launch shroud
- Six 15klb thrust engines
- Cargo envelope: 4.5 m dia x 12.2 m (piloted), 4.5 m dia x ∞ (expendable)
- Self unloadable
- Crew module ground recoverable
- Strap-on boosters usable as separate upper stage

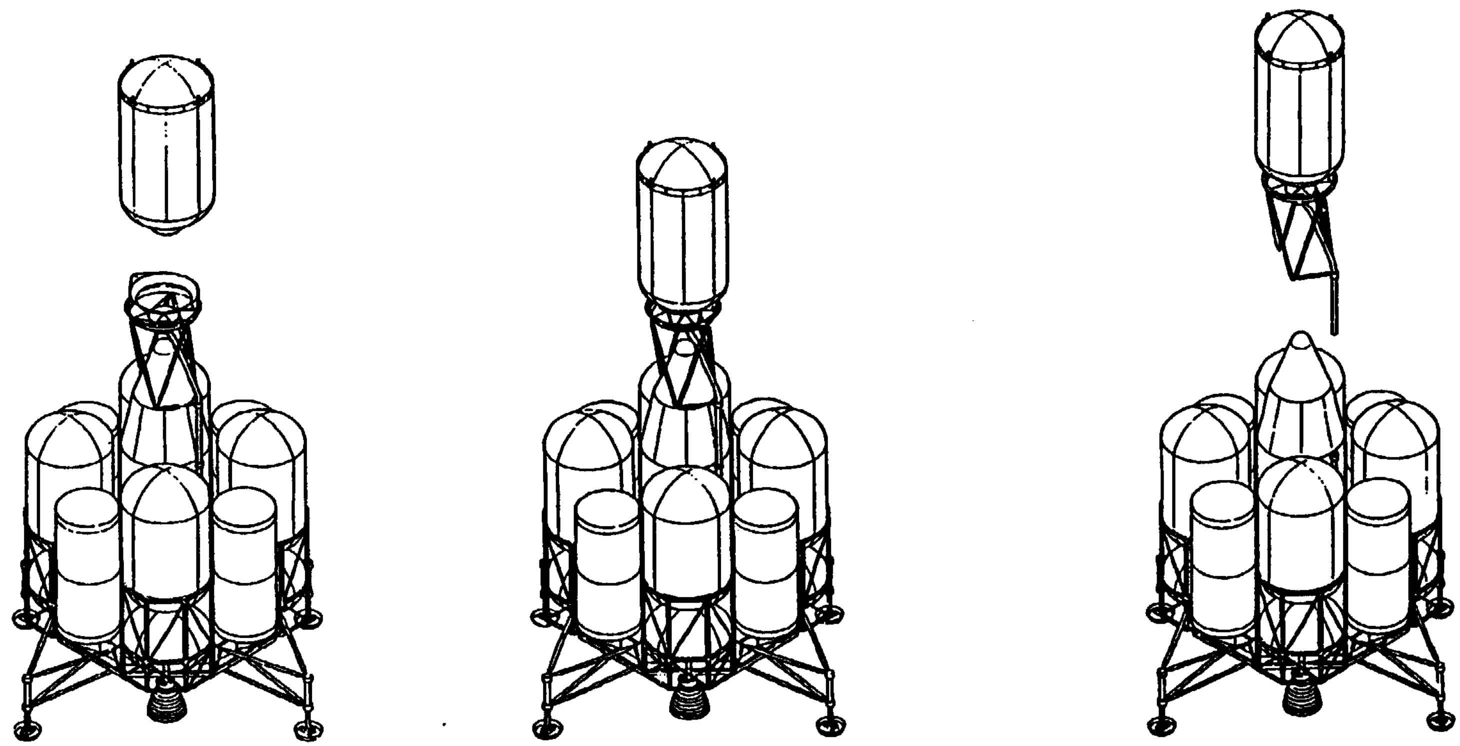
Figure 2.4-1 Ground-based Single Launch STV Concept



Features

- No on-orbit vehicle integration
- Tanker launched 1st with oxygen offloaded from stage
- 13.7 m dia launch shroud
- Six 15klb thrust engines
- Cargo envelope: 4.5 m dia x 12.2 m (piloted), 4.5 m dia x ∞ (expendable)
- Self unloadable
- Crew module ground recoverable
- Strap-on boosters usable as separate upper stage

Figure 2.4-2 Ground-based Multi-Launch STV Concept



- Dock with tanker using LES support structure

- Perform settling burn
- Transfer liquid oxygen

- Disconnect LES / tanker berthing structure & transfer umbilical
- Back STV away from tanker
- Tanker performs deorbit burn

Figure 2.4.1-1 Orbital Refueling Operation

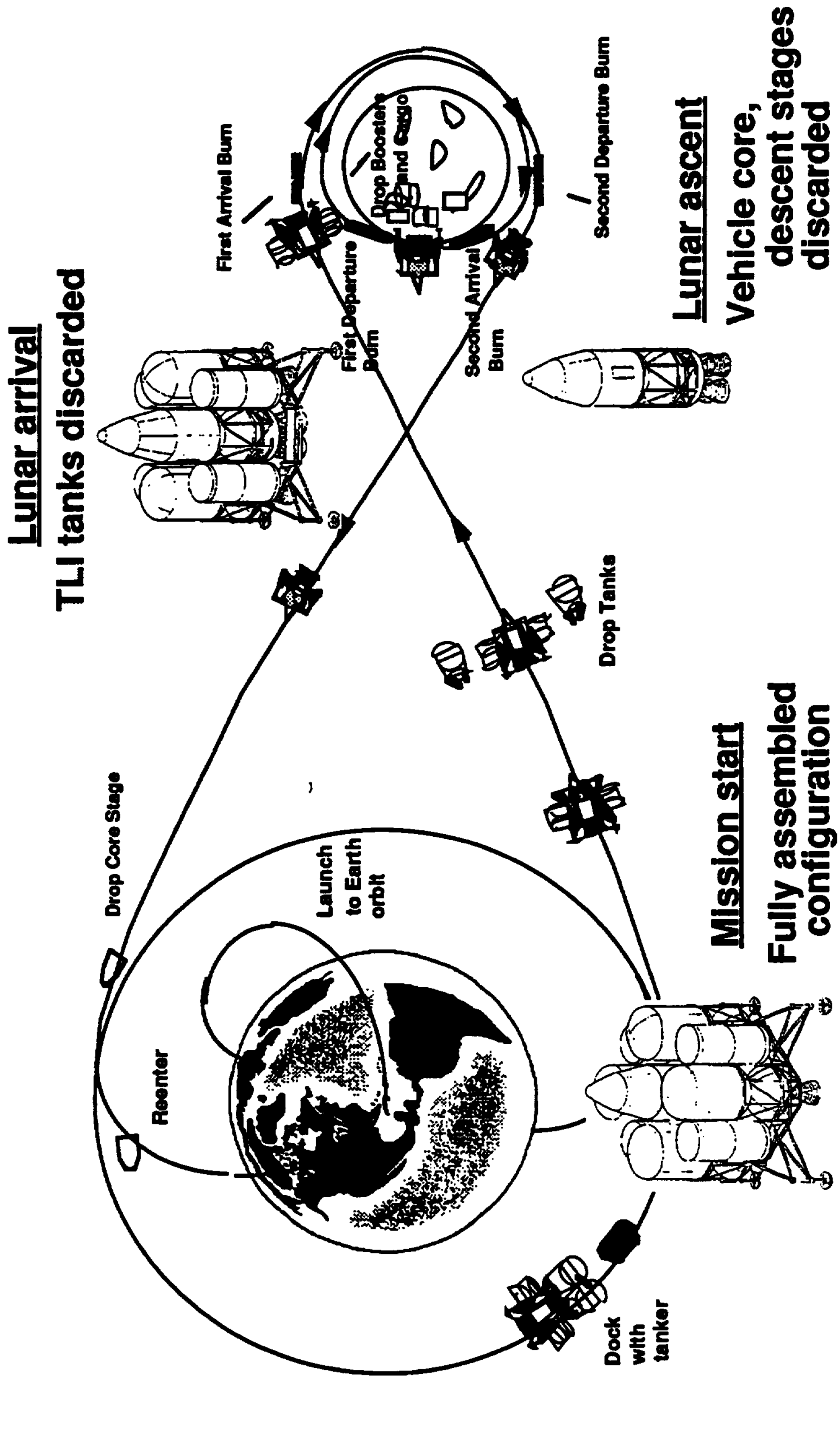


Figure 2.4.1-2 GB Vehicle Mission Operation

lunar surface support equipment, and the crew moves to the lunar habitat for the lunar stay. Cargo can be unloaded from the side of the core, as shown in Figure 2.4.1-3, and moved to the base, either with built-in provisions or utilizing a lunar 'flatbed trailer'. At the end of the lunar stay, the crew loads return cargo and boards using a hoist, checks out the vehicle, then the core vehicle ascends, with the expendable lander acting as a launch platform. The core stage is expended prior to reentry, and the crew module with avionics pallet reenters and lands near the launch site, as shown in Figure 2.4.1-4, where it is inspected and refurbished for the next flight.

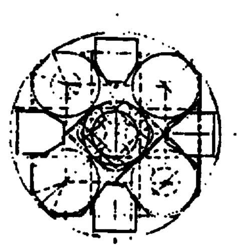
For unpiloted lunar cargo-delivery missions, neither the crew module nor the ascent tankset are required, and the core propulsion module with avionics pallet is left on the lunar surface with the lander and delivery stages.

Mass summaries for the ground-based STV concept are given in Figures 2.4.1-5, 2.4.1-6, and 2.4.1-7 for the piloted lunar, unpiloted lunar, and unpiloted GEO delivery missions, respectively. A weight growth margin of 15 percent was added to the estimated dry weight of each flight element to cover effects of design changes required to meet specifications at the time of delivery.

2.4.2 Flight Element Description

a. Core Stage – The ground-based core stage is made up of a tankset and propulsion module. The tankset has an external load-bearing truss body structure with a forward interface to the avionics pallet and crew module, an aft interface to the propulsion module, and forward interfaces to a pair of TLI drop-tanks and a pair of delivery stages. The truss includes forward and aft aluminum interface ring frames, two aluminum ring frames that provide support for the core LO2 and LH2 tanks, and intermediate graphite / epoxy longerons and stabilizing struts. The propulsion module consists of a thrust structure and a lander interface structure with explosive bolt fittings that attach to four support arms on the lander.

The core stage thrust structure consists of an aluminum double-cruciform beam structure and circular thrust ring with an average cross-sectional area of 4.0 in2 excluding beef-up and pads, and is attached to an interface ring that transfers thrust loads into the lander and core tankset. The thrust structure also includes engine interface / TVC actuator pads and lateral load stabilization struts. Each single-engine



- **Self unloading**
- **Cargo split into four pieces**
- **Ramps integral to cargo pallet**
- **Relatively low c.g. for landing stability**

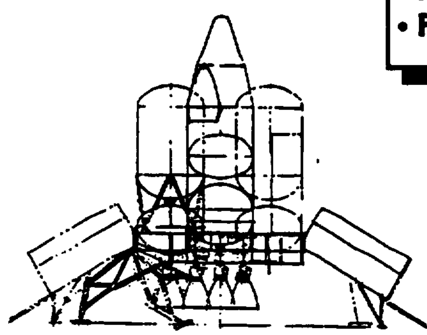
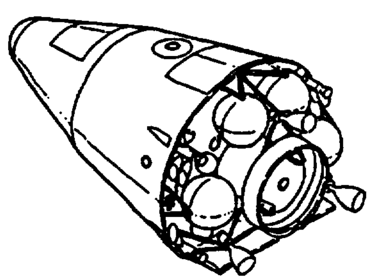
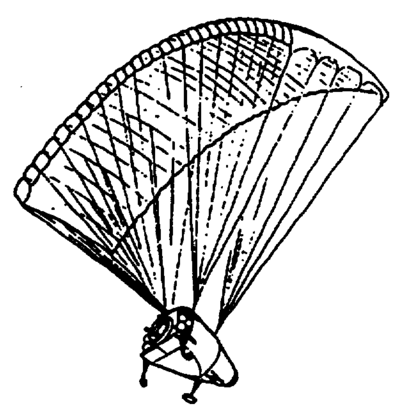


Figure 2.4.1-3 GB Vehicle Cargo Unloading Scheme



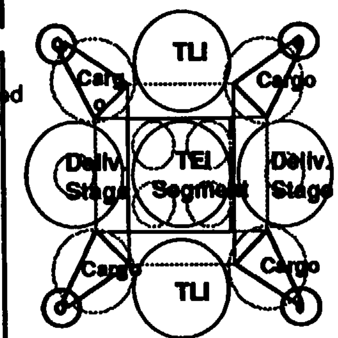


Scarfed biconic shape
L/D > .8
Parafoil recovery
Refurbishable

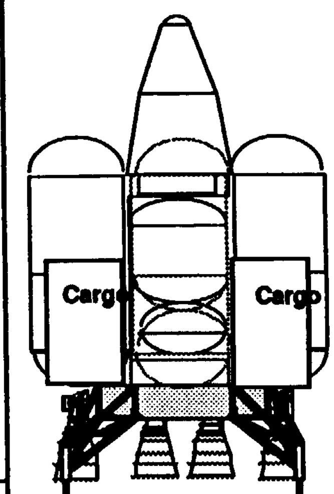
Independent RCS
Capable of surviving water ditch

Figure 2.4.1-4 GB Crew Module Return to Ground

STV Mass Summary Ground-based Vehicle	Lunar Piloted Mission									All mass in kg
	Core Vehicle							Drop-tanks		
	TEI Segment			Delivery Segment						
	Crew Mod	Av. Pallet	Tankset	Prop Mod	Lander	D Stg #1	D Stg #2	TLI Tnkst #1	TLI Tnkst #2	Delivered Cargo
Structure and Mechanisms	3341	155	505	483	1206	502	502	433	433	
Structures & Mechs - Landing gear	281	-	-	-	741	-	-	-	-	
Tankage - Main	-	-	385	-	-	659	659	659	659	
Protection	1315	116	378	-	82	570	570	570	570	
Propulsion - Main	-	-	376	917	828	545	545	380	380	
Propulsion - Reaction Control	162	-	-	-	-	190	190	-	-	
Power Source	-	374	-	-	-	-	-	-	-	
Wiring & Electrical Interface	272	265	28	56	78	39	39	28	28	
Guidance, Navigation & Control	130	464	-	-	-	-	-	-	-	
Communication & Data Handling	189	391	37	21	19	37	37	37	37	
Displays & Controls	108	-	-	-	-	-	-	-	-	
Environmental Control	813	-	-	-	-	-	-	-	-	
Personnel Provisions	635	-	-	-	-	-	-	-	-	
Weight Growth Margin	1087	265	256	222	443	381	381	316	316	0
Total Dry Mass	**8333**	**2030**	**1965**	**1699**	**3397**	**2923**	**2923**	**2423**	**2423**	**11630**
Crew, with Suits	800	-	-	-	-	-	-			
Non-Propellant Consumables	308	-	-	-	-	-	-			
Non-Cargo Items - Residuals	15	-	275	92	435	542	542	527	527	
Inert Mass	**9456**	**2030**	**2240**	**1791**	**3832**	**3465**	**3465**	**2950**	**2950**	**11630**
MPS Usable Propellants	-	-	17294	-	-	44592	44592	44676	44676	
RCS Usable Propellants	44	-	97	-	-	153	153	100	100	
EPS Usable Reactants	2	-	151	-	-	396	396	149	149	
Other - losses, etc	1800	-	75	-	-	296	296	515	515	
Total LEO-Assembled Mass	11302	2030	19857	1791	3832	48902	48902	48390	48390	11630
	34980				101637			96780		
	245026									



Top View

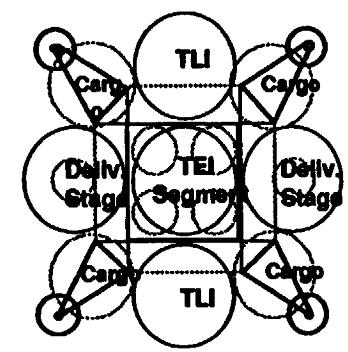


Side View

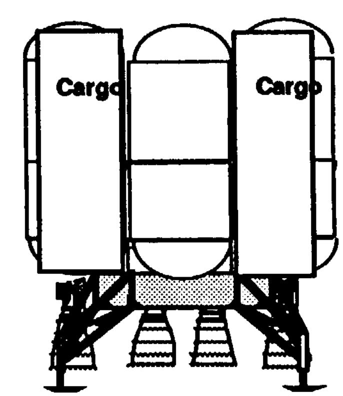
Figure 2.4.1-5 Ground-Based STV Mass Summary - Lunar Piloted

STV Mass Summary
Ground-based Vehicle **Lunar Cargo Mission - Unmanned** **All mass in kg**

	Core Vehicle					Drop-tanks		
	TEI Segment		Delivery Segment			TLI Tnks #1	TLI Tnks #2	Delivered Cargo
	Av. Pallet	Prop Mod	Lander	D Stg #1	D Stg #2			
Structure and Mechanisms	155	483	1206	502	502	433	433	
Structures & Mechs - Landing gear	-	-	741	-	-	-	-	
Tankage - Main	-	-	-	659	659	659	659	
Protection	116	-	82	570	570	570	570	
Propulsion - Main	-	917	828	545	545	380	380	
Propulsion - Reaction Control	-	-	-	190	190	-	-	
Power Source	374	-	-	-	-	-	-	
Wiring & Electrical Interface	265	56	78	39	39	28	28	
Guidance, Navigation & Control	464	-	-	-	-	-	-	
Communication & Data Handling	391	21	19	37	37	37	37	
Displays & Controls	-	-	-	-	-	-	-	
Environmental Control	-	-	-	-	-	-	-	
Personnel Provisions	-	-	-	-	-	-	-	
Weight Growth Margin	265	222	443	381	381	316	316	0
Total Dry Mass	**2030**	**1699**	**3397**	**2923**	**2923**	**2423**	**2423**	**43443**
Crew, with Suits	-	-	-	-	-			
Non-Propellant Consumables	-	-	-	-	-			
Non-Cargo Items - Residuals	-	92	435	542	542	527	527	
Inert Mass	**2030**	**1791**	**3832**	**3465**	**3465**	**2950**	**2950**	**43443**
MPS Usable Propellants	-	-	-	44592	44592	44676	44676	
RCS Usable Propellants	-	-	-	153	153	100	100	
EPS Usable Reactants	-	-	-	396	396	149	149	
Other - losses, etc	-	-	-	296	296	515	515	
Total LEO-Assembled Mass	2030	1791	3832	48902	48902	48390	48390	
	3820		101637			96780		43443
	245680							



Top View

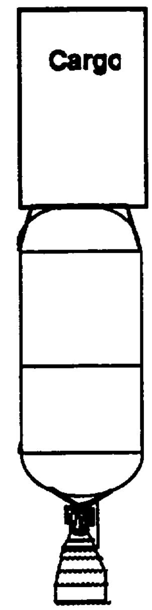


Side View

Figure 2.4.1-6 Ground-based STV Mass Summary - Lunar Cargo

STV Mass Summary	All mass in kg
Ground-based Vehicle	Unmanned Delivery

	Core Vehicle Delivery Segment Av. Pallet	Core Vehicle Delivery Segment D Stg #1	Delivered Cargo
Structure and Mechanisms	134	502	
Structures & Mechs - Landing gear	-	-	
Tankage - Main	-	659	
Protection	116	570	
Propulsion - Main	-	545	
Propulsion - Reaction Control	-	190	
Power Source	381	-	
Wiring & Electrical Interface	211	39	
Guidance, Navigation & Control	192	-	
Communication & Data Handling	216	37	
Displays & Controls	-	-	
Environmental Control	-	-	
Personnel Provisions	-	-	
Weight Growth Margin	188	381	0
Total Dry Mass	**1438**	**2923**	**24000**
Crew, with Suits	-	-	
Non-Propellant Consumables	-	-	
Non-Cargo Items - Residuals	-	542	
Inert Mass	**1438**	**3465**	**24000**
MPS Usable Propellants	-	44592	
RCS Usable Propellants	-	153	
EPS Usable Reactants	-	396	
Other - losses, etc	-	296	
Total LEO-Assembled Mass	1438	48902	**24000**
	50340		
	74340		



Side View

Figure 2.4.1-7 Ground-based STV Mass Summary - GEO Delivery

delivery stage has a thrust structure that consists of a cruciform thrust beam for load distribution into the tankset structure, as well as an engine interface / TVC actuator support pad.

On the ground-based vehicle, the lunar landing gear is part of an expendable lunar landing platform that supports the core stage, delivery stages, and cargo modules during landing, and acts as a support platform for the core stage ascent from the lunar surface.

The core stage tanks are made of Aluminum-Lithium and include a single cylindrical LO2 tank and one cylindrical LH2 tank with associated propellant aquisition devices. The tanks contain enough propellant for lunar ascent and Trans-earth injection. The LO2 tanker is a single tank with internal stiffening and slosh baffling capable of withstanding launch conditions fully loaded. A description of the tanker is given in Figure 2.4.2-1.

b. Drop-tanksets – The ground-based tanksets are a different design, with a graphite / epoxy longeron and ring concept common to both delivery stages and TLI tanksets that easily integrates into the lander platform octagonal structure and that supports the tanksets partially loaded during launch. Tanks are supported within this truss by passive orbital disconnect struts (PODS). The struts are under development at NASA JPL and consist of concentric composite tubes; the outer tube designed for ground and launch loads, the inner one for smaller on-orbit loads. Once in orbit the outer tube pulls away from the inner one, reducing the on-orbit heat leak through the struts.

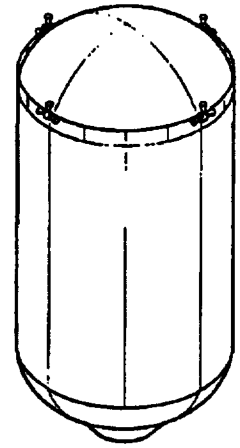
c. Crew Module – The ground-based crew module structure includes an internal pressurized shell with internal bulkheads and partitions and an external aerodynamic shell designed for reentry aerodynamic loads and landing. The crew module has windows for landing and docking maneuvers, and two hatches for EVA and crew transfer. The ground recovery subsystem applies to the ground-based crew module only. This subsystem includes all provisions for recovery of the crew module on the ground at mission conclusion. It includes parachutes, ground landing gear for the nominal dry landing, and emergency splashdown provisions for a launch abort or terminal descent steering failure.

Avionics

Redundant gyros
Horizon sensor
VHMS sensors
Processor

Reaction Control Options

Hydrazine thrusters
Cold gas thrusters
Resistojets
Torque rods
Momentum wheels



Structure

Al-Li domes and barrel sections

Concerns

Toxicity, power for heating
Mass flow, propellant
Propellant, power
Solar tank heating
Saturation, Solar tank heating

Figure 2.4.2-1 LOX Tanker Description

The parachute system includes a primary and backup drogue chute for initial deceleration, and a primary and backup hi-glide parafoil chute for final deceleration to touchdown. Also included are the parafoil control mechanisms for final descent steering and installation provisions for the chutes. The drogue chutes are 53 ft diameter mortar-deployed conical ribbon chutes for deceleration to a terminal velocity of 160 fps. The main chutes are two-stage controllable parafoils; the initial reefed condition slows the module to about 22 fps vertical velocity to minimize drift, then the parachute is opened fully to slow the vertical velocity to about 10 fps for final touchdown.

The ground-landing impact attenuation design includes two primary stroking struts with skids for primary attenuation, and a small castoring wheel (to prevent tipover) attached to a trailing arm strut located in the pointed end of the vehicle (aft end on landing). Large skid pads for low surface loading are part of the exterior vehicle skin and form the cover/door to the landing gear-well housing a gas cartridge deployed gas-filled strut. With the exception of the gas cartridge used for deployment, all components are reusable.

With a dry landing as a primary crew module recovery mode, the terminal descent and impact attenuation hardware are designed by the requirements related to a 'hard' landing. In the case of a launch abort or terminal descent control failure, however, water splashdown is unavoidable. The biconic shape of the crew module minimizes impact deceleration if water entry occurs nose-down, so provisions must be included in the parachute system for achieving this attitude. Other provisions for a water spashdown include flotation bags and associated inflation device of sufficient size to right the module and keep escape hatches well above the water level.

2.4.3 Subsystem Description

An overview of the vehicle subsystems is given in Figure 2.4.3-1 and a description of the subsystems follows:

a. Main Propulsion System – Main Propulsion - The selected ground-based STV main propulsion system is a LO2/LH2 system and uses a total of six advanced expander-cycle engines with a vacuum thrust of 15,000 lb per engine, and an assumed specific impulse of 481 seconds. It includes the engines with electromechanical actuation, as well propellant delivery, pressurization, fill, and vent systems.

b. Reaction Control System – Reaction Control - The reaction control system is a gaseous O2 / gaseous H2 system with an assumed specific impulse of 410 seconds. It includes four GO2/GH2 thruster modules on the delivery stages and four on the crew module, with associated accumulators, pressurization, and control.

c. Electrical Power – The electrical power system is similar to the space-based vehicle version, but also includes battery power on the crew module for a power supply during reentry and landing.

d. Avionics

(1) Guidance and Navigation – Provisions for lunar mission operations, including rendezvous, docking, and lunar landing, with built-in redundancy for piloted operations.

Aeroshell

- TPS
  - Fibrous ceramic
  - Mechanical Attach.

Transpiration cooling

Drop Tanks

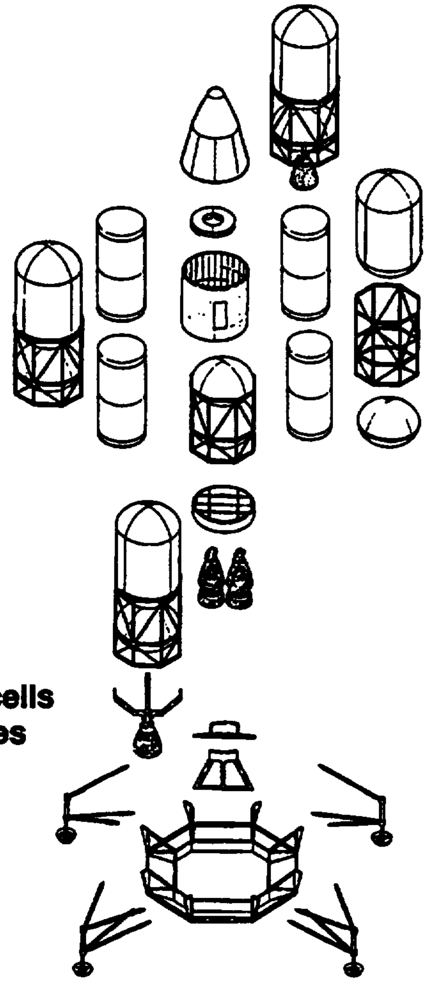
- Aluminum-Lithium
- Titanium fittings
- MLI blankets
- Meteroid protection

Power

- GH/GO fuel cells
- Ni/H2 batteries

Reaction Control

- GO2/GH2 thrusters
- Accumulators fed from main tanks



Crew Module

- Four crew
- Two EVA hatches
- Open ECLSS
- ALSPE shelter
- Parafoil
- Landing gear

Structures & Mechanisms

- Aluminum
- Graphite composites
- Crush core shock struts on landing gear

Propulsion

Advanced Space engine
- Thrust = 15000 lbf

Figure 2.4.3-1 GB Concept Subsystem Overview

(2) Communication and Data Handling – Provisions for communication, vehicle health maintenance, and data handling, with audio/video interfaces for piloted operations and instrumentation for drop-tank monitoring and control.

(3) Displays and Controls – Provisions on the crew modules for limited crew control and status monitoring of the vehicle during critical phases of the mission.

e. Environmental Control and Life Support (ECLSS) – Includes provisions on the crew module for atmosphere supply and control, internal equipment cooling, as well as metabolic and equipment heat rejection, similar to the 90-day reference concept. In this case, though, all life support functions are contained in a single crew module.

The system is an open loop life support system, with no regeneration of either atmosphere or water. Atmospheric gases are supplied from storage and from the fuel cell reactant supply accumulators, and carbon dioxide is removed from cabin air by replaceable LiOH canisters.

An active thermal control (ATC) loop is incorporated into the environmental control system, with coldplates for electronic equipment cooling, a cooling water loop for cabin thermal control, a Freon loop to cool vehicle heat loads, various equipment heat exchangers, and a variety of heat rejection devices designed for specific mission phases. Heat rejection devices include ground support equipment (GSE) heat exchangers, water flash evaporators, and space radiators.

f. Personnel Provisions – Personnel provisions include food, water, and waste management systems, as well as fire detection and crew furnishings, similar to the 90-day study reference. The food management system provides for the storage, preparation, and preservation of food for the crew. The food is shelf-stabilized and is prepared using warm water and heated in a convection oven, similar to the shuttle.

The water management system provides for potable water during the mission duration, and includes a water storage tank with water drawn from the fuel cell by-products, water dispenser, as well as tanks with a contingency water supply. The waste management system for both space- and ground-based vehicles includes a partitioned zero-g commode / hygiene station with waste storage tank and pre-moistened wipes for personal hygiene.

Crew furnishings include flight seats, emergency medical / health provisions, and personal equipment storage provisions. The flight seats, similar to those on the STS Orbiter, provide restraint and impact attenuation for all phases of flight and can be removed and stowed during flight .

2.4.4 Performance

The current ground-based vehicle concept can either deliver 11,630 kg cargo to the lunar surface in a piloted mode or 43,443 kg in a cargo delivery mode. With this cargo split, a total of 418 tonnes of cargo is delivered to the lunar surface over 21 piloted and 4 cargo-only missions, and the sizes of the vehicle flight elements are common to both piloted and cargo-only missions. As was already mentioned, the ascent tankset is not required for the cargo-only lunar mission.

Cargo delivery capabilities of various configurations of the ground-based STV concept are given in Figure 2.4.4-1 as well as CNDB mission payloads and delta-V's. For capture of non-lunar unpiloted missions the delivery stage portion of the lunar vehicle can be used as an independent vehicle. Piloted missions are shown with dashed lines connecting delivered mass (including return stage, crew module, and delivered payload) to delivered payload, and return mass (including crew module and return payload) to return payload quantities. The sample return mission (C1) is captured completely by the ascent stage only. This stage is also adequate for both the lunar (L3) return and GEO servicing (G2) return. To deliver the return stage, crew module, and payload for the G2 mission, a combination of descent stages and lander platform is required. To deliver the lunar return stage and lunar cargo, the full lunar vehicle is required.

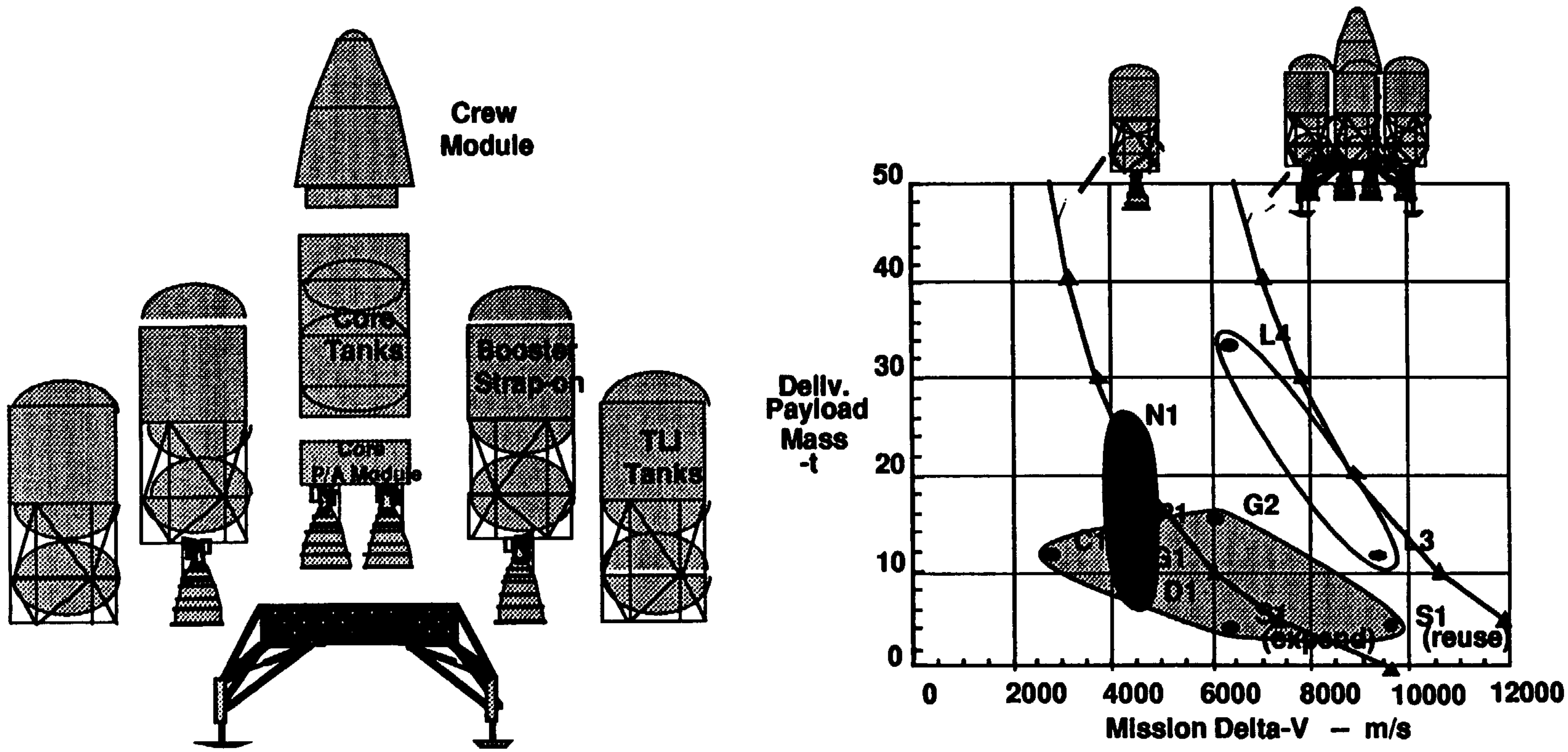


Figure 2.4.4-1 Modular Ground-based Vehicle Capabilities

2.4.5 Ground-Based LTS Cost Estimates

The STV ground based systems which were selected for the LTS mission excluded aerobrake hardware. Instead, the crew module was designed to return in a ballistic trajectory to the Earth launch point after return from the Moon. Therefore, only the cost data associated with advanced thermal protection panels was useful from the space-based vehicle cost estimating exercise.

Other space-based configuration cost data used in developing the final two preferred ground-based configurations are: the main propulsion engine planning estimates (all three systems are designed to use the Advanced Space Engine); the drop tank estimates (maturity and complexity factors were similar, but the sizing was different); fuel supply components unit costs (valves, regulators, etc.); and power, life support, and avionics components (fuel cell unit cost estimates, partially closed life support system hardware, inertial guidance hardware, etc.)

a. Drop Tank Estimates – The drop tanks description for the ground-based systems require less thermal protection than the space-based and the diameters were all common which benefits from design repeat factors and reduces overall hardware acquisition cost. Tank structures cost estimating relationships were calibrated to the external tank, Saturn 1C and COLDSAT study data (see space-based estimate writeup.) The fluid supply system for the dual launch ground-based vehicle option was a little more expensive due to the requirement to transfer LOX fuel in low Earth orbit.

The launch system also required a tanker flight before the flight vehicle was launched with the crew on the next launch. The costs of the two launch system were higher than the single launch ground-based vehicle design due to the added expendable hardware and launch costs associated with the tanker. The tanker development estimate was coordinated with the drop tanks estimate to take advantage of common design and development processes for the same hardware components on both flight elements. The TLI drop tanks for the ground-based units were the same net development estimate as the space-based tanks (390 million in 1991 dollars.)

b. Biconic Crew Module – The ground-based crew module estimate was generated from the LTS configuration weight statement description and prior

Personnel Launch System study cost data. The PLS data was generated just prior to the STV phase I final cost estimates.

The PLS database of avionics unit costs, structures and thermal protection estimates and parafoil chutes information from the vendor (Pioneer Systems) gave the cost analysts a great advantage at developing the estimate in a shorter period of time. Avionics was moved outside the reference PLS design to accommodate the life support and crew provisions for the longer lunar mission requirements. The biconic crew module description is the same for both the single and dual launch ground-based systems.

Figure 2.4.5-1 is a copy of the cost estimate output (in 1991 dollars) from the Boeing Parametric Cost Model (PCM) for the biconic crew module development phase.

c. Creating the Production Estimate – The method of extending the ground-based system production first unit costs is summarized in figure 2.4.5-2. Note that the cost improvement curve application changes for the different system flight hardware elements.

d. Final Ground-Based Estimate Summary – Figure 2.4.5-3 contains the final estimate in 1991 dollars for the dual launch and single launch systems.

The summary ground processing flow for the dual launch system is presented in figure 2.4.5-4. Many hours of systems analysis and cost model setup activities, using designers' and operations analysts' descriptive system characteristics data and technical resource estimates, are required to develop the one simple looking LCC summary chart. Without the design description information at a *subsystem* level, the operation and support and launch facilities estimates have little credibility.

GROUND BASED LTV CREW MODULE
1991 DOLLARS IN MILLIONS

	ENGR	MFG	TOTAL
CREW MODULE STRUCT & MECH	176.545	353.018	529.562
CREW MODULE RADIATION PRO	71.454	65.962	137.416
CREW MODULE REACTION CTRL	8.694	33.252	41.945
CREW MODULE ELEC POWER	28.343	21.415	49.758
CREW MODULE AVIONICS	80.865	125.448	206.313
CREW MODULE ENVIRONMENTAL	28.166	114.479	142.645
CREW MODULE WEIGHT GROWHT	56.237	138.897	195.134
CREW MODULE WEIGHT GROWHT	7.648	0	7.648
HARDWARE FINAL ASSY & C/O	-	129.581	129.581
SPARES		85.247	85.247
HARDWARE TOTALS (FROM ABOVE) ($M)	457.949	1067.299	1525.248
SUPPORT COST (M$)	ENGR	MFG	TOTAL
SYSTEM ENGINEERING & INTEGRATION	65.058	-	65.058
SOFTWARE ENGINEERING	0	-	0
SYSTEMS GROUND TEST CONDUCT	126.422	-	126.422
SYSTEMS FLIGHT TEST CONDUCT	78.328	-	78.328
PECULIAR SUPPORT EQUIPMENT	64.494	46.46	110.954
TOOLING & SPECIAL TEST EQUIPMENT	-	518.432	518.432
TASK DIRECT QUALITY ASSURANCE	-	67.139	67.139
LOGISTICS	31.555	-	31.555
LIAISON ENGINEERING	33.323	-	33.323
DATA	13.755	-	13.755
TRAINING	1.495	-	1.495
FACILITIES ENGINEERING	3.417	-	3.417
SAFETY	1.068	-	1.068
GRAPHICS	2.349	-	2.349
OUTPLANT	1.068	-	1.068
PROGRAM MANAGEMENT	0	0	0
SUPPORT EFFORT TOTAL ($M)	422.333	632.03	1054.362
TOTAL ESTIMATE ($M)	880.282	1699.329	2579.612
SCHEDULE PENALTY ($M)	0	67.973	67.973
TOTAL ESTIMATE (THIS SCHEDULE) ($M)	880.282	1767.302	2647.584

Figure 2.4.5.-1: Ground Return Crew Module Estimate Summary Output from the Boeing PCM

LTS GROUND-ORBITAL CONFIGURATION PRODUCTION ESTIMATE					
	(Constant-Year 1991 Dollars in Millions)				
		Learning	1st Unit Cost	Cumulative	Production Totals
Flight Element Hardware	Quantity	Curve	(91$M)	Learning Factor	(LTS + Tug Only)
LTS Core Stage Hardware	24	90%	373.8	17.10016	$6,392
Reusable Avionics Pallet	4	100%	80.4	4.00000	$322
Advanced Space Engine	144	95%	6.3	107.42655	$672
Trans-Lunar Drop Tanks	48	90%	60.3	31.03674	$1,872
Liquid Oxygen Tanker	24	90%	78.4	17.10016	$2,312
Biconic Crew Module	4	100%	324.8	4.00000	$1,299
CM Launch Escape Sys.	21	90%	66.7	15.23729	$1,017
CM Docking Adpt./Tower	21	90%	10.0	15.23729	$152
Total LTS Mission Hdw. -					$14,038
LTS Space Tug Derivative	18	90%	52.2	13.33436	$696
RL10-A4(+) Engine, Tug	18	100%	2.9	18.00000	$52
Total SEI Tug Hdw. -					$748
Subtotal Production -					$14,786
NASA Program Factors					$7,393
Grand Total, Estimate -					$22,179

Figure 2.4.5-2: A Production Estimate is Developed for the Ground-Orbital LTS Candidate on an Excel Spreadsheet

(1991 $ IN MILLIONS)		GROUND-ORBITAL LTV OPERATIONS CONFIGURATION			GROUND BASED LTV (SINGLE LAUNCH)		
		N/R	REC	TOTAL	N/R		
7.0	LTS Project						
7.1	LTS Hdw. Integ. (NASA Prog. Supt.)	1,232	352	1,584	1,154		
7.2	LTS Dev./Production						
7.2.1	LTS Prog. Mgmt. (Req. Factor & Fees)	4,500	7,041	11,541	4,217	8,134	
7.2.2	LTS System Engr.	203	-	203	179	1,872	
7.2.3	LTS Flight Hdw.					-	
7.2.3.1.1	LTV. Core Stage, Tug, ASE*	4,184	8,134	12,318	4,184	2,468	
7.2.3.1.2	LTV Drop Tanks	345	1,872	2,217	345	(in 7.3.1)	
7.2.3.1.3	LTS Tanker	823	2,312	3,135	-	-	
7.2.3.1.4	LTV Crew Module	3,000	2,468	5.468	3,000	(TBD)	
7.2.3.1.5	LTV FAIT	316	(in 7.3.1)	316	316	-	316
7.2.4	LTS Support Equip.	348	-	348	308	-	308
7.2.5	LTS Payload Accom.	(TBD)	(TBD)	-	(TBD)	-	-
7.2.6	LTS Software	1,875	-	1,875	1,775	-	1,775
7.2.7	System Test Ops.	508	-	508	473		473
7.2.8	Ground Ops. & Ctrl. N/R	358	-	358	358	873	358
7.2.9	LTS Mission Ops.	296	-	296	296	2,834	296
7.3	LTS Operations & Support					104	
7.3.1	LTS Hardware O&S Processing		977	977		(TBD)	873
	LTS O&S Mission Support		3,390	3,390		(TBD)	2,834
	LTS Personnel Training		104	104		2,153	104
	LTS Recovery Support		(TBD)	-		36	-
	LTS Non-nom. Ops. Support		(TBD)	-		692	-
	LTS O&S Logistics Services		2,224	2,224		1,656	2,153
	LTS Consumables & Expendables		37	37		18,919	36
	LTS Software Maint.		732	732		-	692
	LTS Base Ops. Support		1,136	1,136		45,978	1,656
	ETO O&S Services	2,771	25,282	28,053	4,528		23,447
	Low Earth Orbit Support	-	-	-	-		-
		20,759	56,061	76,820	21,133		67,111

*Note: Includes Management Factors from NASA; Engine is GFE to Prime Contractor

Figure 2.4.5-3: Ground Return LTS LCC Estimates Comparison (Dual vs. Single Launch)

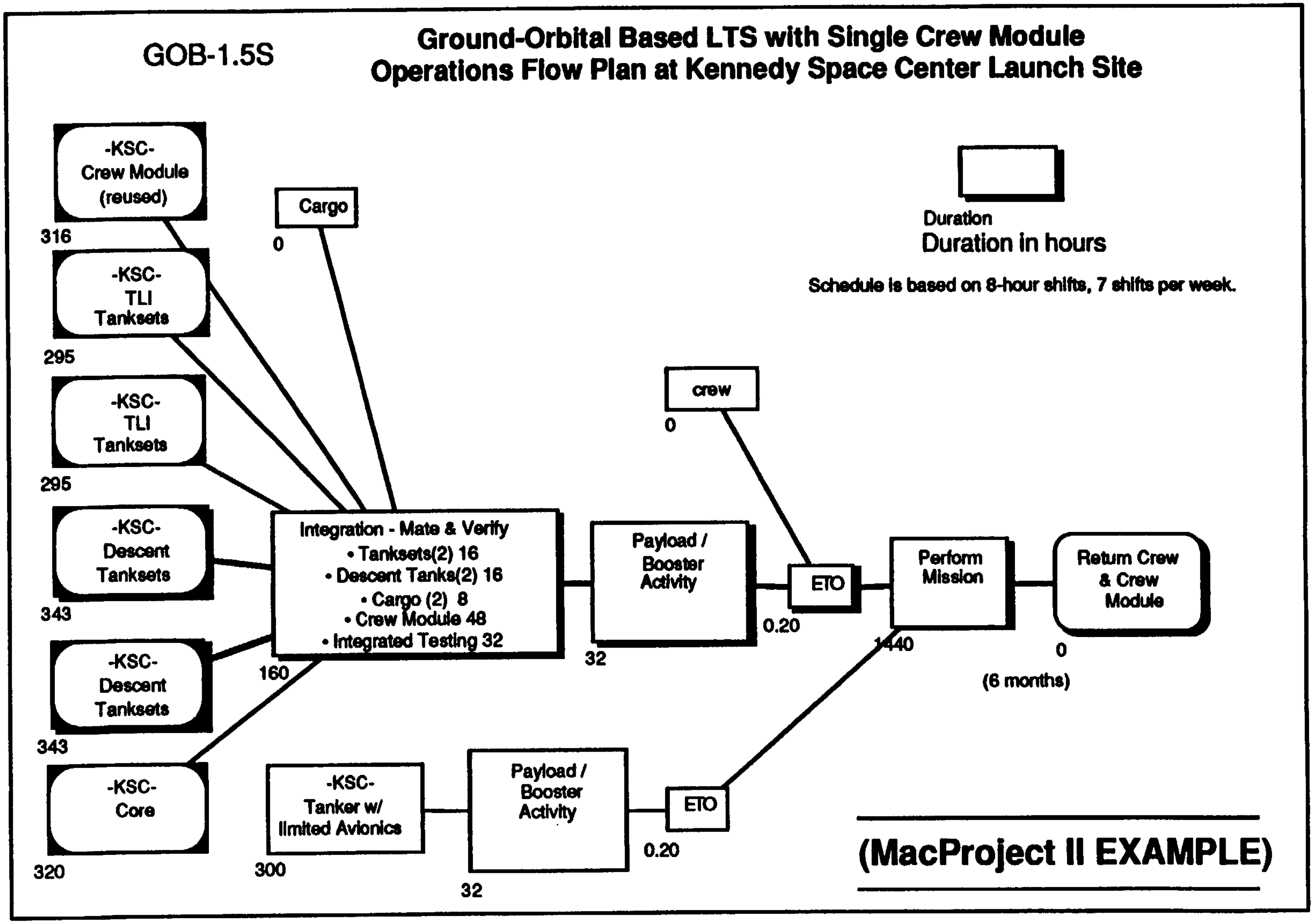


Figure 2.4.5-4: Flow Diagrams for LTS Mission Vehicles are Required to Estimate O&S

2.5 Evaluation Methodology

Program- and system-level trade study and analysis methodologies are presented in this section. The definition of concepts to be examined in this architecture study started with an assessment of orbital options based on use of different basing locations and transportation nodes. The types of trades exercised in this architecture study are as follows:

a. Number of stages.

b. Crew module approaches.

c. Basing approaches.

d. Lunar approach trajectory.

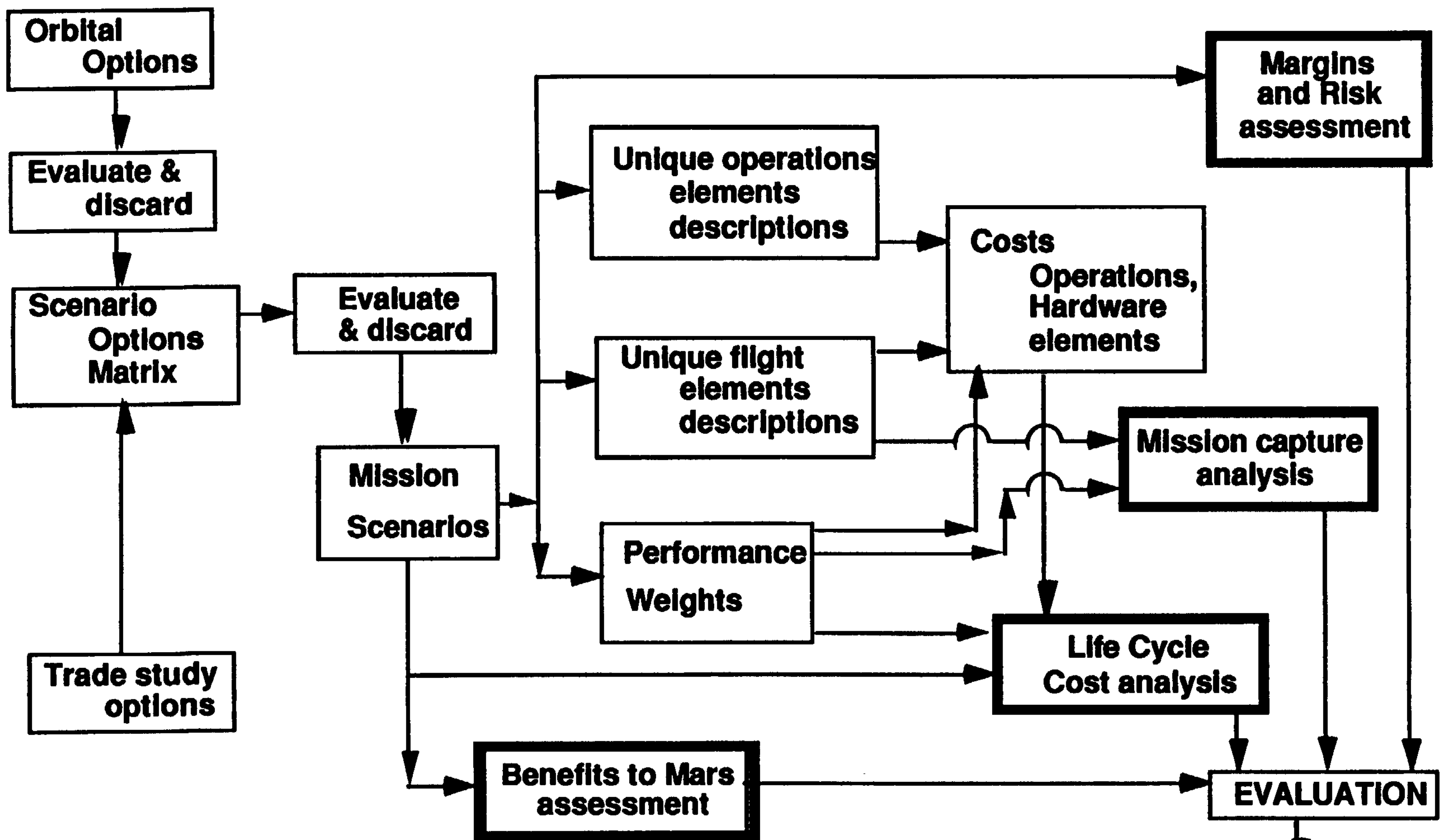
e. Aerobrake versus all-propulsive return.

f. Use of droptanks versus propellant tankers

Options defined for the six architecture trades were combined in a matrix resulting in over 400 possible architectures. Groundrules and assumptions were applied to reduce these combinations to 94 architectures for which performance and mission scenarios were developed. Based on this work, 29 scenarios were selected and initially assessed against the cost and margins and risk evaluation criteria to determine trending. Based on the observed trends, 13 additional scenarios were initially included and one was added later. The resulting 43 scenarios were fully evaluated against the four evaluation criteria to determine the preferred architectures. An overview of this process is shown in Figure 2.5-1.

Using the mission scenarios, unique flight elements were identified and characterized. A functional split was made between flight elements to distinguish mass and subsystem definitions, as well as unique hardware and operations. The ultimate goal was to identify concept differences that distinguished hardware and operations costs.

The process for defining unique flight elements to support the cost assessments included a description of all vehicle options identified in the mission scenarios, an analysis of mission functions to identify functionally unique flight elements, and a mass



Our evaluation process uses a common database for the 4 evaluation assessments

Figure 2.5-1 Architecture Study Evaluation Process definition of unique flight elements to support the cost analysis. The flight element definition process is shown in Figure 2.5-2.

2.5.1 Performance

In parallel with the flight element definition analysis, mission performance of trade study options was calculated using mass trending data generated from a database of previous STV designs. The results of the performance analysis were then used to identify vehicle sizings and provide booster requirements for LCC analysis. As part of the mission performance analysis, a tank-drop optimization analysis was also conducted to determine when (i.e., after which major burns) the droptanks should be expended.

The performance analysis was designed to provide a good relative comparison between concepts as to ETO mass requirements and mass in LEO and LLO. These mass values changed for the downselected vehicle designs as they were developed and optimized. However, the relative differences identified between the scenarios indicated the performance differences would remain essentially the same as any of the different scenarios were optimized.

a. Tank-Drop Optimization Analysis – As part of the trade study analysis, an optimization of tank-drop event numbers and location was performed for 1.5-stage (direct to lunar surface) and 2.5-stage (LLO node) vehicles to check initial assumptions made in the mission scenarios and to provide a basis for future tank-drop assumptions. The analysis was performed for both space-based and ground-based options, using single crew modules for the direct to lunar surface cases and dual crew modules for the LLO node cases. For each case, all combinations of tank-drop events following major burns were examined, including no tank-drop events. For LOI droptanks, it was assumed that the droptanks would not be disposed of until after rendezvous with the lunar excursion vehicle following the lunar surface operations.

Droptank disposal can occur with TLI and TEI droptanks disposed of by reentry into the atmosphere or by being boosted out of the Earth-Moon system. The latter option is accomplished prior to midcourse correction and is the preferred option. For LOI, LD, or LA droptanks, lunar surface disposal is the method of disposal.

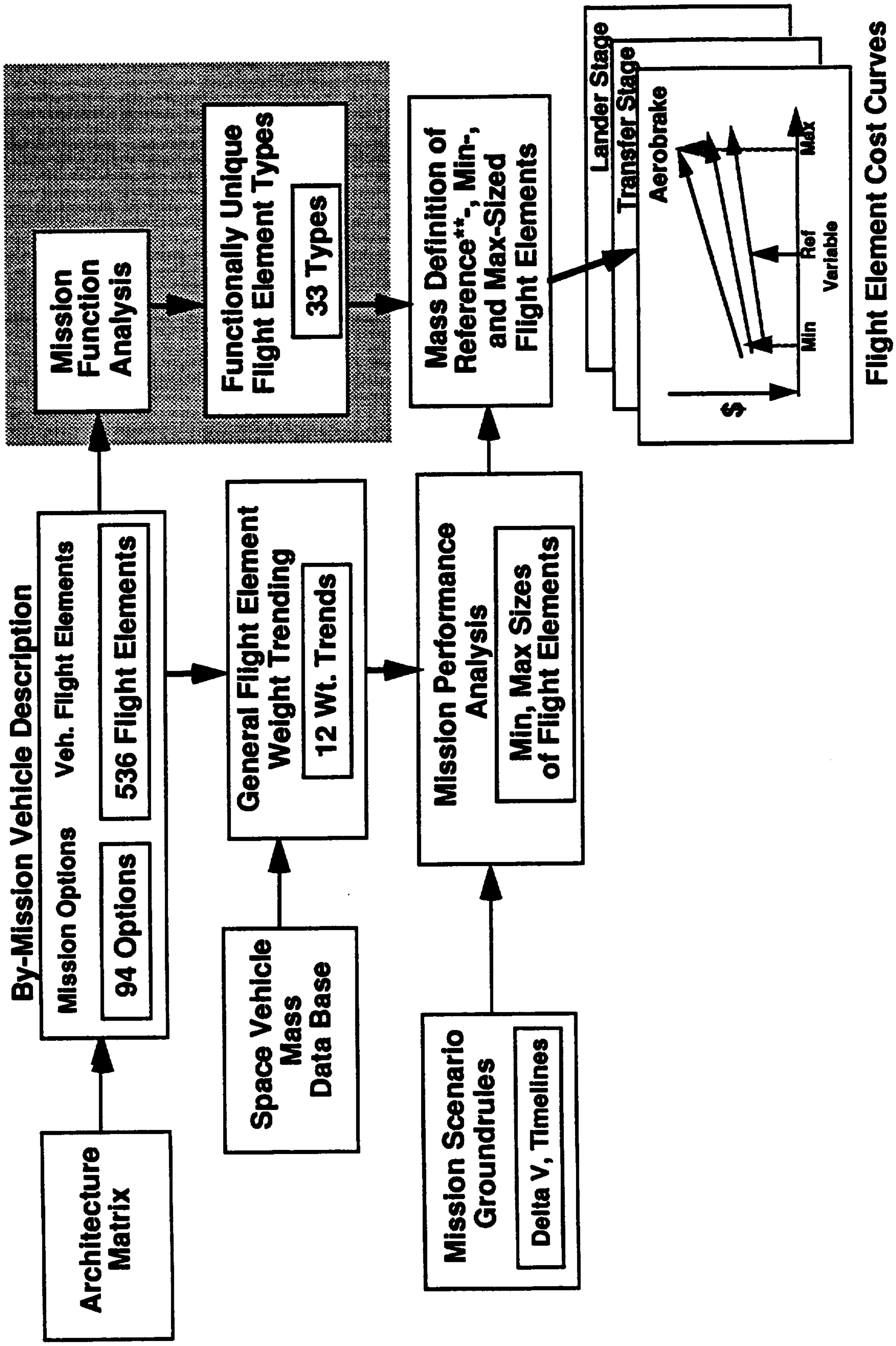


Figure 2.5-2 Flight Element Definition Process

2.5.2 Cost Trade Study Methods

As described in the previous "funding" section of this document (see 1.2.2), the initial phase 1 cost analysis effort was postured to support the system level design trades at a Lunar Transportation System (LTS) architecture level. An estimating plan was conceived to calculate the cost of many different STV candidate configurations using a modular design description approach for parametric cost model inputs. Figure 2.5.2-1 illustrates the overall STV cost estimating methods devised for for phase I system trades support. The Excel © spreadsheet model (shown as a matrix output table of "relative costs" in the lower right hand corner of the figure) was developed as a final compliment to the parametric cost modeling system, and the spreadsheet model was subsequently delivered to the MSFC study program office (after the presentation at the fourth interim review.)

The phase I cost analysis plan was to originally process up to 100 cost estimates for the candidate designs. (This prediction was close to the original down-select of over 240 concepts to about 94 STV design concept finalists!) The figure shows design analysis personnel as providing the: "*Space System Platform ...*" description and drawings set (for cost model global inputs derivation); equipment lists in the form of informal "*Hardware Cost Data Sheets* " or block diagrams; and "*Hardware Mass Prop. & WBS Correlation* " tables of component level weight estimates (with materials and subsystem content assumptions for each line item in the weight estimate tables.)

The weights estimator and configuration design personnel worked very hard to modularize the design description inputs in an orderly manner. Meanwhile, the systems manifest evaluation analyst developed the LCC accumulator spreadsheet and the cost parametricians developed the "straw man" PCM loading files and global inputs sheets for over 100 PCM cost runs of the various flight elements (separate runs were made for both development and theoretical first unit estimates.) All initial trade study estimates were calculated in constant-year 1989 "relative" dollars (as was previously stated, later in the phase I study the IR#5 final STV cost estimates were produced in 1991 dollars.)

The system cost trade study process during phase I was further designed to use maximum and minimum flight element scaling descriptions for the development of hardware cost trend curves. Then, flight element cost trend curves would be created to reduce the number of Boeing Parametric Cost Model (PCM) runs and the number of

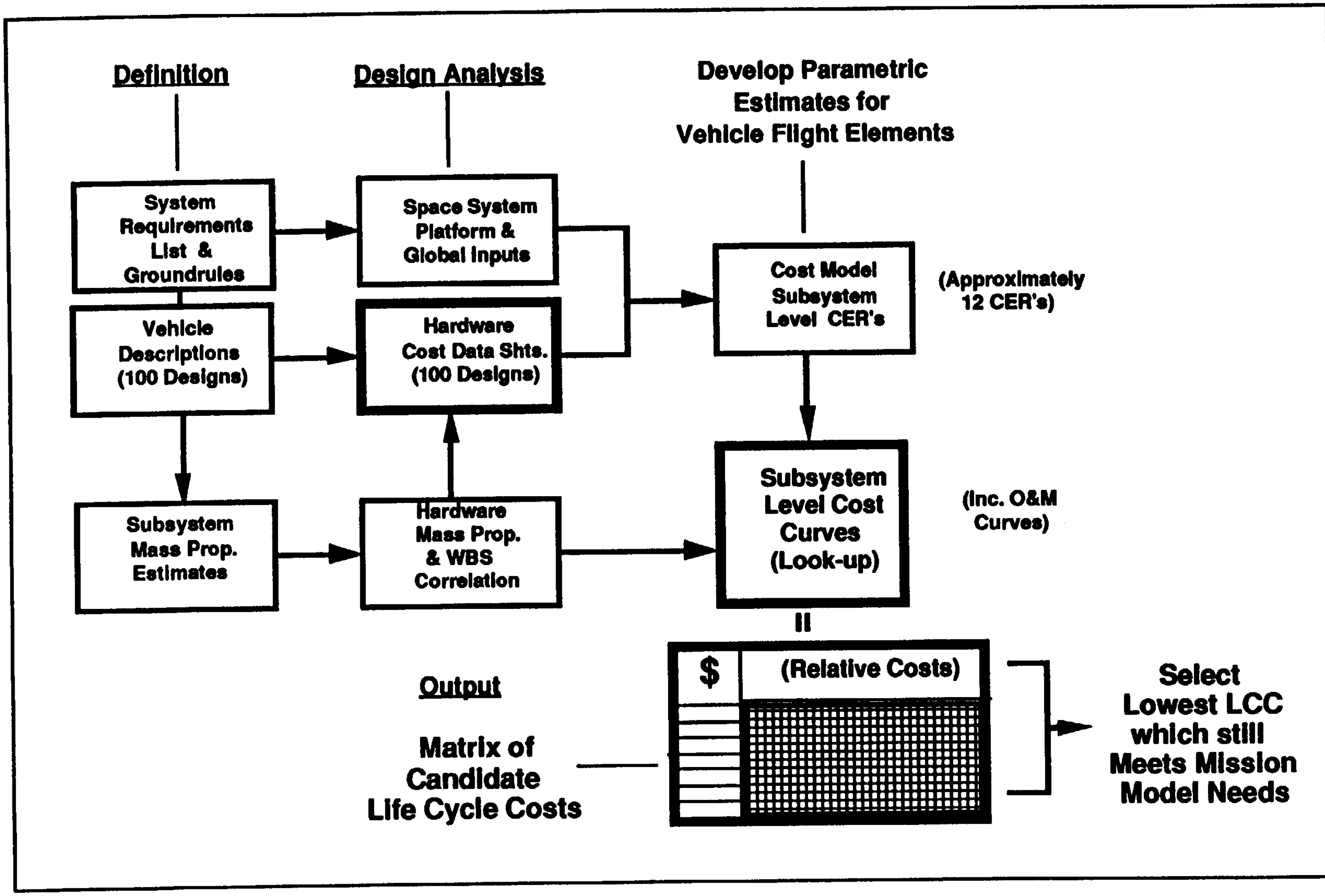


Figure 2.5.2-1: An Initial STV Cost Trade Study Plan is Developed (ref.: COTR Briefing prior to IR#3, May, 1990)

mass properties estimates required for definition of all options. The 96 candidate configurations were reduced to just over forty configurations using a technical screening process. The technical screening process was used to select those designs which were pertinent in trading several basing and mission operation alternatives of special interest to our MSFC customer. (A majority of the remaining designs from the original 94 candidates would be traded at a later date with new boosters.)

Figure 2.5.2-2 depicts this system trades estimating process steps presented at IR#3 briefing by Boeing. Figure 2.5.2-3 illustrates the overall complexity of the trade study cost analysis and design descriptions support effort. The two charts summarize the trade study plan and results which led the Boeing team to select the final three preferred STV candidates for the LTS mission, two ground-based and one space-based vehicle. The three basic finalist designs were presented at the IR#5 STV briefing.

Figure 2.5.2-4 contains a list of the final set of "Boeing-preferred" STV design LTS candidates (meeting the customer-provided mission model and requirements groundrules) and their respective LCC estimates (in "relative" 1989 dollars.) The final phase I cost estimates (in 1991 dollars) for the final three preferred choices, and the 90-Day Study reference vehicle, are explained in the preceding subsections.

2.5.3 Risk and Margins

The STV system and each of the subsystems will be designed with margins for all contingencies. In addition, risks for each mission operation and each mission phase will be mitigated as much as possible using modern engineering techniques. However, some system configurations will inherently have margins and some system configurations will inherently mitigate risks simply because the architecture avoids particular situations during the mission profile. The margins and risk evaluation attempted to identify and quantify the risks and margins that are discriminators between the scenarios.

The breakdown in weighting between risk and margins and the respective subcategories is shown in Figure 2.5.3-1. The risk area is broken into equal weighting between technical and programmatics risk. Technical risks deal with the risk during the operational phase and include such things as mission success, performance and

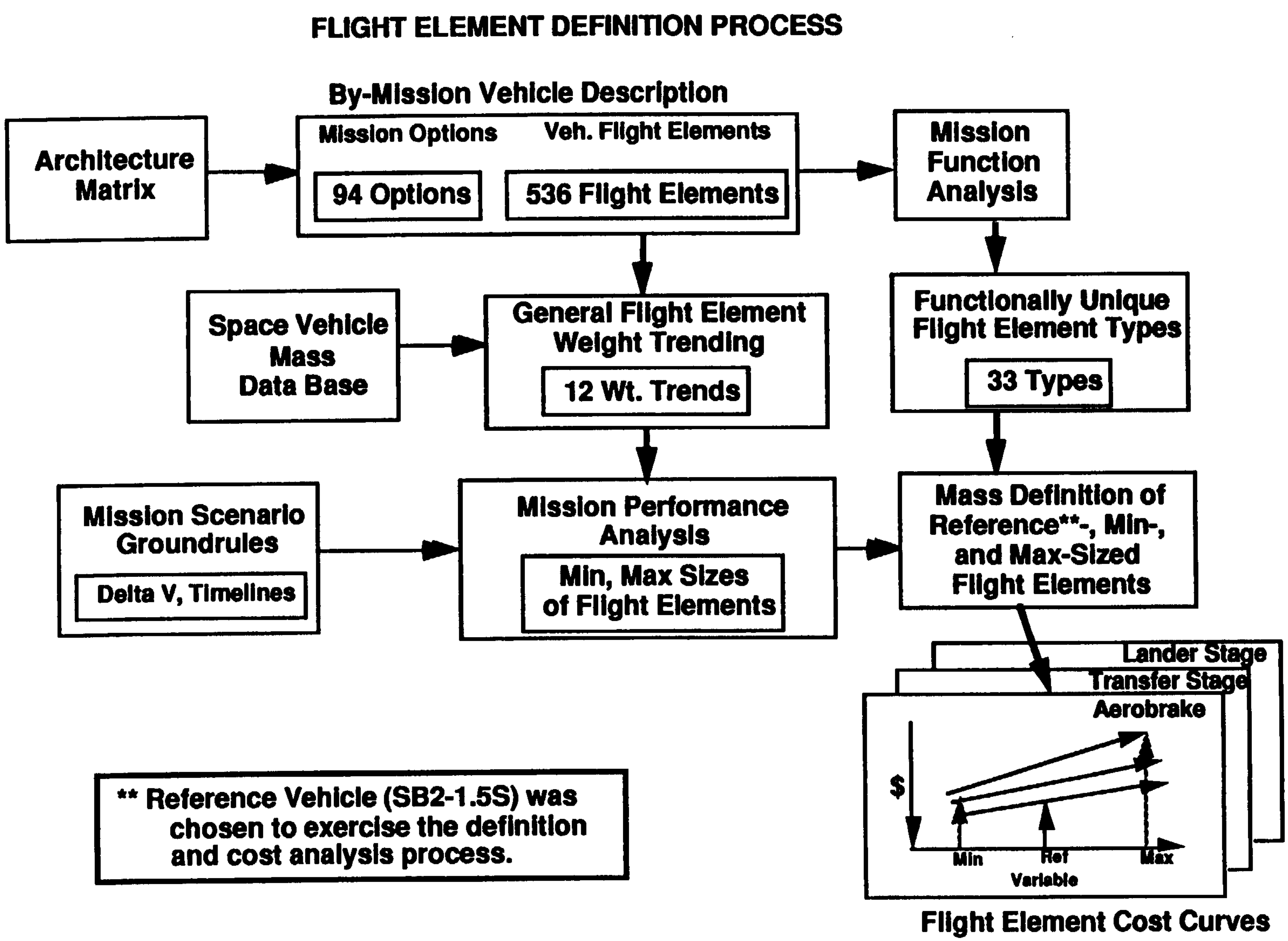


Figure 2.5.2-2: A Cost Trades Process is Presented at the Study Midterm Review
(ref.: IR#3 Boeing Briefing Handout, June 20, 1990)

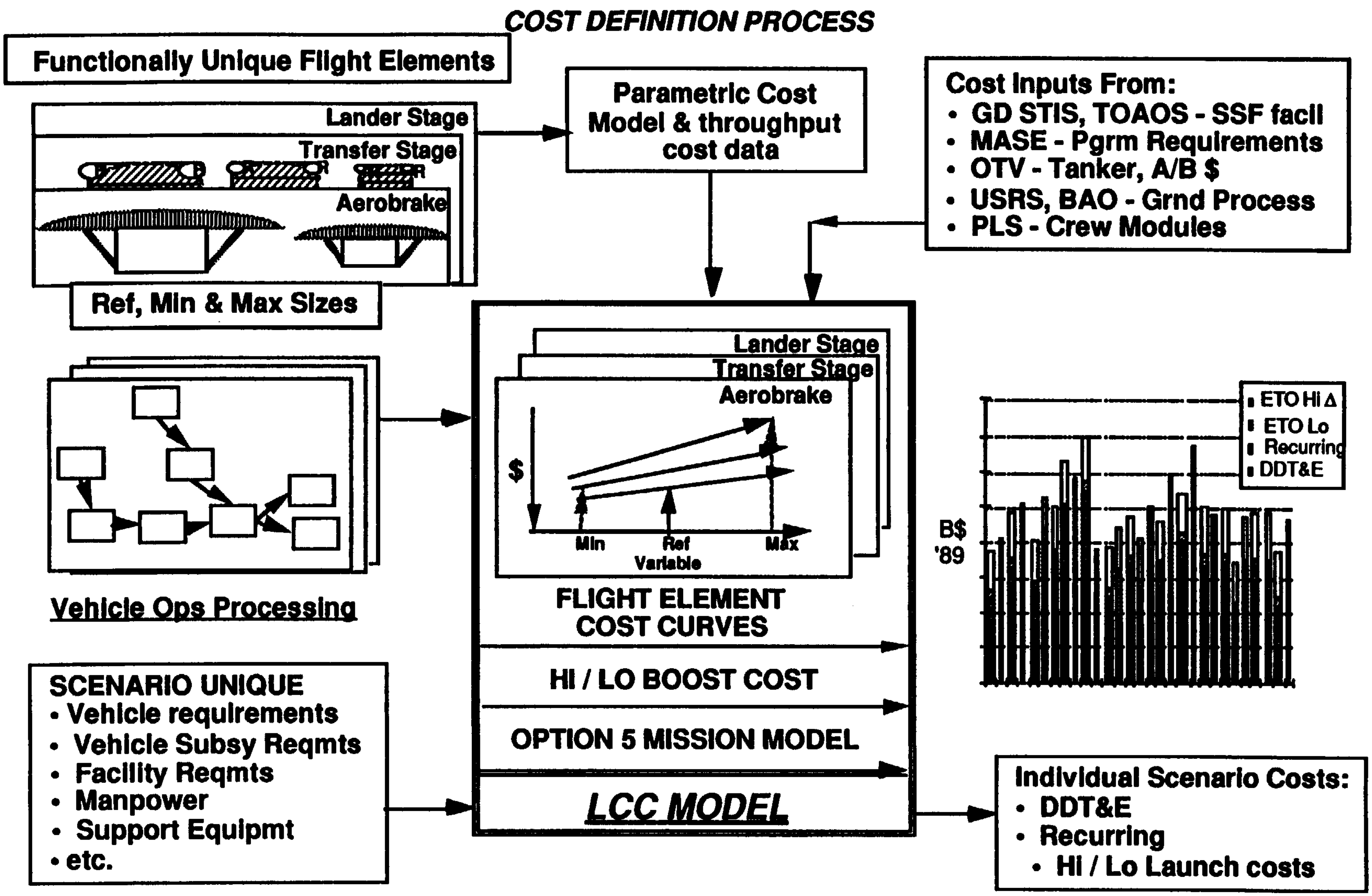


Figure 2.5.2-3: Overview of Final Phase I Trade Study Process is Presented (ref.: IR#4 Briefing Handout, October 18, 1990)

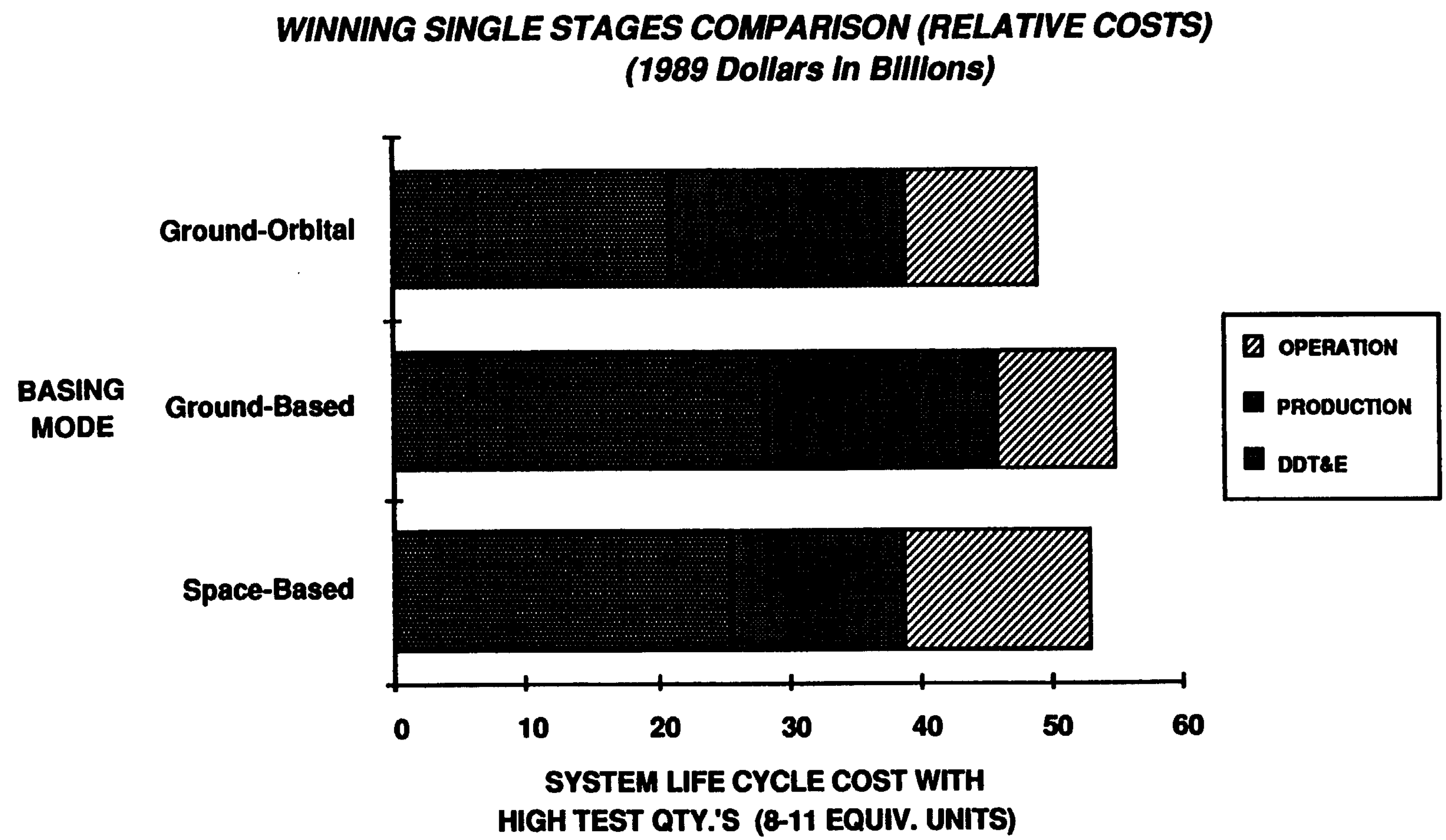


Figure 2.5.2-4: The Three Winning LTS Mission Candidate LCC Estimates are Summarized (ref.: STV Phase I Final Report, Vol. III, Book 1, 4/91)

Evaluation criteria	
Cost	50%
Risk & margin	30%
Evolutionary mission	15%
Mars mission	5%

Screen used for 6/20/90 down select (Cost, Risk & margin)

Risk categories	Risk weight = 20%
Technical	50%
Programmatic	50%

Margin categories	Margin weight = 10%
Payload growth	35%
Operational flexibility	35%
Safety	25%
Repairability	5%

Figure 2.5.3-1 Scoring - Risks and Margins

operation, and safety and reliability. In general, the programmatic risk deals with the anticipated risk associated with the FSD program phase (i.e., cost and schedule). The technical risk category is further broken into 10 risk subcategories as shown in Figure 2.5.3-2, which are weighted as to their respective importance. Each system concept was given relative grades of either 1, 2, or 3 (1 for low risk, 2 for medium risk, and 3 for high risk) for each of these categories with low risk being best. Figure 2.5.3-3 contains the detailed definitions and respective scoring approach for all of the risk categories. The risks evaluated here exclude design for risk mitigation.

The five margin categories (mission growth, payload growth, operational flexibility, safety, and repairability) and the scoring rationale are shown in Figure 2.5.3-4. The margins evaluated here exclude design margins.

To support the cost and margins and risk assessments, and the subsystem design task, operations flows were developed for the mission scenarios. Operations were defined from the start of KSC processing of a new vehicle to the end of the mission on its second flight. This covers all major events, excepting final disposal, in the vehicle's life, including refurbishment for reflight. Figure 2.5.3-5 shows the operations element definition process.

A diverse source of inputs was considered in developing the operations flows. Studies have been performed in the past by several major contractors whose primary purpose was to define on-orbit operations of an OTV (STV of lunar vehicle). Operations were defined at a major task description level, with a ROM estimate of task duration hours assigned. Figure 2.5.3-6 demonstrates the difference in complexity between space-based and ground based scenarios. The number of operations steps required was considered as a minus in the risks and margins analysis task.

Technical	Weight = .50
• Space environment exposure time	.04
• In space assemble	.12
• Number of rendezvous	.02
• LEO node support	.02
• LLO node support	.02
• Micro g's propellant transfer	.04

Technical (continued)	
• Abort penalties	.12
• Dual ECLSS system available	.08
• Reuseable motor/CC	.02
• Reuseable aerobrake	.02

Programmatic	Weight = .50
• Operational Risk	.20
• Development	.30

Figure 2.5.3-2 Risk Categories

Technical Risk	Score	Comment
Time exposure to environment between refurb/support	=1 during mission flight only =18 for 6 month LLO =165 for multiple 6 month periods	P(Sucessful completion of 10 days)=.99999; P=.99999**18 for 6 months or 18 times more likely of unsucess; P=.99999**180 for 5 years or 165 times more likely of unsucess
In-space assemble	=n where n is the number of mating surfaces	Measures the inherent risk of mating surfaces
Number of rendezvous	=m where m is the number of rendezvous required during the mission	Measures the inherent risk of rendezvous. Mating of launcher & lunar surface required by all concepts.
Node support in LEO required	=1 independent of LEO support =2 on orbit assemble =3 LEO support mandatory	Mission critical support received for LEO node versus GB
Nodes support in LLO required	=1 independent of LLO support =3 LLO support mandatory	Mission critical support received for LLO node versus direct LS
Micro g's propellant transfer	=1 none required =3 required to accomplish mission	Systems with no transfer are judged to have less risk
Abort penalty	=0 Multiple propulsion stages at LS: +1 Two pressurized cabs: +1 Free direct return to earth (ballistic crew cab): +1	Each additional point indicates reduced risk so zero maps to 5 on a 1 to 5 scale and the score of 3 would be best (1)
Dual ECLSS system	=1 Hybrid CC =3 Dual or single CC	Hybrid cab versus dual or single cab configuration
Reusable motor and CC	=1 multiple reuse =2 some stages not used =3 no reuse of stages	Avionics and shuttle engine reuse has shown history or reliability confidence
Aerobrake reuse	=1 no reuse =3 multiple reuses	No multi use aerobrake currently available without GB checkout and refurbishment
Assemble in Space	=1 none =2 a few pieces =3 for full assemble	Space operations is viewed as inherently higher risk

Low = 1 (low risk is better) Medium = 2 High = 3

Figure 2.5.3-3 Technical Risk Definitions

Margin category	Score	Sample discriminators
Payload growth	=n where n is the number of MT payload capability to the lunar surface	Measures the margin above the minimum required payload capability
Operational flexibility	=1 otherwise =2 If daily but LLO rendezvous =3 If launch &return opportunities are daily	GB launch cancellation results in 24 hour delay SB launch cancellation results in 6 to 11 days delay
Safety	=m where m=N-(N-1)**n/N**(n-1) and where N=1/(1-.9999)	Safety complexity (n) for Rendezvous (2) aero-maneuver (10), Ballistic return (5), EVA (4), fuel transfer (3), proximity (1). Probability of safe mission is the product of .9999**n for each maneuver. m is the number of times less safe a mission is than .9999
Repairability	=1 =2 =3	Rated as 2 for this screening

Low = 1 Medium = 2 High = 3 (high margins are better)

Figure 2.5.3-4 Margin Categories

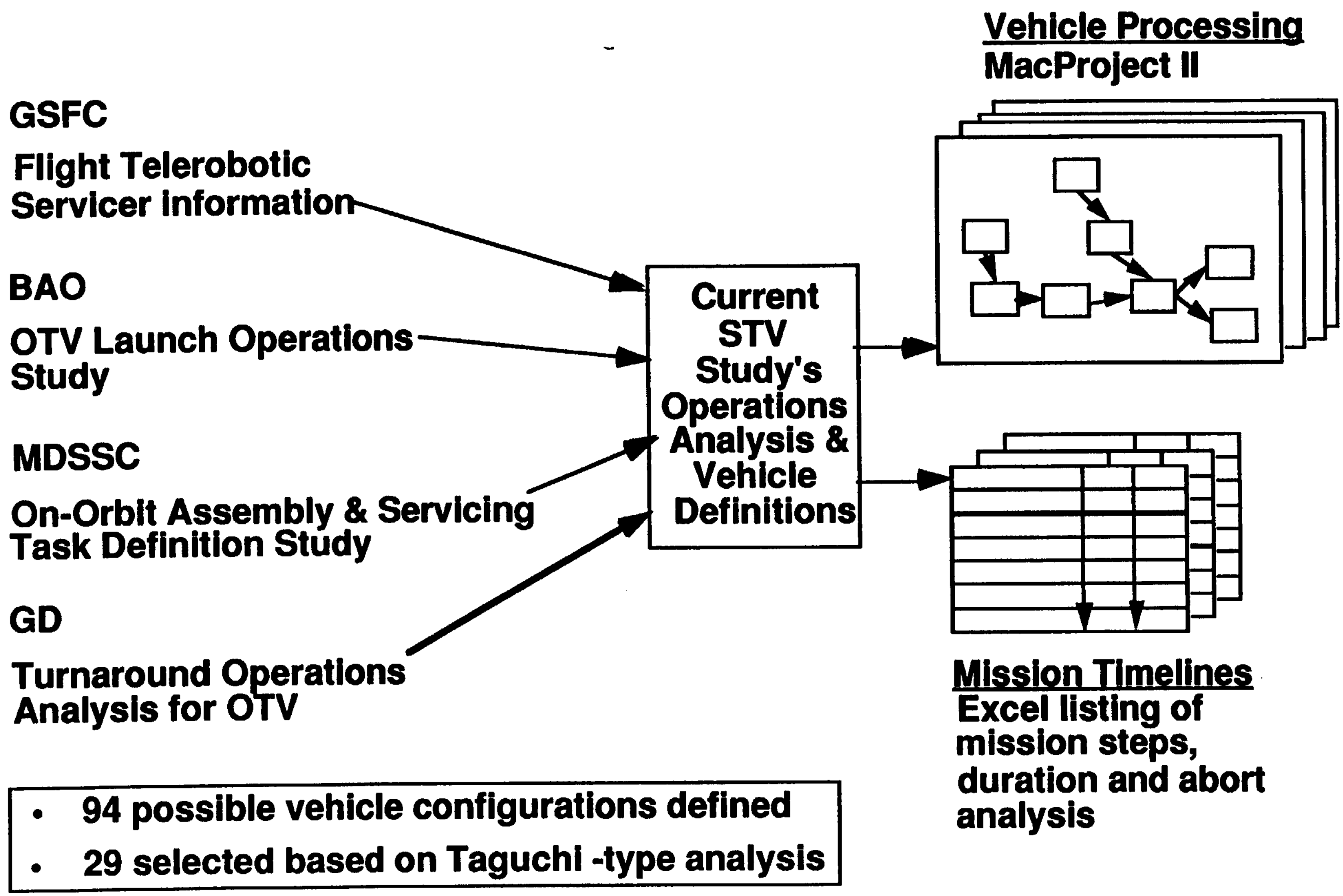


Figure 2.5.3-5 STV Phase I Study Operations Definition Process

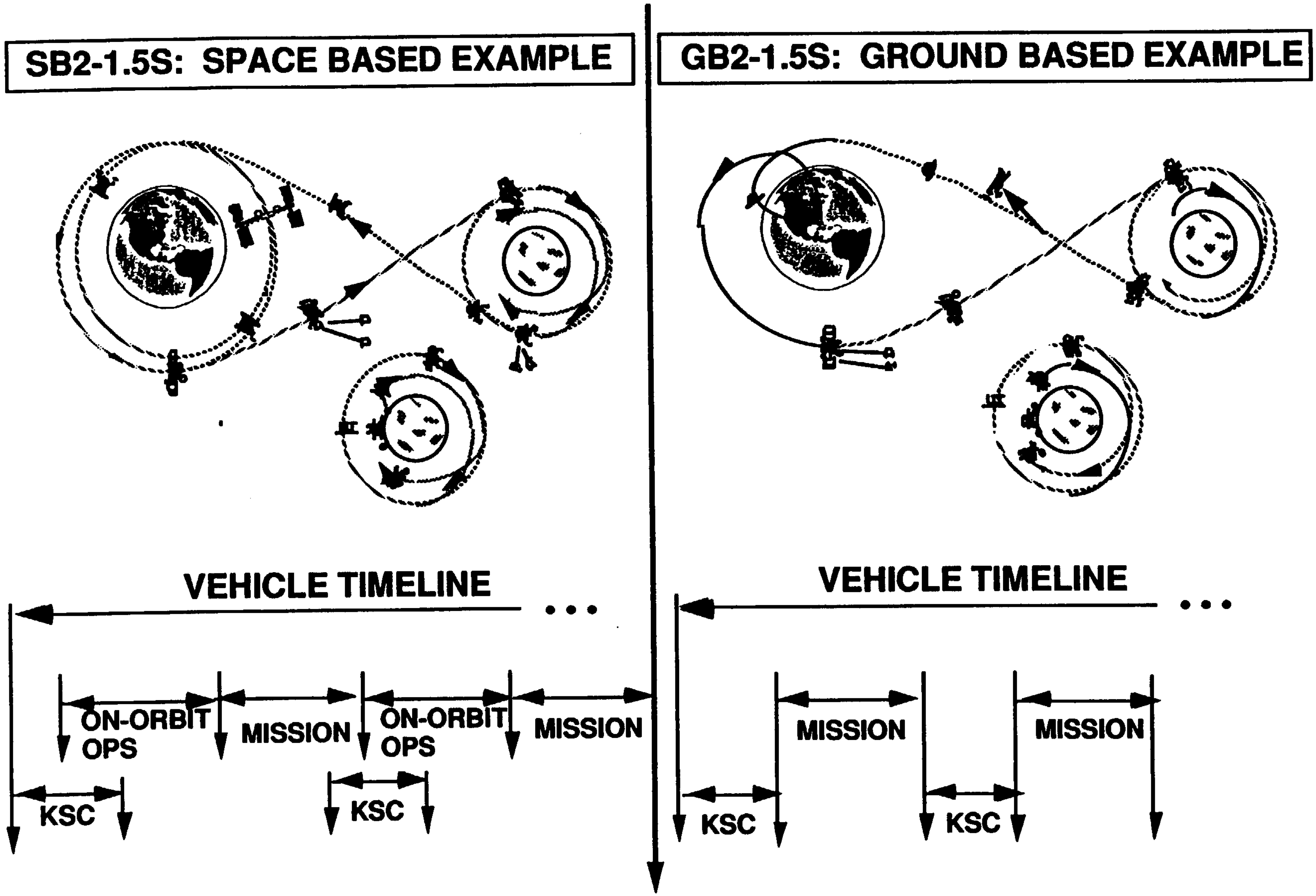


Figure 2.5.3-6 Breakdown of Operations by Phases (Complexity Comparison)

2.5.4 Applicability to Mars

The Mars mission benefit was one of the evaluation criteria for STV concept selection with a 5% weighting of the total evaluation criteria. The purpose of this analysis was to determine how much the STV concepts, designed for the lunar missions, can benefit the Mars missions and vehicle designs as they are projected at the current time.

Mars vehicle designs include a transfer vehicle (MTV) and an excursion vehicle (MEV). MTV options include cryogenic vehicles, nuclear energy propulsion (NEP) vehicles, solar energy propulsion (SEP) vehicles, and nuclear thermal rocket (NTR) vehicles. For this analysis, it was assumed that the MEV is cryogenic and has an aerobrake, no matter what the MTV type. Because the cryogenic MTV would benefit most from the lunar missions, it was chosen as the baseline for this analysis. To determine the overall benefit of each of the lunar vehicle concepts, specific benefits were weighted independently and scored and then combined with equal weighting for the MTV and MEV.

Types of Mars mission benefits were broken into subsystem- and system-level benefits (e.g., structures, aerobrake, and propulsion) and further into specific areas of benefit (e.g., landing gear, mate and demate umbilicals, and aerobrake on-orbit assembly) and then weighted independently for the MTV and MEV. These were then graded as to the level of benefit received from the lunar mission technologies (1 = technology benefit and 2= hardware or operations benefit). Figure 2.5.4-1 shows the areas of benefit and weighting for each of these areas.

The Mars vehicle weighting for each system or subsystem item was multiplied by the lunar vehicle benefit and summed to achieve a total score for each lunar vehicle concept. The scores for the MTV and MEV were then weighted equally to yield the overall Mars benefit score for each lunar vehicle option.

Overall Weighting

Benefit to Mars Missions	5%

Mars Vehicle Type Weighting

	MTV (50%)				MEV (50%)
	CRYO	NEP	SEP	NTR	CRYO
Structures and Mechs	8%	17%	19%	8%	8%
Aerobrake	15%	0%	0%	8%	16%
Propulsion	16%	16%	16%	12%	16%
Propellant Management	12%	0%	0%	12%	12%
Power	4%	30%	28%	13%	4%
Avionics	8%	8%	8%	8%	8%
Habitation / Crew Interface	8%	8%	8%	8%	8%
Robotics	10%	10%	10%	10%	9%
Integration and Test Ops	6%	6%	6%	6%	7%
Transportation Infrastructure	13%	5%	5%	15%	12%

Not used for scoring

Lunar Vehicle Benefit

Technology-only Benefit	1
Hardware or Operations Benefit	2

Mars Vehicle Weighting X Lunar Vehicle Benefit = Mars Benefit Score

Figure 2.5.4-1 Mars Missions Benefits and Weighting Criteria

2.5.5 Non-SEI mission capture

Evolutionary mission capture was one of the evaluation criteria for STV concept selection with a weighting of 15% of the total evaluation criteria. The purpose of this analysis was to determine how well the STV concepts designed for the lunar missions could capture other NASA and DoD missions identified as design reference missions (DRM). A general groundrule used for this analysis was that only "smart" stages based at the SSF or the ground could be used as the primary stage for these other missions.

The concepts were scored both by stage efficiency, that is, how efficient the lunar-sized stage can perform the other missions (required propellant mass and total start mass, excluding payload), and by Earth-to-orbit launched mass. These values were averaged over the mission model by the percentage of each mission included and then scored 1 to 5 (1 = best and 5 = worst) and weighted 80% mass fraction (i.e., stage efficiency) and 20% ETO mass. This weighting accentuates the stage efficiency in performing other missions. Because the NASA-only mission model and the combined NASA/DoD mission model differ as to the types of missions that were included, the analysis was done for each mission model and they were given equal weight in this analysis.

2.6 Lessons Learned

The evaluation criteria of cost, margins and risk, benefits to Mars, and non-SEI mission capture were weighted 50%, 30%, 5%, and 15%, respectively, according to the estimated importance of these criteria to the overall program. Sensitivity to this weighting split was also explored to determine the optimum vehicle design or designs. In this way, several general trends were noted from the architecture analysis. Ground-basing was favored as a basing option, with lunar-direct as the favored trajectory option. A single combined crew module was the favored crew module option, and the fewest number of stages was favored. The number of stages was the most influential trade, with trajectory options being the least influential.

2.6.1 Measures of Goodness

a. Performance – Although not a primary weighting criteria, vehicle performance contributes to vehicle costs, as was noted before. For single crew module concepts that go directly to the lunar surface, the lowest five-flight ETO mass concepts were the ground-based 1.5-stage vehicles. The worst cases were the combination space- and ground-based options, with 5% to 30% heavier mass than other options. These were poorer performers because both the stage aerobrake and crew module heat shield go all the way to the lunar surface. The space-based options were also poor because stage aerobrakes go to the lunar surface (no staging in LLO).

For single crew module concepts that use LLO for hardware storage, the lowest five-flight ETO mass were the space-based 2.5-stage vehicles These vehicles have a reusable LEV in lunar orbit and relatively lightweight transfer crew modules. The worst cases were again the combination space- and ground-based options, because of a heavier crew module (ballistic return) taken to the lunar surface. The ground-based options also have the heavier crew module, but benefit from not having aerobrakes.

For dual crew module concepts that use LLO for hardware storage, the lowest five-flight ETO mass was again the space-based 2.5-stage vehicles. Again the combination space- and ground-based options were the poorest performers, because of the heavier transfer crew module. Similarly, the ground-based options have the heavier crew module but benefit from not having aerobrakes. The dual crew module cases generally were 13% to 15% lighter than the corresponding single crew module cases. A significant conclusion that can be drawn within the dual crew module option data is that all ETO mass values are within 5% to 8% of each other. Thus, the dual crew module scenarios are not as performance sensitive to basing (and related configuration) impacts as are the single crew module scenarios.

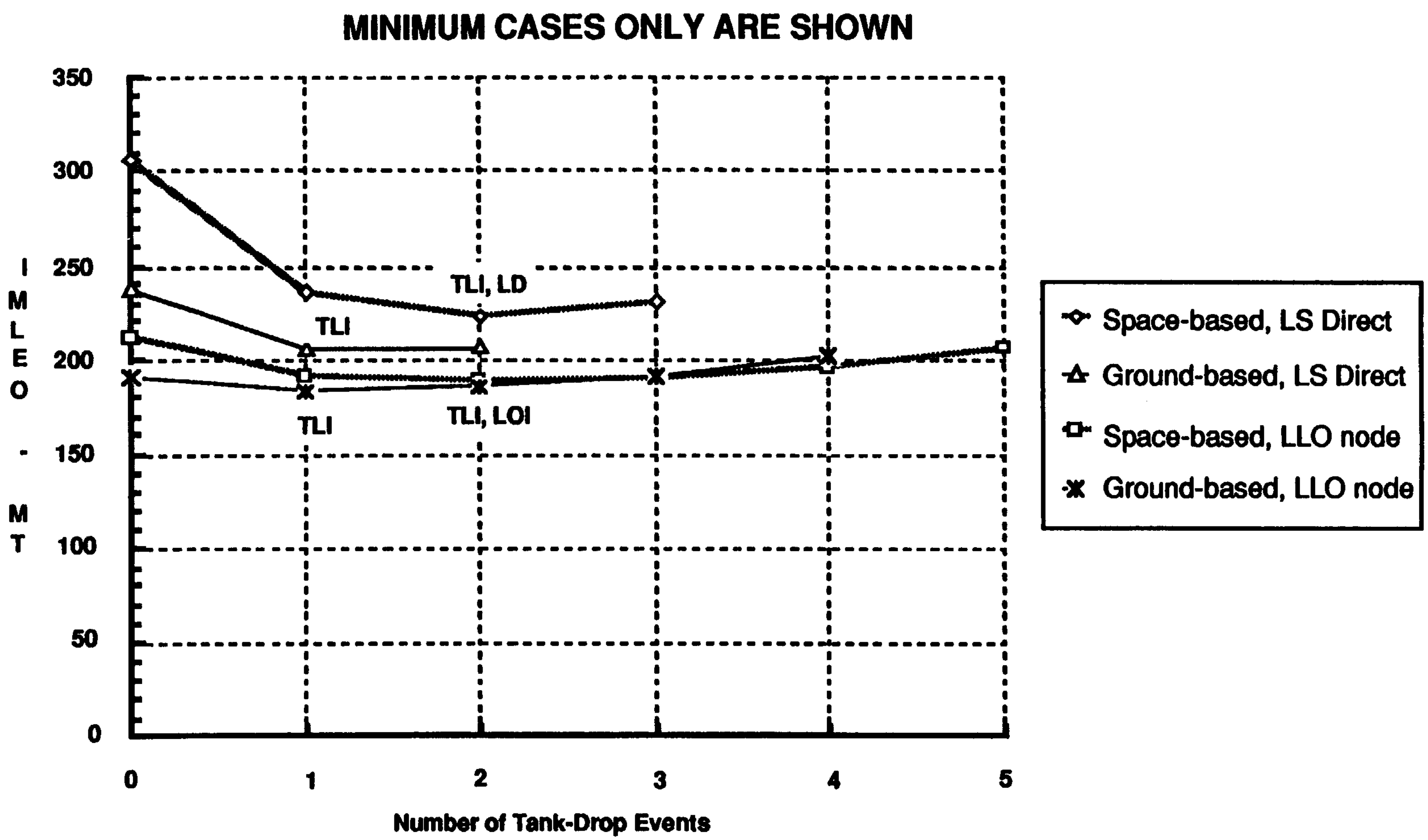
The same trends that apply to the dual crew module cases also apply to the hybrid crew module cases. The hybrid crew module ETO masses are 2% to 5% higher than the corresponding dual crew module masses but are 10% to 11% less than the corresponding single crew module cases. Again, the hybrid crew module scenarios are not as performance sensitive to basing (and related configuration) impacts as are the single crew module scenarios.

One of the architecture trade studies was the impact of an all-propulsive as opposed to an aeroassisted Earth-orbit insertion. For the two cases run, the all-propulsive option required 13% to 30% more ETO mass.

In the tank-drop analysis, for the 60 tank-drop cases run, the minimum cases are plotted on Figure 2.6.1-1 as total vehicle IMLEO versus number of tank-drop events. For space-based missions, the lowest mass occurs with tank-drop events following the first and second burns (TLI and LOI for options using LOR, and TLI and lunar descent for lunar direct options). The ground-based minimum occurs with only one tank-drop event following TLI for either the LOR or lunar direct scenarios.

Sensitivities to the minimum cases are shown in Figure 2.6.1-2, showing tank-drop cases within 5% of the minimum IMLEO, as well as the worst cases for each basing option. High penalties occur for no droptanks on direct-to-surface vehicles and for lunar ascent droptanks on LLO node vehicles.

Note that when the downselected ground-based scenarios were further defined and optimized, TLI and lunar descent droptanks were used. In the more detailed design process, landing legs were left on the lunar surface. This change in staging resulted in the optimum choice for the ground-based options being the use of TLI and lunar descent droptanks instead of just TLI droptanks.



Space-based missions have lowest mass with tank-drop events after the first (TLI) and second (LOI or LD) burns.

Ground-based missions have lowest mass with one tank-drop event after the first (TLI) burn.

Figure 2.6.1-1 Tank Drop Analysis Results

Tank-Drop Cases Within 5 Percent of Minimum IMLEO

Space-based, LS Direct		Ground-based, LS Direct		Space-based, LLO Node		Ground-based, LLO Node	
Tank-Drop Events	IMLEO	Tank-Drop Events	IMLEO	Tank-Drop Events	IMLEO	Tank-Drop Events	IMLEO
TLI, LD	222.3	TLI	205.7	TLI, LOI	188.7	TLI	183.9
TLI, LD, TEI	230.7	TLI,LD	206.6	TLI, TEI	188.9	TLI, LOI	186.0
TLI, TEI	236.2 (+6%)			TLI, LOI, TEI	190.9	LOI	187.5
				LOI	191.4	TLI, LD	188.7
				LOI, TEI	192.5	TLI, LOI, LD	191.6
				TLI, LD, TEI	193.9	NONE	192.1
				TLI, LOI, LD	194.3		
				TLI	196.1		
				TLI, LD	196.1		
				TLI, LOI, LD, TEI	196.4		
				LOI,LD	196.8		
				TEI	197.3		

Minimum Case

Current Trade Assumption

Worst Case Tank-Drop Events

No Drop-Tanks	305.7 (+38%)	No Drop-Tanks	238.2 (+16%)	LA	248.8 (+32%)	LA, LOI	231.3 (+26%)

IMLEO performance penalty for dropping tanksets after TEI vs. after LD or LOI is insignificant for the space-based vehicles.

Figure 2.6.1-2 Tank Drop Sensitivities

b. Cost – The STV design candidates architecture trade study yielded life cycle cost estimates of minimal difference. In some cases, only the system margins and risk scores made the lunar mission preliminary design concept candidates more decisive winners in the down-select process. See section 1.4 for a further explanation of the general engineering misconception by some that cost equals performance [changes] for space systems.

The cost analysis results indicated that the more STV flight elements (stages) developed for the system and then expended in each lunar mission, the least attractive they are in total life cycle cost (pretty intuitive, if you think about it.) Basing requirements are a driver in front end development costs. Booster launch costs are a high value item in the Operations and Support phase for all systems. Ground-based systems requiring much bigger boosters than the National Launch System derivatives at the time were penalized heavily in up-front system development costs (in billions of dollars), but still were lower than space-based systems of equal stage count.

The process used to develop the comparative systems cost data was quite successful for both STV study contractors, but the time allotments for LCC accumulator model development (spreadsheet model) and parametric cost model inputs setup was entirely too short. The cost analysis activity of 21 to 30 days should have been spread over 45 to 60 days to reduce the overtime stress. Creating the mission scenarios and design inputs took 60 days longer than was originally anticipated in the Boeing phase I study plan.

c. Risk and Margins – By comparing space based to ground based concepts, ground based has approximately 50% (16 out of 30) less steps performed before the start of the lunar mission. This can be looked at two ways. It implies that there is less risk in a ground-based system because there are less tasks to be performed. The other observation is that the decision to start the lunar mission for a ground-based vehicle is made before boost to LEO, where as for a space-based vehicle, it is made after. This is significant because the ETO acoustic and dynamic environment is predicted to be the worst the lunar vehicle will experience.

d. Applicability to Mars – All vehicle concepts provide benefits to the Mars mission, however, several design options more clearly benefit the Mars missions by advancing technologies and by providing operational experience. Overall, a space-based, multiple-stage vehicle that uses lunar orbit rendezvous and has a hybrid or dual crew module was shown to provide the most benefits to the Mars program.

Lunar orbit rendezvous is favored because it provides operational experience in remote, on-orbit rendezvous and fluid transfer techniques. Multiple stages are favored in general because of the benefit of on-orbit rendezvous and dock, as well as propellant management experience gained from on-orbit assembly of multiple flight elements.

The hybrid crew module is favored over single and dual crew modules bec ause of the applicability to Mars multiple crew modules. Ground-basing is favored for the use of a ballistic ground-return crew module, but space-basing is favored overall because of the long-term on-orbit storage experience gained. Space-basing is also favored because of aerobrake technology benefits, as well as on-orbit assembly and operations experience.

e. Non-SEI Mission Capture – All vehicle concepts captured all missions in the mission model, excluding the Mars mission, using flight elements sized for the lunar missions. In some cases, only the core vehicle was required; in others, the core vehicle and additional drop-tanks were needed to accomplish the mission. Overall, the all-propulsive, combination space/ground-based, 2.5-stage vehicle that uses lunar orbit rendezvous and has a hybrid crew module is the most efficient at capturing all non-SEI missions.

3. STV Phase 2 Summary

Phase 2 of the STV Concepts and Requirements study included lunar STV concepts designed for specific launch vehicle capabilities, with emphasis on ETO capability rather than single mission cargo capability. This part of the report summarizes the Phase 2 findings, specifically the relationships between program objectives, program requirements, and design characteristics.

3.1 Design Driver Assumptions

3.1.1 Available Funding Resources

No limits were placed on available funding for the STV design and operation, so life cycle cost was used as a discrimator in this phase of the study.

3.1.2 Space Program Infrastructure

a. NLS Launch vehicle characteristics – It was assumed for this study that the National Launch System (NLS) or similar program to upgrade the nations ETO transportation system would precede the lunar exploration program, and that elements of the NLS program could be adapted for the LTS. The candidate NLS vehicles and their most promising Heavy Lift Vehicle (HLV) derivatives for lunar exploration are shown in figure 3.1.2-1. These vehicles are characterized by a common oxygen-hydrogen core stage using four or six Space Transportation Main Engines (STMEs), and parallel burn booster elements utilizing either Advanced Solid Rocket Motors (ASRMs) or additional liquid rocket oxygen-hydrogen or oxygen-RP motors. Payload capabilities shown assume a kick stage or use of the upper stage with a suborbital burn to improve usable payload weights. Usable payload masses range from 70 tons to LEO for the solid-boosted NLS to 200+ tons with new twin liquid booster concepts. For the STV study it is assumed that if the solid-boosted NLS derivative is all the nation can afford, then each lunar mission would require two launches, and if the twin liquid-boosted HLV is available, then each lunar mission can be flown in a single launch.

b. No SSF accommodations for STV – Due to a scaling-back of the Space Station Freedom design, this portion of the study assumed that no SSF accommodations would be initially available for the STV. This drives the design to

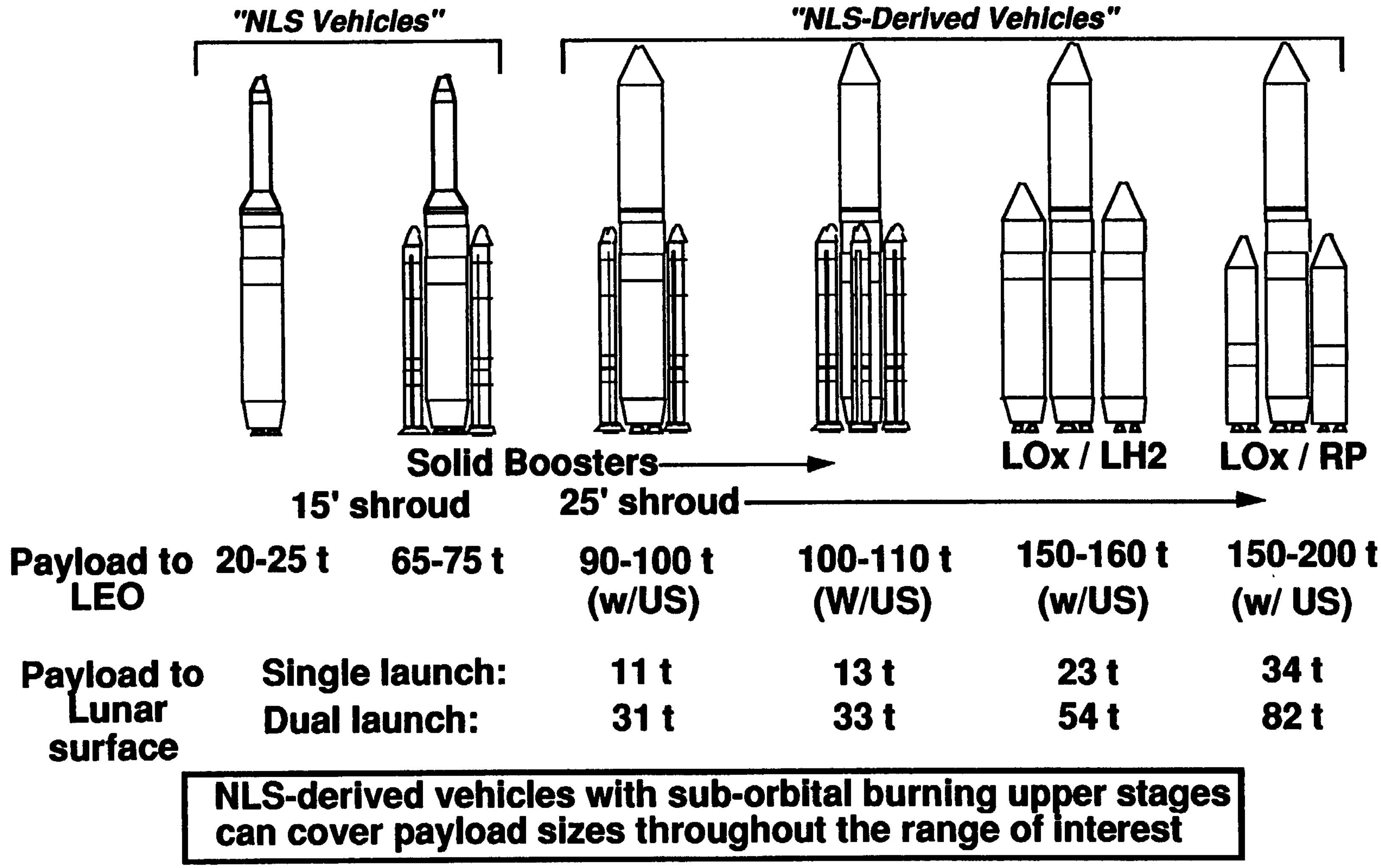


Figure 3.1.2-1 NLS Launch Vehicle Family

using either a single launch or multiple launches with Earth-orbit-rendezvous (EOR) to assemble the Lunar Transportation Vehicle. This also limits the reusability of the vehicle, as no on-orbit storage or refurbishment facilities are assumed available.

3.1.3 Available Technology

Late 1990's technology was assumed for the vehicle designs for this phase of study, with additional technology development added as a cost discriminator.

3.1.4 Lunar / Mars Exploration Requirements

During phase 2 of the STV study, the primary mission requirements were for lunar exploration and supporting a lunar outpost.

a. Baseline scenario description – The basic mission requirement was to place four crew and adequate cargo on the lunar surface to conduct exploration missions of the type outlined in the Stafford Synthesis Report. A description of typical lunar surface payloads is given in Figure 3.1.4-1.

b. Alternate scenarios – Other lunar exploration scenarios studied included a Rovers-first scenario, in which unmanned teleoperated rovers are sent to 'prospect' for points of interest on the lunar surface, after which a manned roving habitat is sent for further prospecting. Also studied was a 'lunar campsite' scenario in which a crew is sent for a period of 45 days to explore and conduct experiments, leading to a full lunar outpost. Further explanation of these scenarios is given in Section 3.3.4.

3.1.5 Non-SEI Mission Requirements

A description of non-SEI mission requirements was given in Section 1, the Design Guide. During phase 2 of the STV study, these requirements were not levied on the design

Major Elements	Transit Mass, t	Stowed Volume, m3	Transit Power, Watts	Transit Thermal Rejection, Watts
Payload Unloader*	10.0	240	3000	0
Attachments for Payload Unloader	6.3	32	0	0
Initial Habitat Module	12.0	150	2000	0
Airlock	3.0	47	300	0
Power Module	7.5	100	0	0
Lab Module	12.0	150	2000	0
LEV Servicer	2.3	12	0	0
Enhanced Habitat ECLSS	3.9	50	500 (Est.)	0
Logistics Module	3.0	120	3000	0
Submillimeter (IR) Interf Elts**	2.0	100	500	0

* Selected as DRM L1 payload
** Selected as DRM L2 and L3 payload

Figure 3.1.4-1 PSS Cargo Description

3.2 Reference concept description

3.2.1 Groundrules and Assumptions

Our groundrules and constraints were: 1) operate ground-based using one or two launches of the various NLS derived vehicle concepts (two launches require EOR), 2) utilize as many NLS and NLS Upper Stage (NLSUS) components and facilities as practicable, 3) minimize the number of individual elements to be developed and produced during the scenarios (ie. minimize the number of stages and manned modules), 4) utilize direct lunar descent and ascent to maximize missions flexibility and safety (no Lunar Orbit Rendezvous (LOR)), and 5) reenter the manned module directly to the launch site to minimize operations cost and complexity.

3.2.2 System Design and Operation

The Reference LTS Concept is ground-based with either direct launch of a complete LTS using a single large NLS derivative vehicle, or two launches of the reference NLS vehicle with Earth Orbit Rendezvous (EOR) and docking as the method for assembling the LTS. The major components of the reference LTS are shown in figure 3.2.2-1 and as an integrated single launch mission stack on top of the launch vehicle in figure 3.2.2-2. As is shown, use of common elements for either a personnel transport mission (with some cargo capability) or cargo mission allows maximum mission configuration flexibility while minimizing subsystem change. This concept can be flown with either a cryogenic or storable propellant ascent stage with the launch vehicle capability shown and is sized such that the storable propellant stage option can deliver two metric tons to the lunar surface during a piloted mission and return 100 kg in addition to the crew. With smaller launch vehicle capability, only the cryo ascent stage option is viable.

With a cryogenic ascent stage, analysis has shown that it should be possible to store liquid hydrogen in carefully insulated tanks on the lunar surface for periods exceeding six months with acceptable boil off losses. This option is preferred because it: 1) requires no additional engine development or production, using the same engines for descent and ascent, and 2) provides an additional seven to eight tons of payload on each piloted flight, thereby eliminating approximately three lunar cargo flights at $1.3B

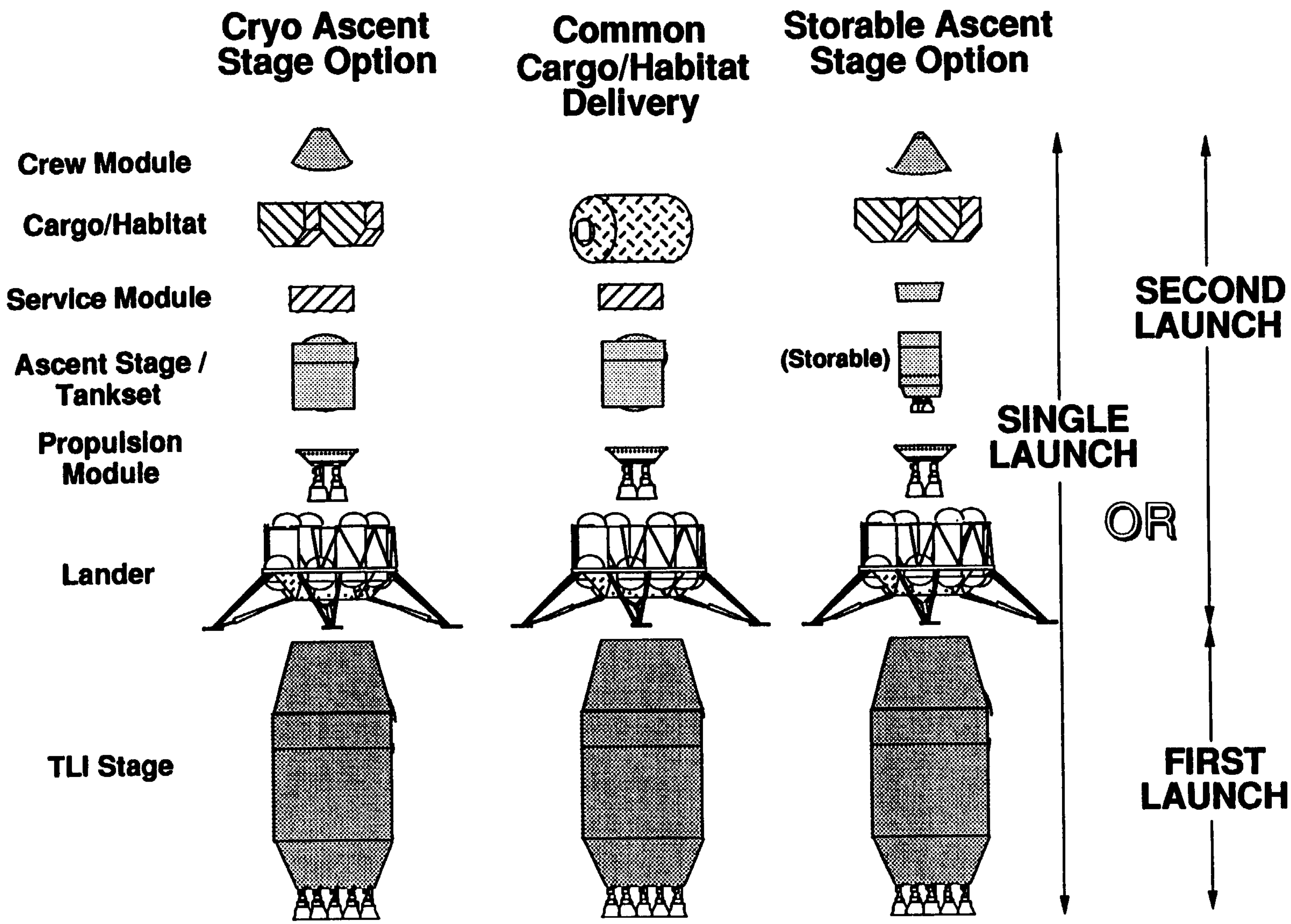
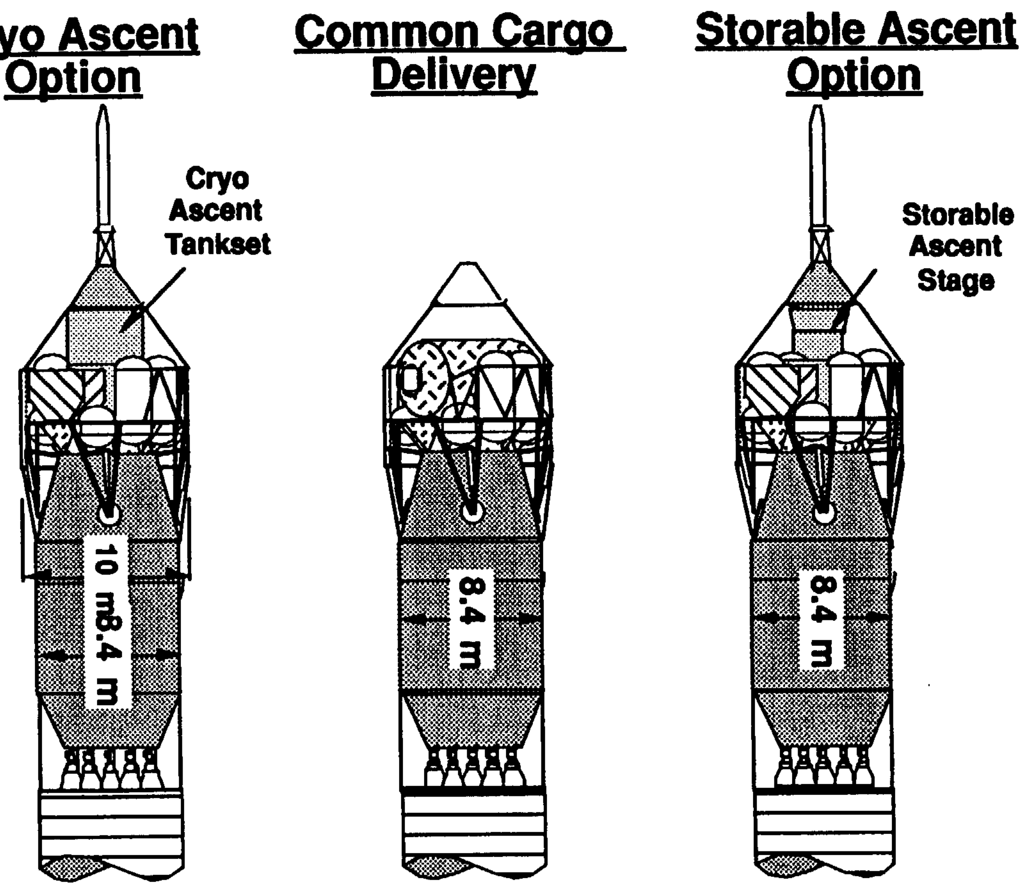


Figure 3.2.2-1 Lunar/Earth Transfer Vehicle Components Reference LTS Concept

Staged Mass: 280 t

IMLEO: 211 t

Note:
- Suborbital staging from launch vehicle
- 6.7 t combined crew modules (with additional 2.0 t crew/consumables)
- Cryo ascent option performance includes payload penalty (-450 kg) for 45-day lunar surface boiloff .



Lunar Payload:

	Cryo Ascent Option	Common Cargo Delivery	Storable Ascent Option
RL10 A-4 engines:	9.4 t	34.1 t	2.0 t
RL10 B-2 engines:	13.5 t	37.7 t	5.2t

Figure 3.2.2-2 Launch Configurations- Reference LTS Concept

each. We estimate that the the cryo ascent version has just over $10B in Life Cycle Cost (LCC) savings relative to the storable ascent version.

The mission profile for the reference concept includes Earth-to-orbit, Lunar transfer, lunar operations, and Earth return phases. For a dual-launch option requiring EOR, the TLI stage performs a suborbital LEO insertion burn and then waits at 160 nmi for the second launch. At this altitude, air drag does not pose a threat to the massive (150 t) TLI stage. The lander, after launch, may or may not do a suborbital burn (depending on the mass to orbit capability of the launch vehicle). If a sub-orbital burn is required, the lander tank size would increase over what is shown. After rendezvous, docking, and checkout, the TLI stage pushes the combined stack to trans-lunar injection. After separation, the TLI stage is discarded by flying on a predetermined impact trajectory to the surface of the moon where accounting and tracking of it is simplified.

A single launch piloted mission profile is depicted in figure 3.2.2-3. The system operation is similar to the dual-launch case, except that the entire stack is integrated on the launch vehicle, and no EOR is required. In both cases, the LTS is integrated in the launch vehicle such that the crew module sits on top of the shroud making it accessible to the ground crew prior to launch,and easily removed by the launch escape system (LES) in case of an abort. For a cargo only mission, a "beanie" style cap structure is attached to one shroud segment to cover the manned module opening.

During the lunar operations phase, as shown in figure 3.2.2-4, the lunar landing segment consumes the propellant in the lander tanks during lunar capture, descent, and landing. Once lunar surface operations are completed, in preparation for the return to Earth, any remaining propellant in the lander tanks may be transferred to the ascent stage prior to departure. When launching the crew capsule from the Lunar surface, the ascent stage uses the lander as a launch platform, leaving the empty drop tanks, landing gear, and associated structure behind on the lunar surface. The core stage propellants are consumed during ascent and direct earth injection. The empty core stage is then jettisoned prior to Earth atmospheric entry and left to reenter on a safe disposal trajectory to the earth's surface. The crew module continues on a reentry trajectory and lands at KSC.

For a cargo delivery mission using a single launch, the trajectory may be simplified by employing a lunar-direct trajectory, as shown in figure 3.2.2-5. This is possible since

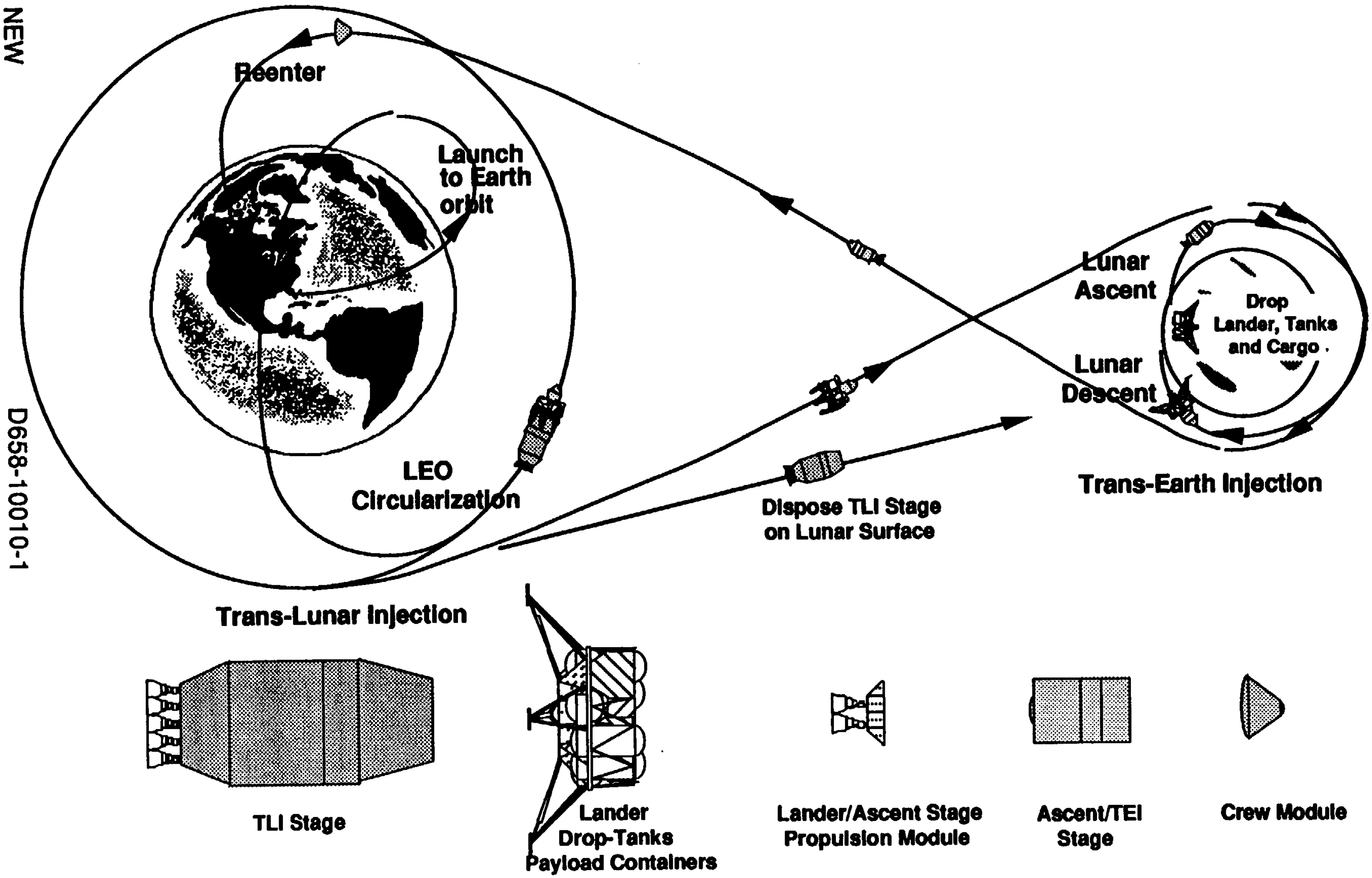


Figure 3.2.2-3 Single Launch Piloted Mission Profile Reference LTS Concept

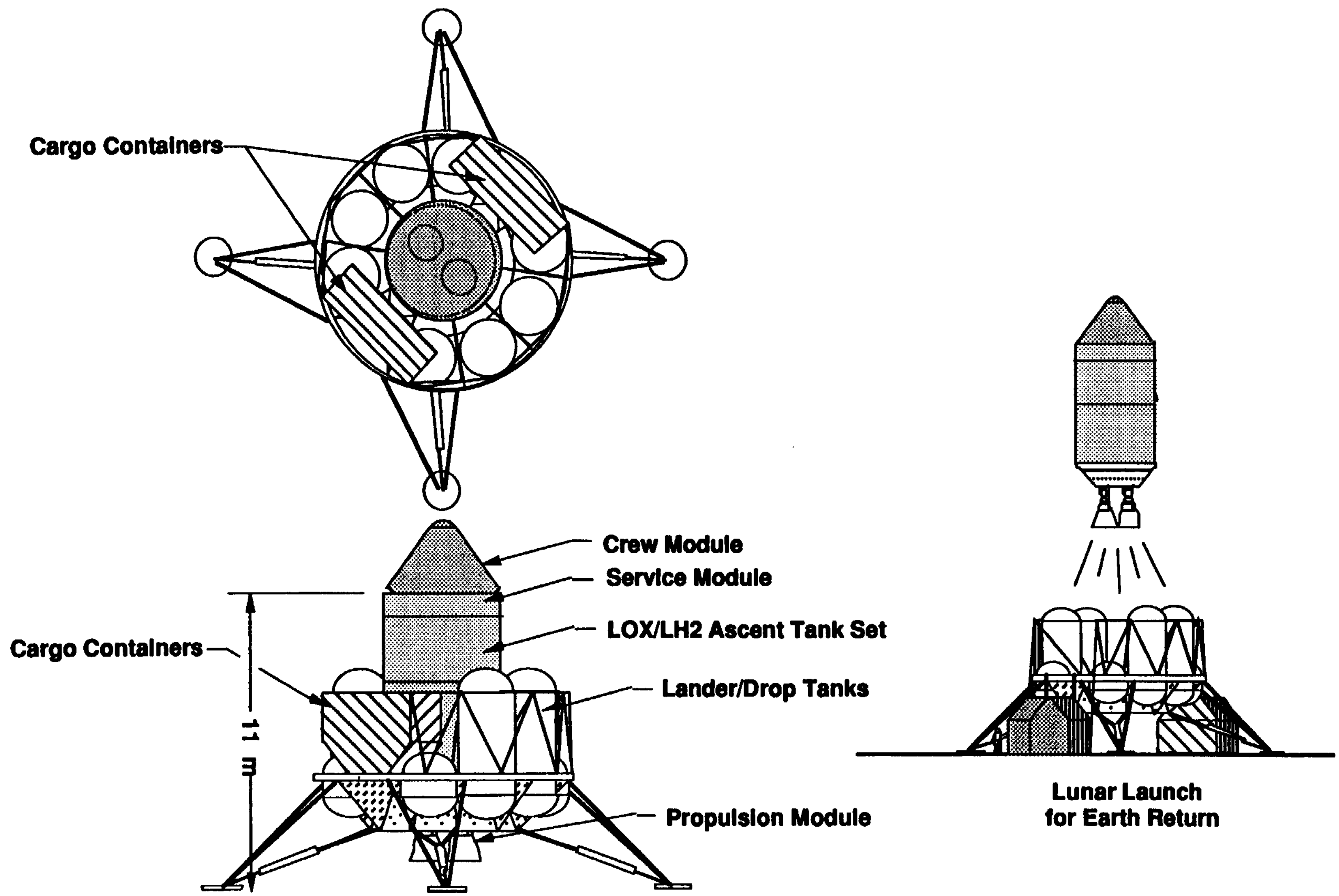


Figure 3.2.2-4 Piloted Lunar Lander Configuration Reference LTS Concept

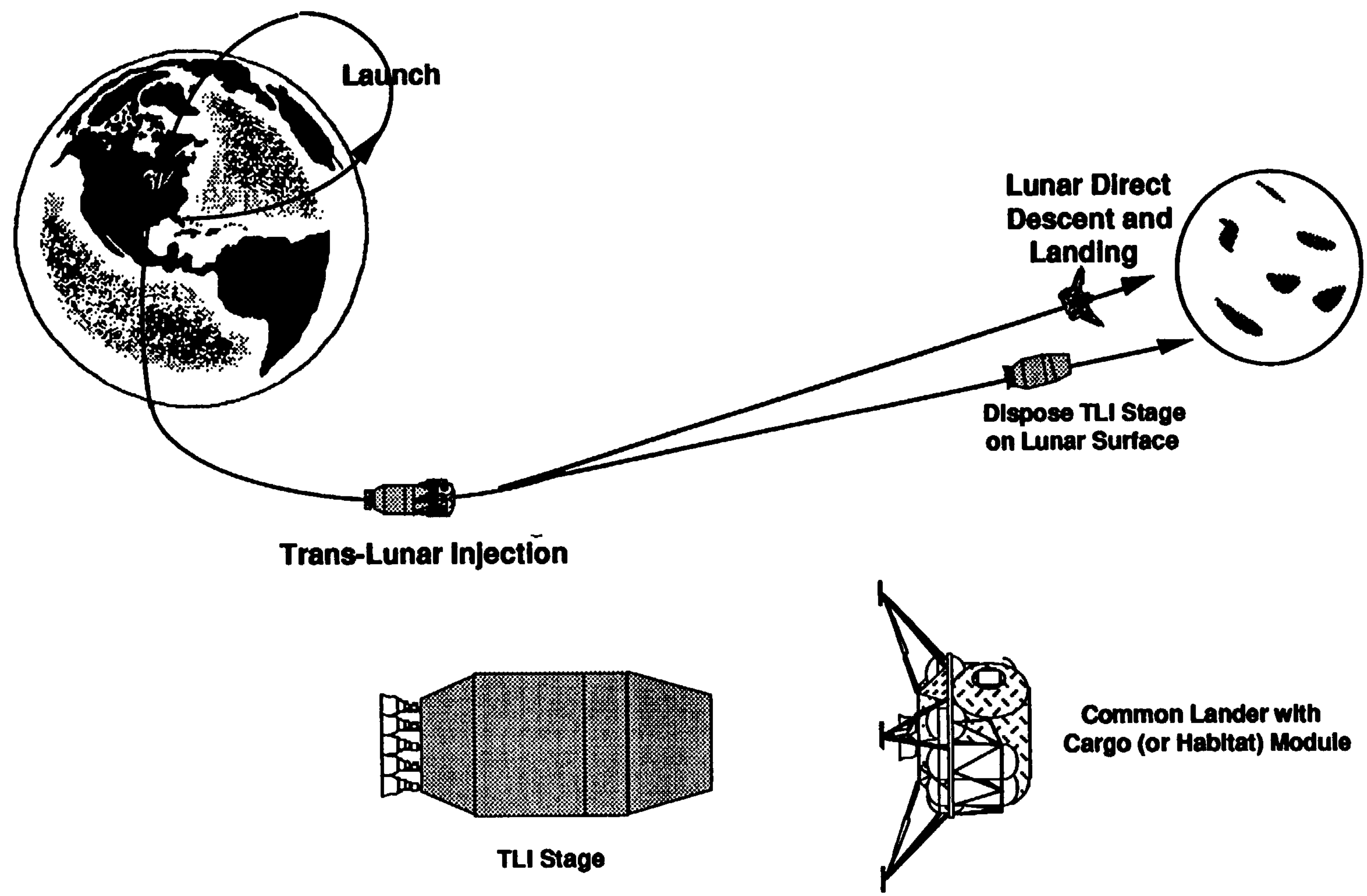


Figure 3.2.2-5 Single Launch Cargo Mission Profile Reference LTS Concept

retrieval of an aborted vehicle or cargo from a low Earth or low Lunar orbit may not be feasible anyway. As shown in figure 3.2.2-6, the cargo mission uses basically the same lander, propulsion module, and service module as the piloted mission, with additional avionics in the service module to take the place of the crew module. The cargo module could be either deployable or a habitat for a Lunar Campsite mission. The lander tanks are emptied during descent and landing, and the lander remains on the lunar surface for possible future use.

A summary weight statement for the Reference LTS Concept is presented in Figure 3.2.2-7.

3.2.3 Flight Element Description

a. TLI Stage – The TLI stage may be an adaptation of the proposed NLS upper stage (NLSUS) in that it would use the same engines and avionics, but would have much larger tankage and five engines. The engines used for performance and cost estimates of this concept are RL10A-4Bs and are also used on the lander element (see Main Propulsion section for engine description). For the dual launch case, the TLI stage would be non-load bearing structure and be mounted inside a shroud during launch. This allows it to be covered with Multi-Layer Insulation (MLI) to minimize boiloff over the 30-day LEO stay. For the single launch case using the advanced 150 to 200 tonne NLS derivative, the TLI stage is load-bearing structure mounted below the shroud and would be covered with foam and thermal paint (same as Saturn S4-B).

b. Common Crew Module – The reference LTS features a single reusable crew module, shown in figure 3.2.3-1. This single crew module carries a crew of four and serves as a transit habitat during the coast to and from the moon, as well as the excursion module during the lunar landing and ascent (Apollo had a separate Lunar Excursion Module or LEM), and as the Earth reentry and landing vehicle. Use of a single manned module for the complete round trip eliminates the development of a separate expendable LEM with its many duplicate subsystems, saving an estimated $5B in LCC.

The module has a basic Apollo Command Module shape but is fourteen and a half feet in diameter, allowing it to be transported in the Shuttle for possible use in other mission scenarios while providing additional interior space required for a crew of four

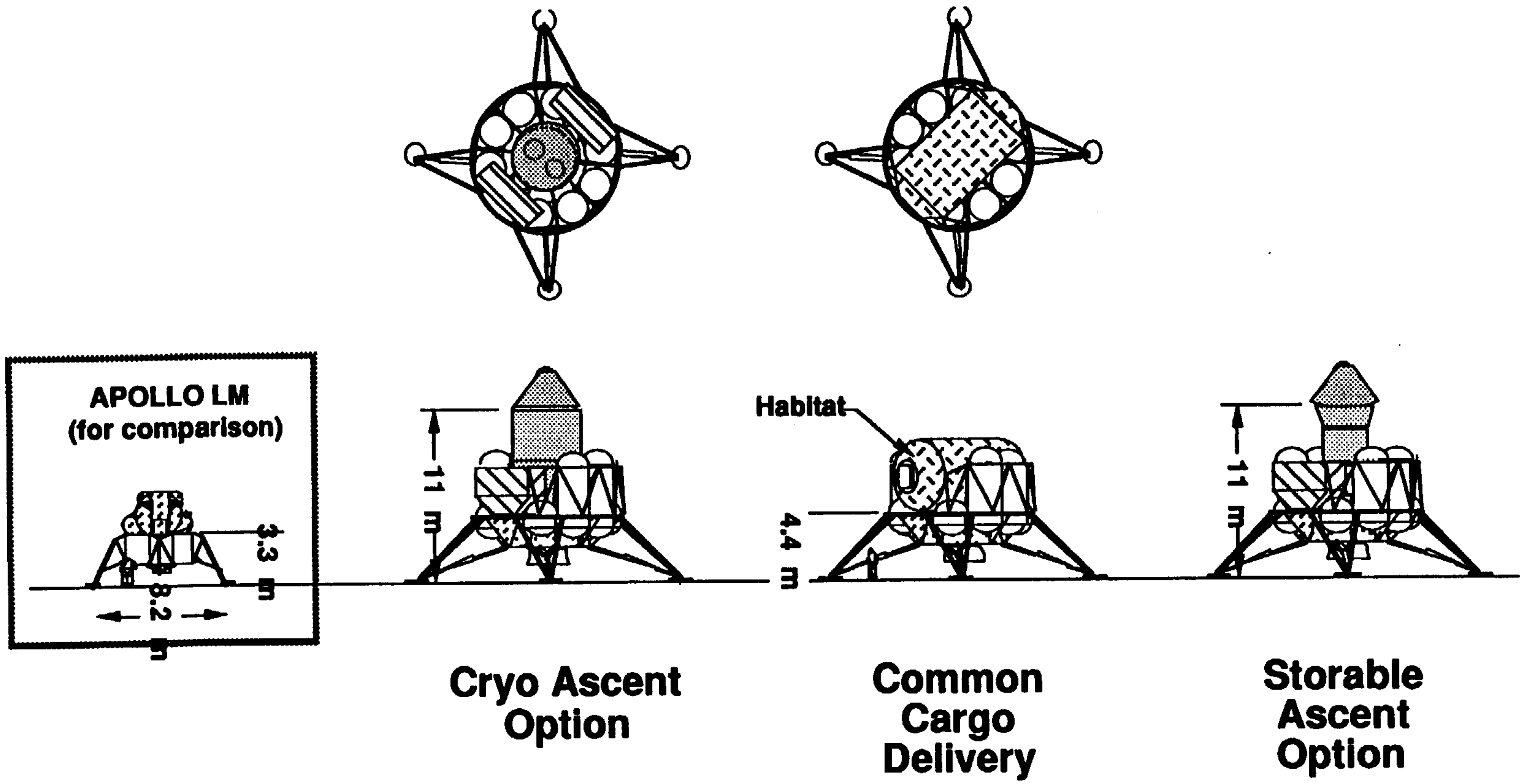


Figure 3.2.2-6 Lunar Landing Configurations Reference LTS Concept

	Reference Concept		
	Piloted - Cryo Asc	Cargo	Piloted - Stor Asc.
Design Data			
Delivery to LEO:	Suborbital	Suborbital	Suborbital
Lunar Surface Staytime (days):	2	n/a	2
TLI Stage Engines:	(5) RL10-A4	(5) RL10-A4	(5) RL10-A4
Descent/Core Stage Engines:	(3) RL10-A4	(3) RL10-A4	(3) RL10-A4
Ascent Stage Engines:	same	n/a	(3) XLR-132
Shroud Outside Dimensions (m):	10 x 12	10 x 14	10 x 12
Enclosed Dynamic Envelope (m):	9.2 x 12	9.2 x 12	9.2 x 12
Mass Summary (kg)			
Staged Mass	280238	279967	280239
(LTS Vehicle at LEO)	(209100)	(208833)	(209101)
Cargo	9930	34130	2000
Crew Mod, Crew, Consumables	8675	0	8675
Service Module Inert	1940	2123	1940
Ascent Stage/ Tankset Inert	1380	0	2890
Ascent Stage Usable Prop	13284	0	19705
Lander Prop Module Inert	1490	1490	1490
Lander Platform Inert	0	0	0
Lander Stage/Tankset Inert	3540	3540	3540
Lander Usable Prop	32261	30950	32261
Stage Adapter	1850	1850	1850
TLI Stage - Inert	11770	11770	11770
TLI Stage Usable Prop	122980	122980	122980
Usable Prop to LEO	66980	66976	66980
Ascent Reserves (on TLI Stg)	4158	4158	4158
Fairing / LES			
Launch Escape System	3493	0	3493
Shroud - Jettisonable	9030	9030	9030

Figure 3.2.2-7 Reference Concept Mass Summary

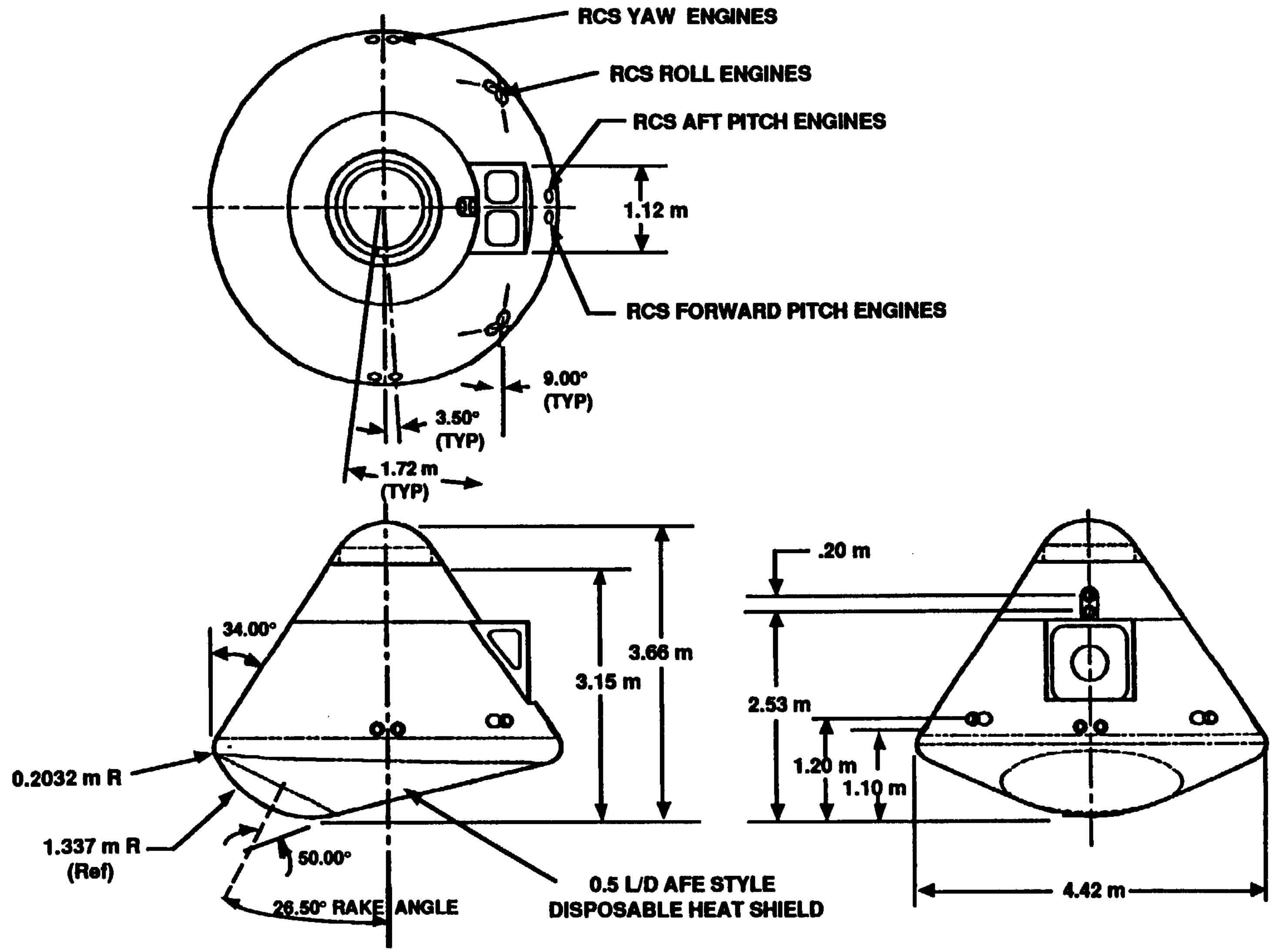


Figure 3.2.3-1 0.5 L/D Direct Entry Earth Landing Crew Module Reference LTS Concept

and longer duration habitation missions. It features a raked elliptic cone heat shield, modified for 0.5 L/D in order to give the capsule adequate cross range for a direct moon-to-Earth reentry with landing at KSC.

Selection of four instead of three crew was based on the desire for two-shift operation to increase the work output during a surface visit and maximize return on investment. Even larger crew sizes may be desirable to support extensive extra-vehicular activity (EVA) and they can be accommodated for small crew module weight penalties (an estimated 2.2 tonne increase for two extra crew). However, based on SSF design work, it appears that teleoperation of on-board systems would substitute adequately for an extra pair of eyes and ears required to support a two-person EVA, with off-duty crew serving as back-up. Thus the third crew person would remain on earth, making a four-person lunar crew viable for two shift operation.

The preferred Earth landing concept, shown in figure 3.2.3-2, features a steerable high glide parachute, such as the parafoils being developed by the Advanced Recovery Program at MSFC, in combination with an air bag impact attenuation system, similar to the system developed for the ALS P/A module land recovery system. The high glide chute system allows very accurate short-runway or prepared-site landings. The heat shield is a disposable, low cost, light weight silica fiber matrix thermal protection system that is jettisoned by the deployment of the air bags prior to touch-down.

If the high-glide chute system is deemed too costly or too high of a risk to develop or use, then a set of ballistic parachutes, such as those used on the Apollo Command Module could also be used, in conjunction with an omni-directional crushable impact-attenuation system for improved stability during touch-down. This concept, shown in figure 3.2.3-3, is capable of either land or water landings and allows the module to land on minimally prepared landing sites. In order to minimize the effect of wind drift while descending on the chutes, a terminal decelerator system, such as a retrorocket would be required. In addition, up-linking landing site wind information to the crew module, prior to reentry, would allow the module to bias its reentry trajectory to counteract expected wind drift, further reducing landing zone size requirements.

c. Equipment Module – The LTS concept has an equipment module which contains different subsystem modules for different missions. For the cargo only mission, the equipment module is attached to the lander propulsion module and has a

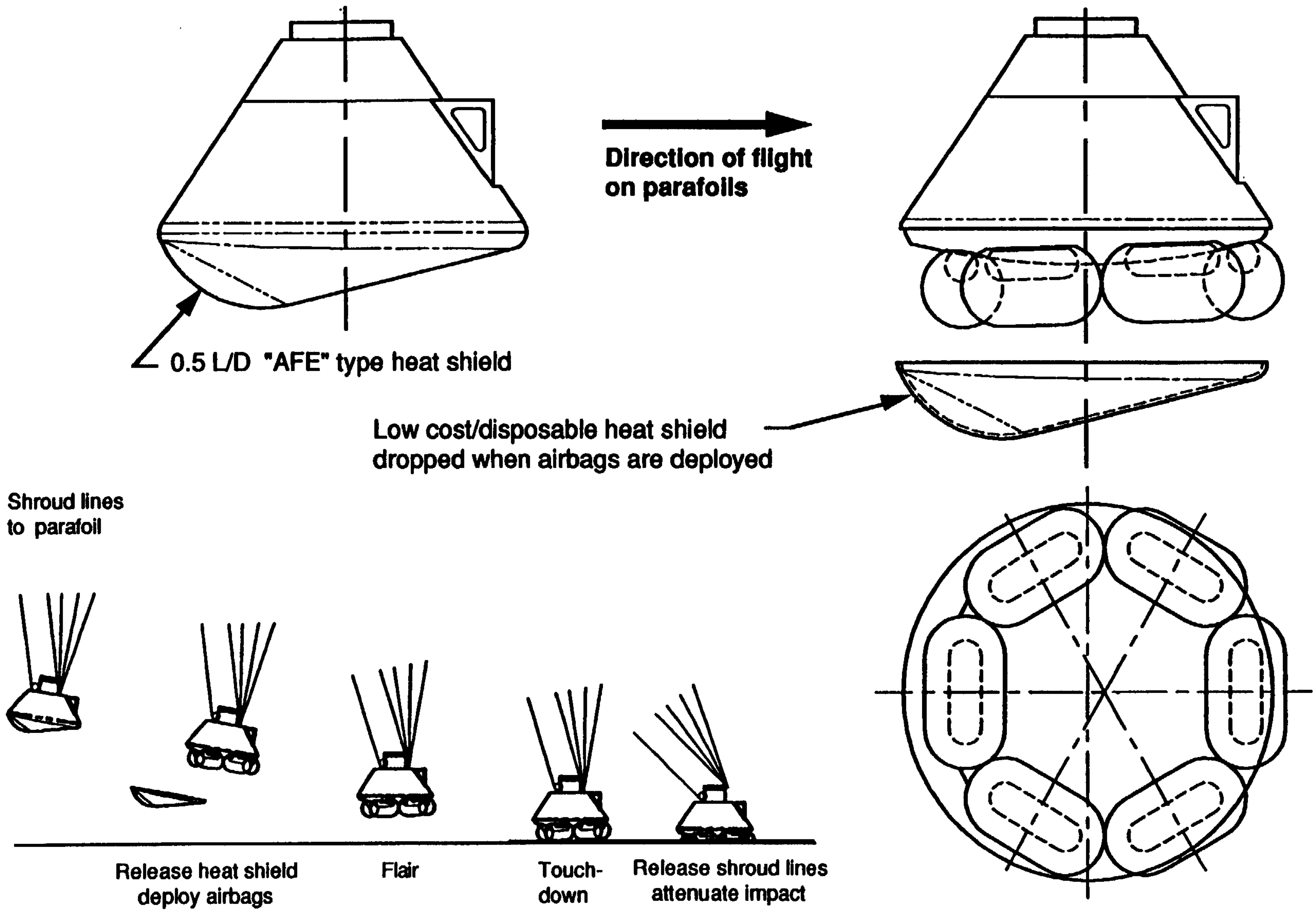


Figure 3.2.3-2 High Glide Parachute Earth Landing System Reference LTS Crew Module Concept

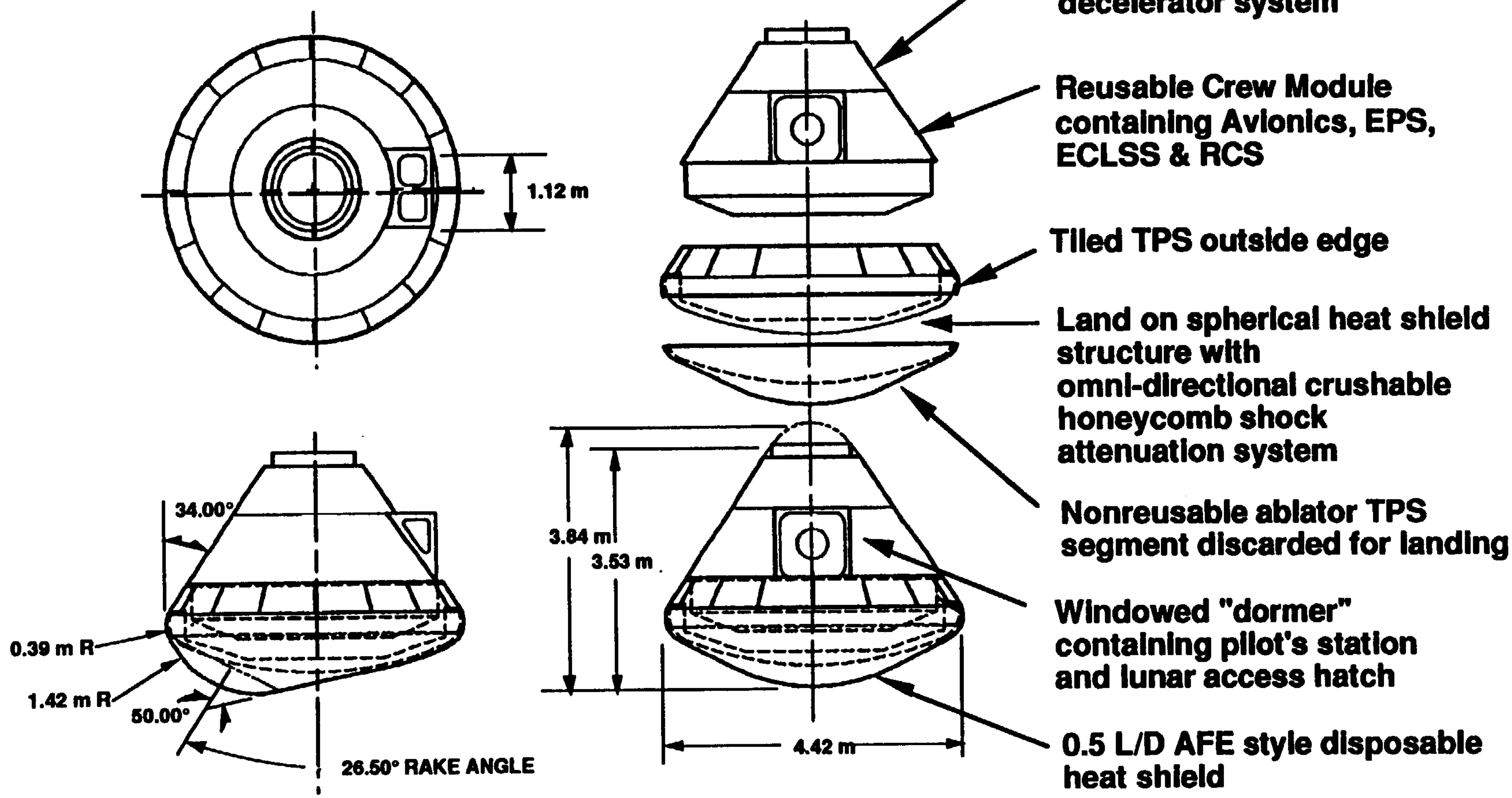


Figure 3.2.3-3 Ballistic Parachute Earth Landing System Reference LTS Crew Module Concept

framework to carry and unload the cargo pallets, plus a full set of avionics and power boxes with a passive Thermal Control System (TCS). For the piloted missions, the equipment module is attached to the ascent stage tankset and has a simple adapter to support the crew module. It has a larger power supply and an active TCS with radiators and water boilers to handle the increased demands of the Environmental Control and Life Support System (ECLSS), but the Inertial Measurement Unit (IMU), the navigation unit, and the Computer Processor Units(CPU) boxes are omitted because they are in the crew module.

The avionics is assumed to be a repackaged version of the NLSUS avionics with additional on-board sparing to ensure six month mission life. In-house simulations of lunar return trajectories have shown that the currently proposed common lunar module can leave the surface of the moon at any time and accurately land at Kennedy Space Center (KSC) using an updated version of the NLS guidance package.

d. Ascent Stage – The ascent stage would ideally be a derivative of the NLS upper stage (NLSUS), and consists of a propulsion module, tankset, and equipment module adapter. For early missions of around 45 days lunar stay time, the tankage is similar in design to that on the proposed NLSUS, but has thermally-optimized support struts and 80 layers of Multi layer Insulation (MLI). For later missions requiring longer lunar stay-time, liquid hydrogen tank Vapor-Cooled Shields (VCS) and an on-board LO2 refrigerator would be added to reduce propellant boil off to about 360 kg during a six month stay.

The variation in payload mass with design lunar surface stay time for various boiloff reduction schemes is shown in figure 3.2.3-4. Relative payload performance for the storable ascent stage version, which suffers no boil off losses is given on the same figure. Note that a cryogenically fueled ascent stage designed for six months on the moon using current thermal control technology (24 layers of MLI) would have no performance advantage over a storable stage, but would still have considerable economic advantage since it requires no new ascent engine package and uses current technology.

e. Lander – The lander module is a common element to both cryogenic and storable-propellant ascent stage concepts and consists of four liquid hydrogen and

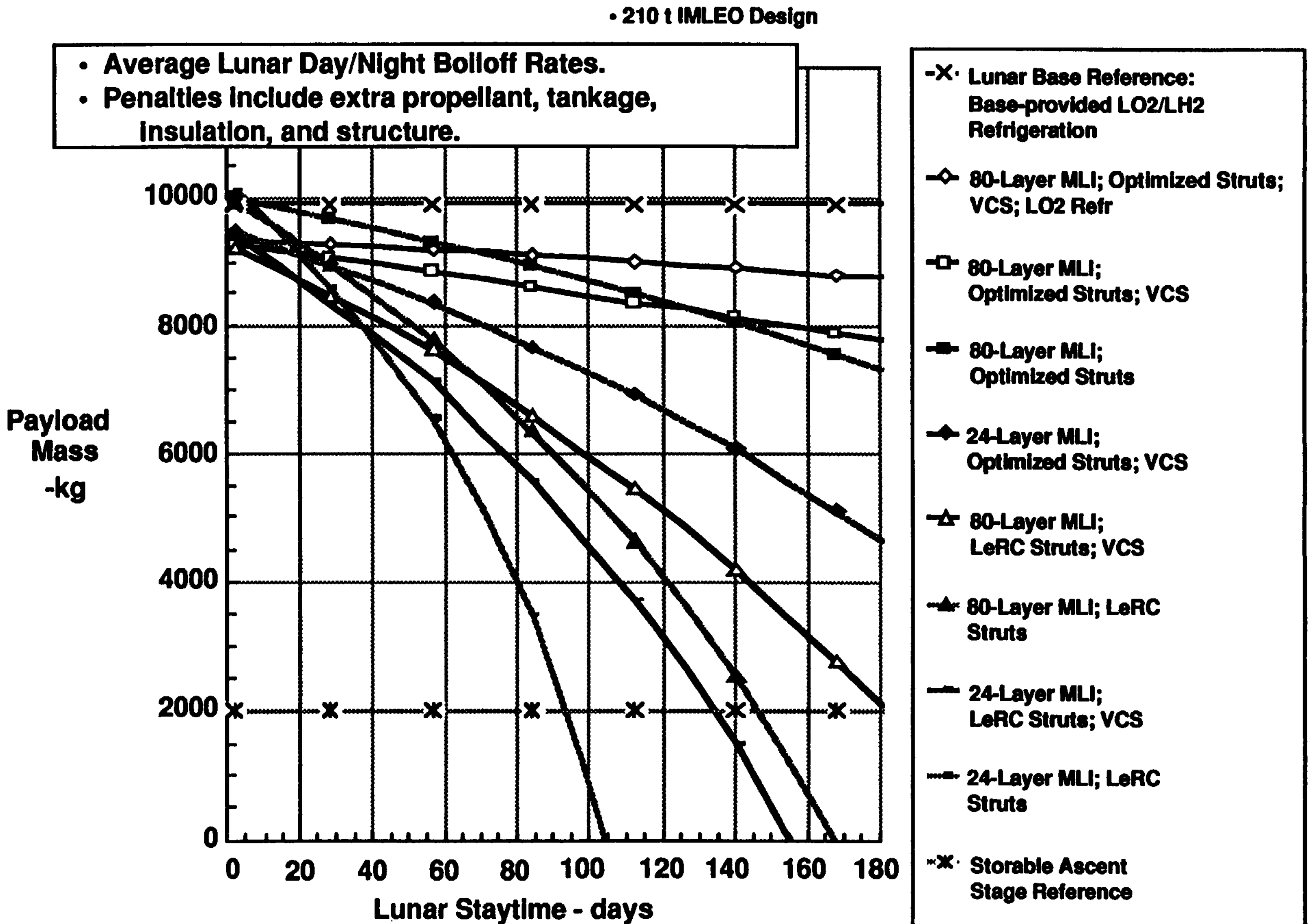


Figure 3.2.3-4 Lunar Boiloff: Piloted Payload Degradation

four liquid oxygen tanks, tri-pod landing gear and associated support structure. The lander can be configured to carry either an ascent stage with cargo modules, or a cargo/habitat module. For the piloted ascent option, the lander also serves as the launch platform for the Earth return launch. For a cargo delivery mission, the cargo is palletized and launched mounted transverse on the lander. This lowers the lander center of gravity (cg) to avoid tip-over, and simplifies cargo unloading. The landing capability is based on Apollo LEM technology and designed to handle one meter/sec horizontal velocity and a 15 degree adverse slope at touchdown, without tip-over. The gear lower center strut includes a shock absorber for attenuating residual touchdown velocity. The gear is stowed folded for launch and is deployed using springs.

The landing gear lower struts are attached to the Lander lower structural interface ring. This ring provides the interface between the TLI stage launch adapter and the Lunar Lander stack. A system of struts transfers loads between this lower ring and a higher, inboard structural ring which provides the detachable structural interface between the Lander module and the Propulsion Module. The propulsion module is pyrotechnically disconnected from the top of the lander structure at lunar ascent launch. An outboard structural ring, at the same level as the inboard ring, is connected to the lower inboard rings with gussets. These gussets provide the load paths for the landing gear and are the structural support interface for the Lander module tanks.

3.2.4 Subsystem Description

a. Lander Main Propulsion – The lander main propulsion system (MPS) provides propulsion for all post TLI burns, including course change after TLI stage separation, lunar capture, descent and landing, as well as lunar ascent, trans-Earth injection, and ascent stage disposal for piloted missions. The lander main propulsion systems for the Piloted and Cargo mission configurations are shown schematically in figures 3.2.4-1 and -2 respectively.

The MPS is designed to 1) optimize weight and cost versus mission reliability by providing optimum staging and quantity of engines, 2) minimize configuration change between Piloted and Cargo missions, 3) provide a low center of gravity to simplify lunar landing, and 4) minimize propellant loss due to boil-off during coast and lunar stay mission segments.

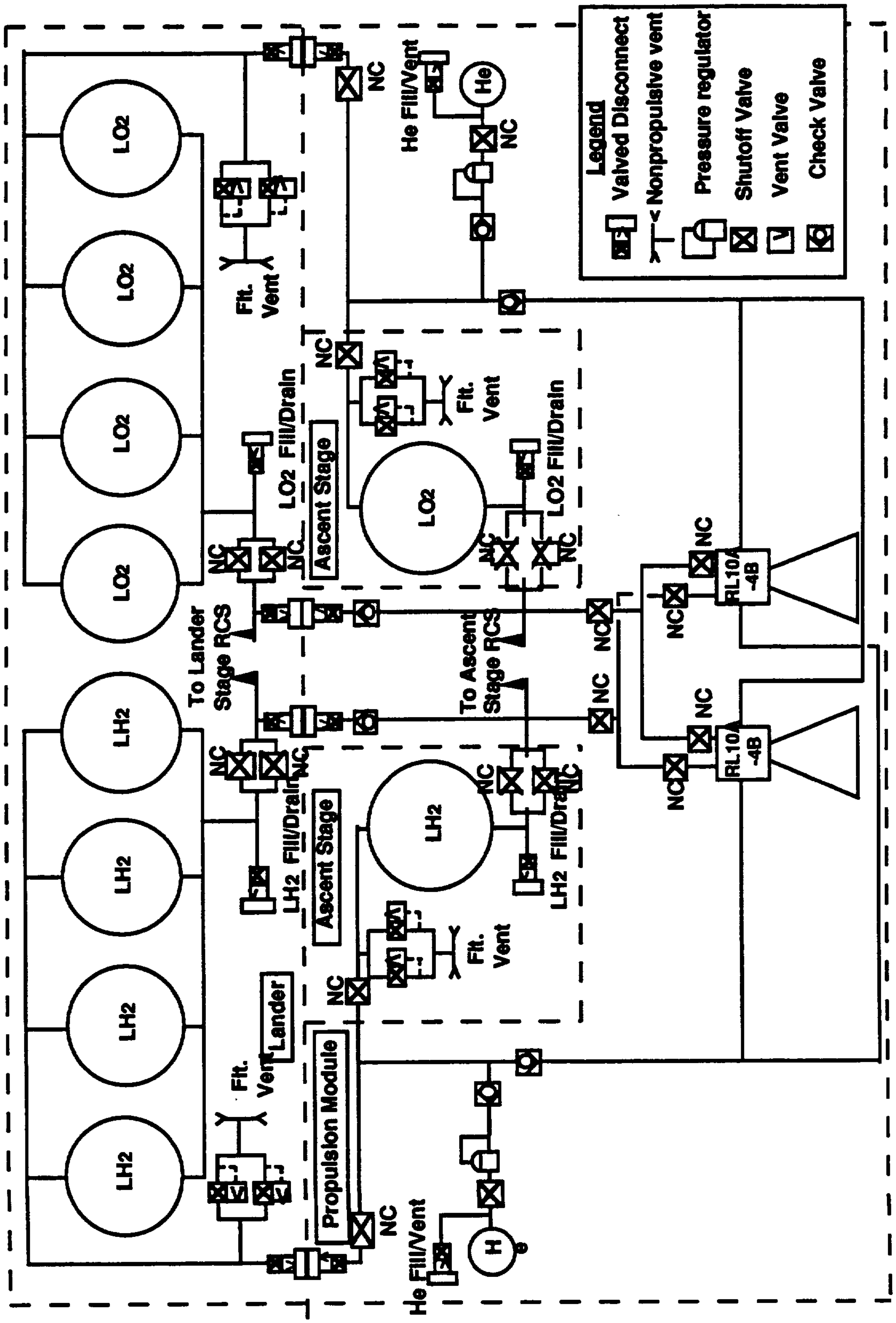


Figure 3.2.4-1 Piloted Lunar Lander MPS Schematic Reference LTS Concept

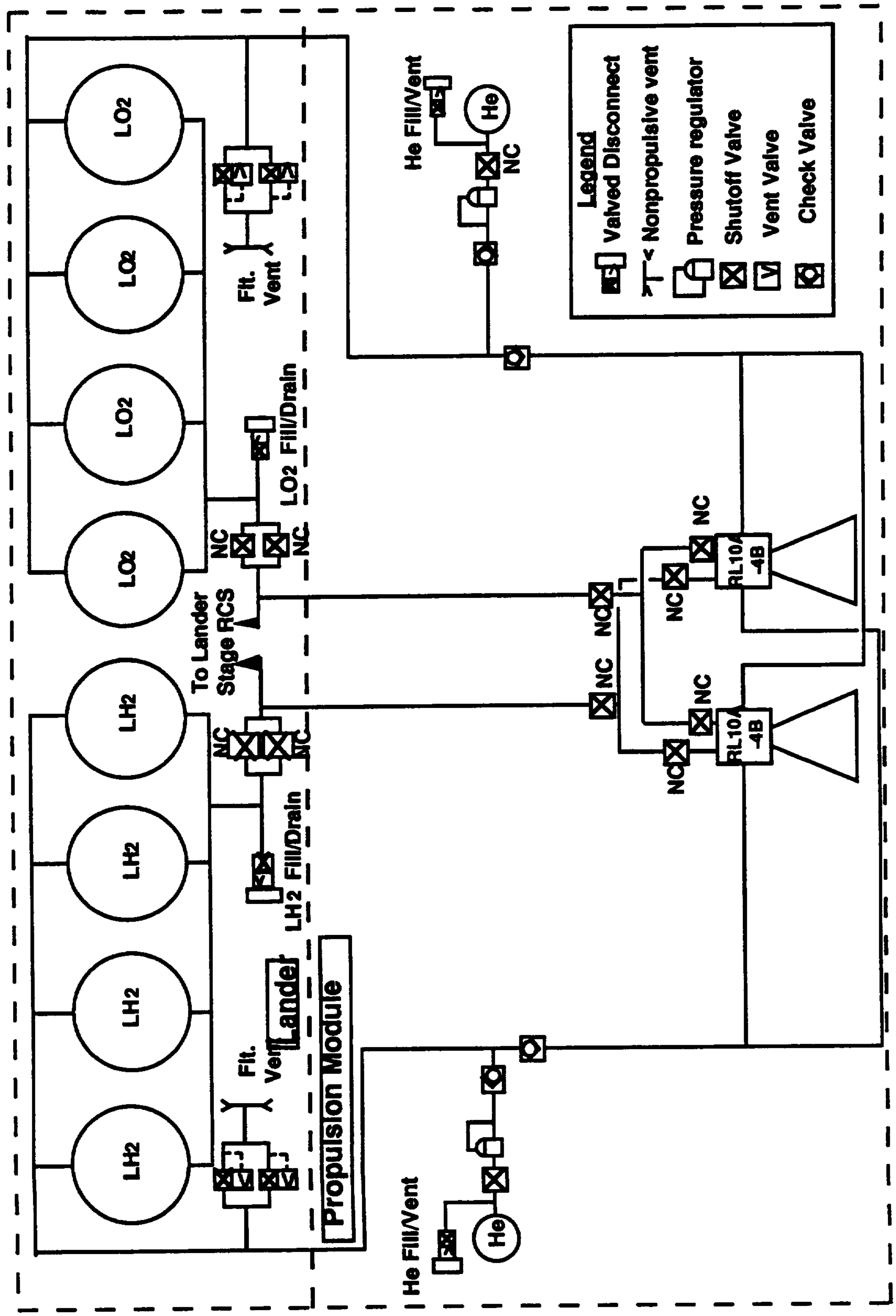


Figure 3.2.4-2 Cargo Lunar Lander MPS Schematic Reference LTS Concept

Optimum staging and configuration change minimization is accomplished through modularization. For the piloted mission, the MPS is located on three modules; the Propulsion Module, the Ascent tankset, and the Lander. For lunar ascent, the propulsion module ascends with the ascent tanks, leaving the descent tanks with the lander on the lunar surface. For the cargo mission, the ascent stage is not needed and the associated ascent stage equipment (tanks, manifolds, flight vents, fill/drain ports, and valves) along with the valved disconnects between the Lander and the Propulsion Module are deleted and the feed lines from the Lander tank manifolds are plumbed directly to the propulsion module engine feed manifolds.

(1) Main Engines – The main engines used for the lunar missions are RL10A-4Bs, a modification of the LO2/LH2 RL10 A-3-3A, with a twenty inch length extendible nozzle yielding an expansion ratio of 84:1. The engines are rated at a maximum vacuum thrust of 20,800 lbf each with a specific impulse of 449 seconds with an oxidizer-to-fuel mixture ratio of 5.5:1. A more complete description is presented in figure 3.2.4-3.

The current design for thrust vector control of each engine includes two electromechanical ball-screw linear actuators equipped with redundant electric motor drives. Recognizing the high-power demand and inherent mechanical disadvantages of electromechanical ball-screw actuators, a promising alternative design includes self-contained electrohydraulic actuators powered by a turbo-alternator driven with hydrogen gas drawn from the LH2 tank pressurization line.

The main engines are designed to be capable of starting at zero NPSH with either liquid or vapor at the interface in order to settle propellants for full thrust operation. They also include provisions for supplying autogenous tank pressurization gases once the engines have been started, to ensure homogeneous tank pressurization as the tanks drain.

(2) Propellant Tanks – Both the Ascent Stage and Lander tanksets are launched full, with external load-bearing tank support structures designed to support the tanks during launch and lunar landing. The lander tanks are single wall 2090 Al-Li structures and are rigidly attached to the landing

	RL10A-4B
Vacuum Thrust, lbf	20,800
Chamber Press, psia	565
Area Ratio	84:1
Specific Impulse, sec. (1)	449.5
Operation	Full Thrust
Conditioning	Overboard Dump
Weight, lbm	385
Length, inches	70
Diameter, inches	46
Life, missions	1
Availability	Jan 1991
DDT&E, $M ('89)	10
Unit, $M ('89) (2)	2.3

(1) At MR=5.5
(2) Based on a lot of 66

Figure 3.2.4-3 RL 10A-4 Description

platform structure, although they could be designed for easy removal on the lunar surface. They are supported for launch thrust and lunar landing loads by the landing gear and tank support structure and stabilized laterally by graphite/epoxy tankset support struts with titanium end fittings.

The Ascent Stage tanksets are also single wall 2090 Al-Li structures supported inside a graphite/epoxy honeycomb shell that distributes launch loads from an aft launch vehicle interface ring, through the Lander and Propulsion Modules, and on to the equipment and crew modules. The tanks are supported in the shell structure by thermally optimized Passive Orbital Disconnect Struts (PODS). The struts are under development at NASA JPL and consist of concentric composite tubes; the outer tube designed for ground, launch and lunar landing loads and the inner tube for smaller on-orbit and lunar stay loads . Once in orbit, and after lunar landing, the outer tube pulls away from the inner tube, reducing the heat leak through the struts.

(3) Propellant Feed and Pressurization – The propellant feed and pressurization system is designed to minimize changes between the piloted and cargo mission configurations, minimize propellant loss due to boil-off during coast and lunar stay segments, and provide high system reliability. This is accomplished by modularizing appropriate components, by allowing depressurization of the propellant tanks (thereby reducing propellant temperature), and by providing an appropriate amount of checks and interlocks.

For the Lunar lander segment, propellant is fed from tanks on the Lander through valved disconnects to the Propulsion Module, enabling staging after the lunar stay, or during an abort. During an abort, continuous propellant flow is provided to the engines during tank switch-over and staging. As the descent tanks near depletion, the ascent tank pressure-isolation valves open, pressurizing the tanks. When the lander tanks are drained, the ascent tank propellant valves are opened and propellant flows from the ascent tanks. Check valves on the lander tank lines ensure that propellant does not flow back into the lander tanks. After propellant flow from the ascent tanks has been verified, the lander tanks propellant acquisition and pressurization valves are closed and the system vented, after which staging occurs.

Propellant feedlines are vacuum-jacketed, insulated stainless steel lines and include restrained bellows joints that articulate to compensate for thermal contraction and engine gimbal motion. Main feedlines and manifold are 6.0 inches in diameter for both LO2 and LH2, and engine feedlines are 2.5 inches in diameter. All valves are electromechanically actuated normally-closed valves and disconnects are rise-off-actuated.

Propellant gauging is accomplished by pressure-volume-temperature (PVT) type sensors that are being developed by Ball Aerospace for NASA JSC. In principle, they give a reading of the amount of propellant in a tank in low gravity regardless of liquid orientation, not requiring settling thrusts as might be required for an array of distributed point sensors. If the PVT gauge fails, then extra settling thrusts could be done to gauge the propellant with a backup system of distributed point sensors. The extra propellant required for this would translate into reduced lunar surface stay time because less lunar boiloff could be tolerated. The propellant gauging sensors are included in the vehicle instrumentation system.

Tank pressurization is autogenous and includes pressurization lines for delivery of pressurization gases (GH2 and GO2) from the engine-mounted bleed ports through manifolds and valves and to the individual propellant tanks. A helium pressurization system mounted on the propulsion module provides pressurant until the engines are started. Check valves ensure that helium is not lost through the engine bleed ports, and that GH2 or GO2 is not fed to the helium tanks.

(4) Propellant Fill and Drain – Provisions are also made on each module for vented fill and drain of the tanks associated with that module. The propellant fill and drain system includes 4.0-inch vacuum-jacketed lines, valves and disconnects from the launch vehicle, or ground supply interface to the main engine feedline manifolds.

(5)Tank Vent and Relief – Two separate tank vent and relief systems for both fuel and oxidizer are provided, one for the Lander and one for the Ascent Stage. During fill and ground-hold operations, the ground-vent system maintains acceptable tank pressure by venting gas overboard. In space, the thermodynamic vent system (TVS) combines several thermal control functions, releasing propellant boiloff gases overboard to maintain acceptable tank pressures, acting as heat exchangers to draw heat from the remaining liquid, and acting as mixers, creating a fluid jet to keep the propellant well mixed and equalizing pressures throughout the tank.

The TVS-mixer unit controls tank pressure in orbit by accepting either vapor or liquid at its inlet, expanding it through an orifice (thereby cooling it), and then extracting heat from the remaining tank fluid in a heat exchanger before being vented overboard. A small, highly reliable pump provides liquid flow through the warm side of the heat exchanger and also serves to keep the tank contents well mixed.

b. Reaction Control System – The Reaction Control System (RCS) provides attitude control during coast periods, rendezvous and docking maneuvers, lunar landing, and atmosphere reentry and provides limited delta-V capability. The Reference LTS Concept RCS is a supercritical GO2/GH2 system, selected for its minimal system weight, singularity of propellant types aboard the vehicle, clean effluents, and overall system reliability.

For the piloted mission configuration, two RCS systems are provided, one on the Lander Module and one on the Ascent Stage. The additional Lander system is provided because of the greater vehicle mass and the reduction in propellant required due to the greater thruster moment-arm allowed. During lunar transfer, insertion, and descent, both systems are active, increasing vehicle control.

Schematics and thruster arrangements for the Lander Module and Ascent Stage are shown in Figures 3.2.4-4 through 3.2.4-7. For each system, two O_2 and two H_2 tanks are provided and filled before launch. While one set of tanks is supplying fuel to the thrusters, the other set can be refilled, being supplied from the Ascent Stage main tanks for the Ascent Stage RCS and from the Lander Module for the Lander RCS. The

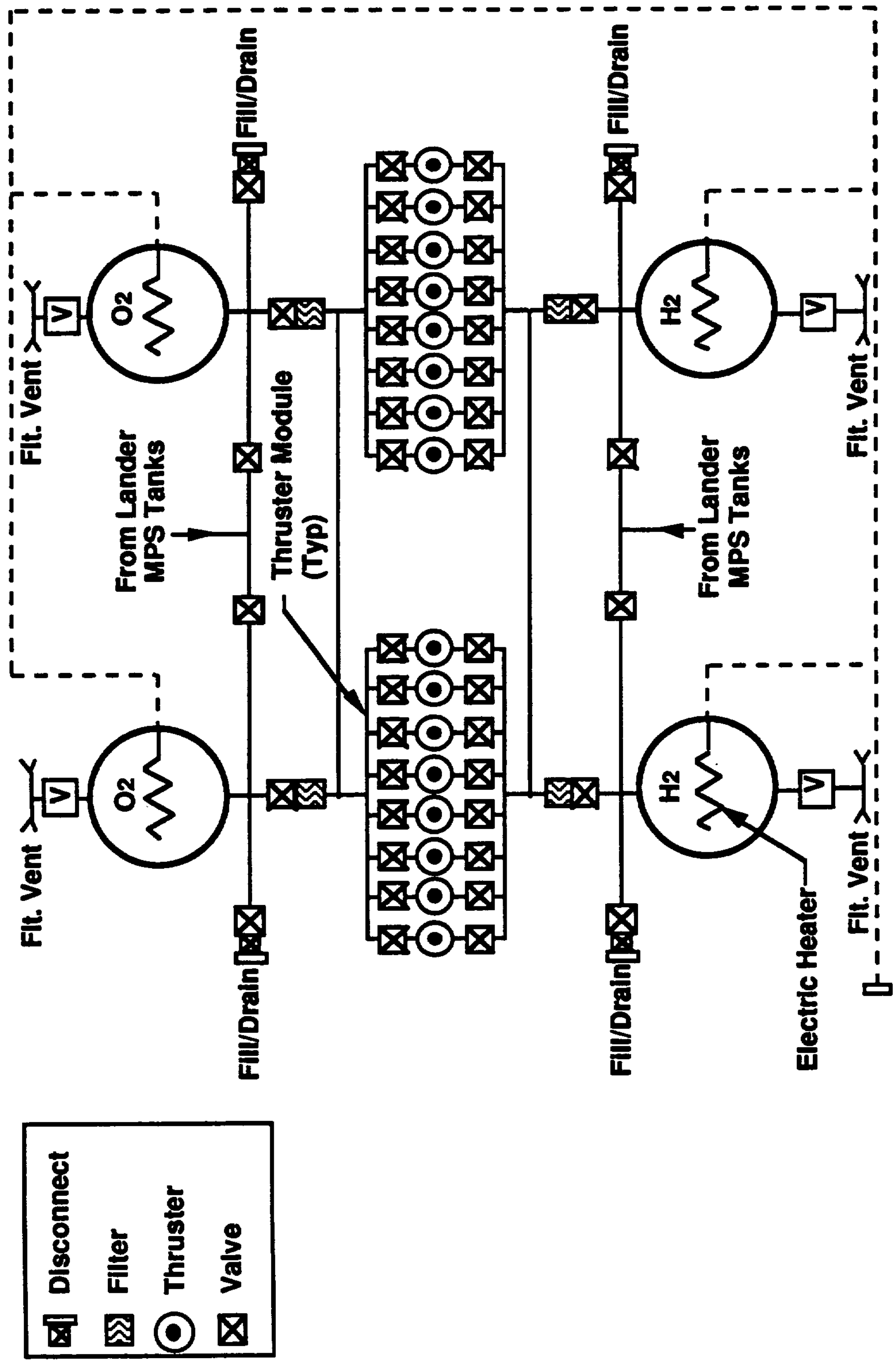


Figure 3.2.4-4 Piloted Lander RCS Schematic Reference LTS Concept

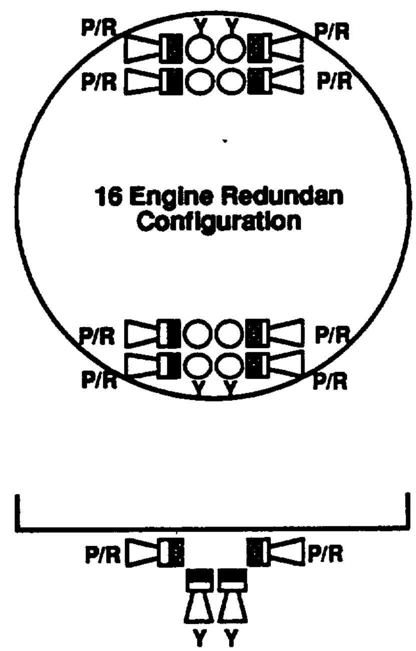


Figure 3.2.4-5 Lander RCS Thruster Arrangements Reference LTS Concept

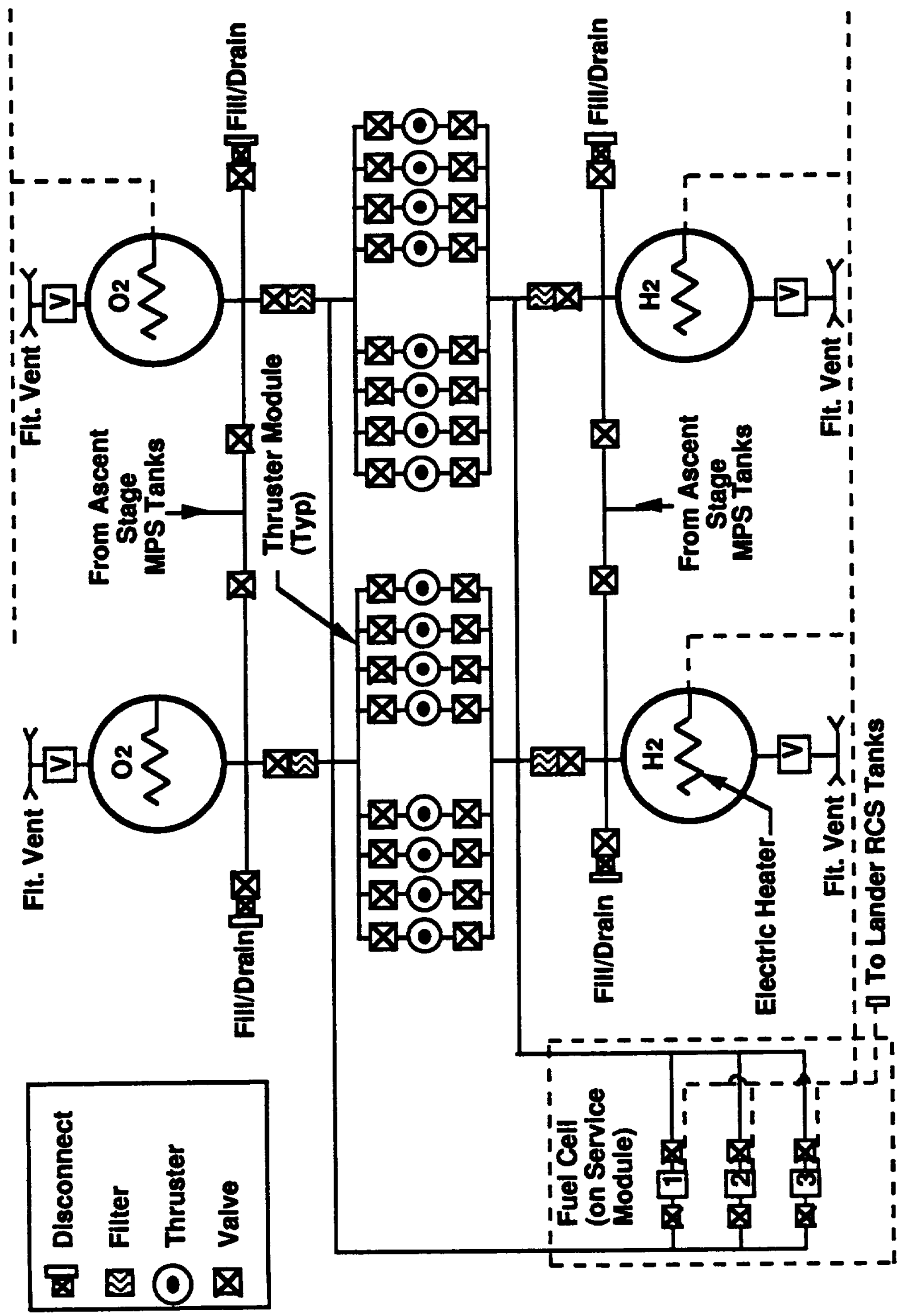


Figure 3.2.4-6 Ascent Stage RCS Schematic Reference LTS Concept

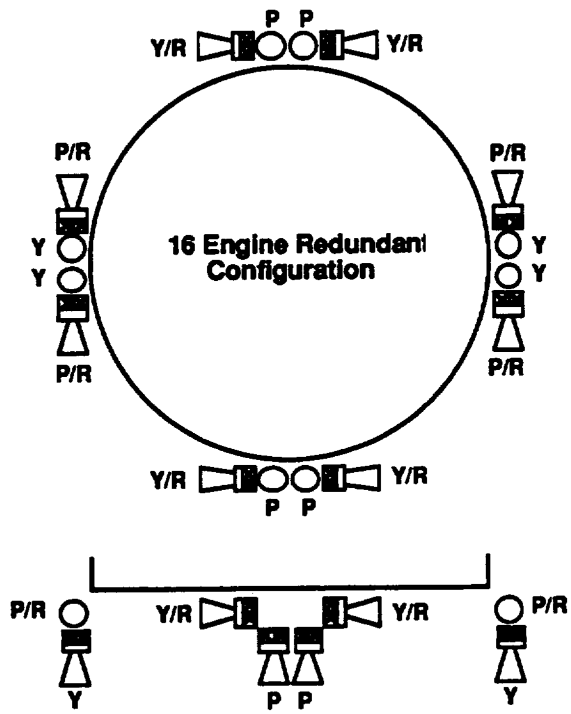


Figure 3.2.4-7 Ascent Stage RCS Thruster Arrangements Reference LTS Concept

RCS propellant tanks are pressurized by heating the propellant in the tanks to supercritical pressure. The tank heaters are powered by fuel cells, located on the Service Module and fueled by the Ascent Stage RCS tanks.

For the Cargo mission, as shown in the schematic in Figure 3.2.4-8, the Lander RCS tanks would be resupplied from the Lander fuel tanks and the fuel cells, located on the Service Module, would be fueled by the Lander Module RCS tanks. Additionally, the fuel cells would also be supplied fom the Lander RCS tanks.

c. Electrical Power – The electrical power subsystem hardware includes a power source, distribution and control components, and associated cables and wire harnesses for power distribution. It features redundant O2/H2 fuel cells fed from accumulators filled from the vehicle main propellant tanks, as well as distribution and control units and associated wire harnesses. The power supply is located on the equipment module, with interfaces to the crew module and ascent stage or lander for power distribution.

The primary power sources for all on-board systems are three 28 VDC, 4.6 kilowatt hydrogen/oxygen fuel cells. The fuel cells are derived from the current STS design, but operate on propellant-grade reactants, and are reduced in size from the STS design due to lower power requirements. On the STV, each fuel cell consists of two stacks of 32 cells each, with an nominal power output of 4.6 kilowatts. With three running continously, the total power output is 14.0 kilowatts nominal, 24 kilowatts peak. In the event of a fuel cell shutdown, the remaining two fuel cells can provide mission power requirements. In the event of two fuel cell shutdowns, the mission would be aborted, and the remaining fuel cell could provide emergency power to critical subsystems for abort capability. For peak power loads during main engine actuation, three expendable Lithium Thionyl Chloride (LiSOCl) batteries are included in the power supply to supplement fuel cell power. The batteries are sized to provide a total of 5.0 kilowatt-hours of power to the main engine actuators.

Fuel cell reactants are drawn from accumulators included in the Reaction Control Subsystem. The redundant accumulators are sized to provide oxygen and hydrogen reactants for both RCS and EPS functions for a period of time needed to fill the other accumulators. Once filled, the reactants are isolated and heated to supercritical

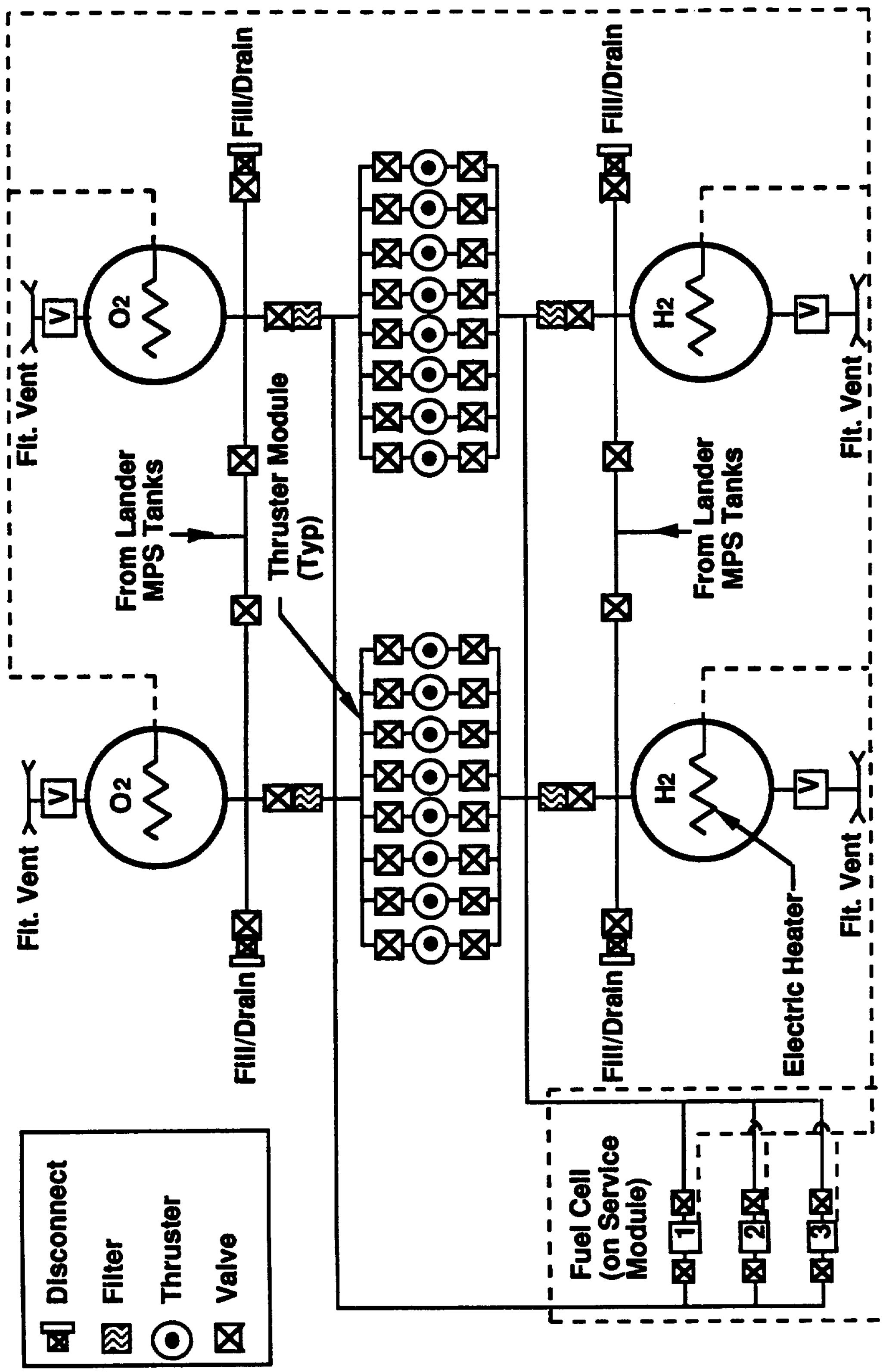


Figure 3.2.4-8 Cargo Lander RCS Schematic Reference LTS Concept

pressure. Reactants are then drawn off to supply the fuel cells through a system of CRES manifolds.

The power distribution system consists of power distribution and control assemblies, inverters, and remote switching devices. The power distribution assemblies interface with other vehicle subsystems and external power supplies and provide relay switching functions required for control of discrete vehicle elements and power switching such components as heaters, transmitters, power amplifiers, and propellant management electronics.

Electrical inverters are included to supply three-phase power to such equipment as main engine actuators and valves, fuel cell controls, and certain ECLSS components. The inverters are similar to current Shuttle inverters.

For power transfer, the equipment module has wire harnesses and interfaces to the core stage and crew module. The crew module wire harnesses distribute power to ECLSS and crew displays and controls, and the ascent stage wire harnesses distribute power to health monitoring equipment, propellant management equipment, and main engine valves and TVC actuators, as well as to the lander.

d. Avionics – Guidance and navigation equipment provides the means to determine the flight path and attitude of the vehicle throughout the mission. Navigation computes vehicle position and velocity, and guidance provides autonomous trajectory control by adapting to dispersions in thrust, vehicle and payload cg variations, and unmodelled uncertainties. Attitude control provides "attitude hold" pointing, attitude rotation from one fixed attitude to another, and fixed rotation rate for mission-unique requirements. Propulsion control and critical fluids control accept attitude and velocity commands and provide required valve commands to RCS engines and valves. Adaptive guidance and control optimizes the trajectory to minimize the error, g-loading and constraints (such as heating rate during earth entry) for given center-of-mass offsets and other non-nominal dispersions. Robust flight controls provide control and command for vernier velocity changes as directed by guidance in presence of faulted jets, with sufficient control authority to provide required turning rates in space and orbital/entry maneuvers.

A precise navigation fix of position and velocity is required prior to all rendezvous, lunar landing or earth entry maneuvers. To provide the vehicle state vector, a set of six inertial grade Ring Laser Gyros (RLG) to measure direction of delta-V, and a set of six accelerometers to measure magnitude of delta-V are packaged in a Hexad Inertial Measurement Unit (IMU). The IMU skewed axis expands fault tolerance while minimizing the number of components. Growth to a less costly space qualified GPS/GLONASS-aided IMU is highly desirable. Both GPS and GLONASS systems are needed to provide a minimum of four state vectors because the only available GPS satellites are almost behind the earth and will provide at most only one state vector.

During orbital rendevous and docking operations (if required) a Ku-band communications antenna will be deployed to measure range, range-rate and angles for relative navigation to a target. New technology for microwave/RF fiber optic waveguides will allow remoteable antennas without excessive losses in transmission from PA output to antenna, relaxing antenna placement restrictions and reducing vehicle integration requirements. Non-cooperative targets will be tracked by skin tracking out to about 10 nm. For a cooperative target (transponder), maximum tracking distance is about 200nm. Antennas will be stowed prior to deorbit. A laser tracker could provide autonomous docking capability with a reflector target located on the target vehicle.

Communication capability is provided between the vehicle and all Earth and orbital support elements. This equipment is located on the crew module during a piloted mission and on the equipment module during a cargo mission .

The communication and tracking (CT) function provides reception of uplinked switching commands (if necessary), and downlink data and voice channels. S-Band is the primary low rate interface for downlink telemetry and voice (and uplink for an unmanned mission). Ku-Band is the primary high data rate 2-way link via Deep Space Network (DSN) used for digital, voice and TV communications with earth, provided the antenna/platform is not being used for rendezvous navigation. High resolution closed circuit CCTV, VHM, and science data dumps are possible with bandwidth in application access of 180 to 300 Mbps. Image compression chip technology may allow NTSC (color) quality communication over S-Band. Microwave/RF fiber optic cable waveguide technology would allow remote antenna placement from the power amplifiers. This reduces vehicle configuration and mission operations requirements.

Instrumentation and data handling subsystems provide all computation, health monitoring, and control of the vehicle and its subsystems. Vehicle Health Monitoring (VHM) is a rather new avionic function that extends individual subsystem built-in-test, condition monitoring, status monitoring and command state verification monitoring by considering the vehicle as a whole. The VHM function determines state of health of the vehicle and passes this information to a "system manager" which is the Mission Management (MM) function. Relation among disjointed subsystems and all vehicle stage elements are taken into account as an autonomous entity.

The avionics architecture includes a federated set of processors, as shown in Figure 3.2.4-9. The fault tolerant processors interface to three robust photonic networks that are contained in a common medium, resulting in a significant reduction of physical connectors, known to be the largest contributors to unreliability. Separation of signals is by wavelength division multiplexing. Functional partitioning of flight critical signals from essential and non-essential signals reduces validation costs and recertification when components are changed or new ones are added. The absence of MDMs between computers and subsystem sensors and effectors places requirements on subsystem components to be able to connect directly to the data buses (autonomy level 3). Appropriate redundancy coupled with physical separation of redundant channels gives rise to a "zero-down-time" network.

Bus network types that are current networks or about to have space application include: Shuttle 1Mbps data bus (pre MIL-STD-1553), US/NATO combat aircraft MIL-STD-1553B, MIL-STD-1773 the fiber optic equivalent of 1553 with transmissive or reflective needs, 10 Mbps IEEE 802.4 bus utilizing token passing as the access method of IEE standard 802 local area network (LAN); a potential network on Freedom, 50 Mbps HSDB Linear (SAE AS4074.1) and HSDB Ring (SAE AS4074.2) and 100 Mbps FDDI (Space Station). The three data bus media that form the physical layer for the above standards are twisted wire pair, coax and optic fiber.

The modern avionics trend is toward common modules and standard interfaces, allowing growth and technology changeout/upgrades without "gutting" the vehicle.

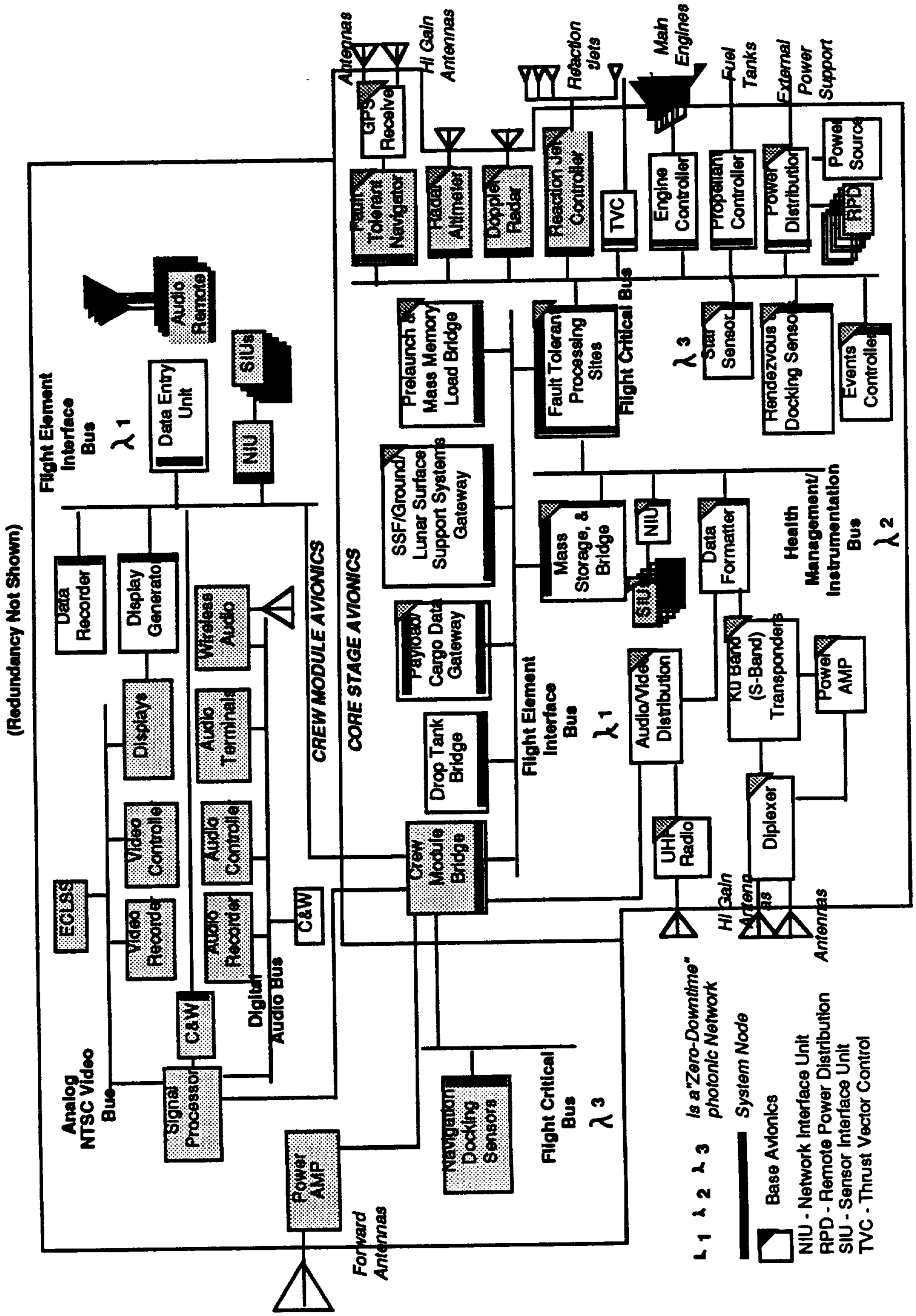


Figure 3.2.4-9 Avionics Layout

Implementation costs are reduced, maintainability (high level of BIT and standard interfaces) increases, and resource utilization is maximized since the system uses only a few module types (less than twenty). Some common types include Space Station Freedom DMS Standard Data Processor and a low power processor, both based on Intel 80386 instruction set, Network Interface Units, Bus Interface Adapter and MultiBus II backplane, and US Congress-mandated use of common modules by ATF (USAF), A-12 (Navy), and LH (USA). DoD's Joint Integrated Avionics Working Group (JIAWG) uses MIL-STD-1750A processors, 23-bit processors, SAE HSDB (linear), MIL-STD-1553, bulk memory modules, programmable input/output modules, and power supply modules. Standard requirements for each module include backplane interface, test/maintenance interface and BIT coverage.

Controls and Displays (MI) provide crew interface to the vehicle monitoring and control functions by providing color displays with graphics, icons and audible cues. The crew is given limited control and status monitoring of the vehicle during critical mission phases. Crew controls are simple menu selections since piloting skills may degrade after six months in lunar environment.

The current design developed in consultation with astronauts and crew systems experts features a system of three reconfigurable liquid crystal displays (LCD). The LCD's can display graphical or numerical output and are driven by separate controllers for redundancy. The displays and pushbuttons are reconfigurable and would assist in reducing information overload by presenting only data applicable to the current flight phase. This technology requires low power and is state-of-the-art in current military and commercial systems.

e. Environmental Control / Life Support –

(1) Environmental Control – Provisions on the crew module for atmosphere supply and control, internal equipment cooling, as well as metabolic and equipment heat rejection.

The environmental control and life support subsystem provides, monitors, and controls the crew module internal environment, as well as provides for crew safety and welfare.

Basic life support functions as applied to the STV can be grouped as shown in Figure 3.2.4-10. Seven of the groups are fundamental to crew life support, including atmosphere revitalization, temperature and humidity control, water management, health and hygiene, waste management, atmosphere pressure and composition control, and food management. Another group, fire detection and suppression, relates to protection of the crew in the case of an accidental fire. Lastly, EVA support is provided for ingress to and exgress from the crew compartment on-orbit and at the lunar base. To identify a life support system approach, these life support functions can be applied in an interactive system configuration, as shown in Figure 2.3.4-11. Shown are interfaces with other vehicle systems (i.e., fuel cells) as well as identification of additional requirements for storage facilities (i.e., trash). The system is an open loop life support system, with no regeneration of either atmosphere or water. This open loop approach was arrived at by analysis of an ECLSS closure break-even curve, as discussed in Section 3 subsystem trades. Since an adequate supply of water is provided as by-product of the fuel cell power supply system, only minimal water stores and supply tanks are required for STV, and recovery of cabin humidity condensate is not required. Atmospheric gases are supplied from storage, and carbon dioxide is removed from cabin air by replaceable LiOH canisters.

The schematic reflects the fault tolerance levels required for critical equipment, with triple critical system components rather than separate triple systems. For instance, there are three fans and three heat exchangers in the cabin temperature and humidity control circuit with any one fan and heat exchanger able to handle the total cabin heat load. The fan housing and ducting are considered passive components not prone to failure and therefore not requiring backup. There are three separate cooling water circuits feeding triple heat exchangers, three separate Freon circuits feeding up-sized single heat exchangers, and radiator panels containing triple fluid paths. There are also double backup cooling-water pumps and Freon circulation pumps.

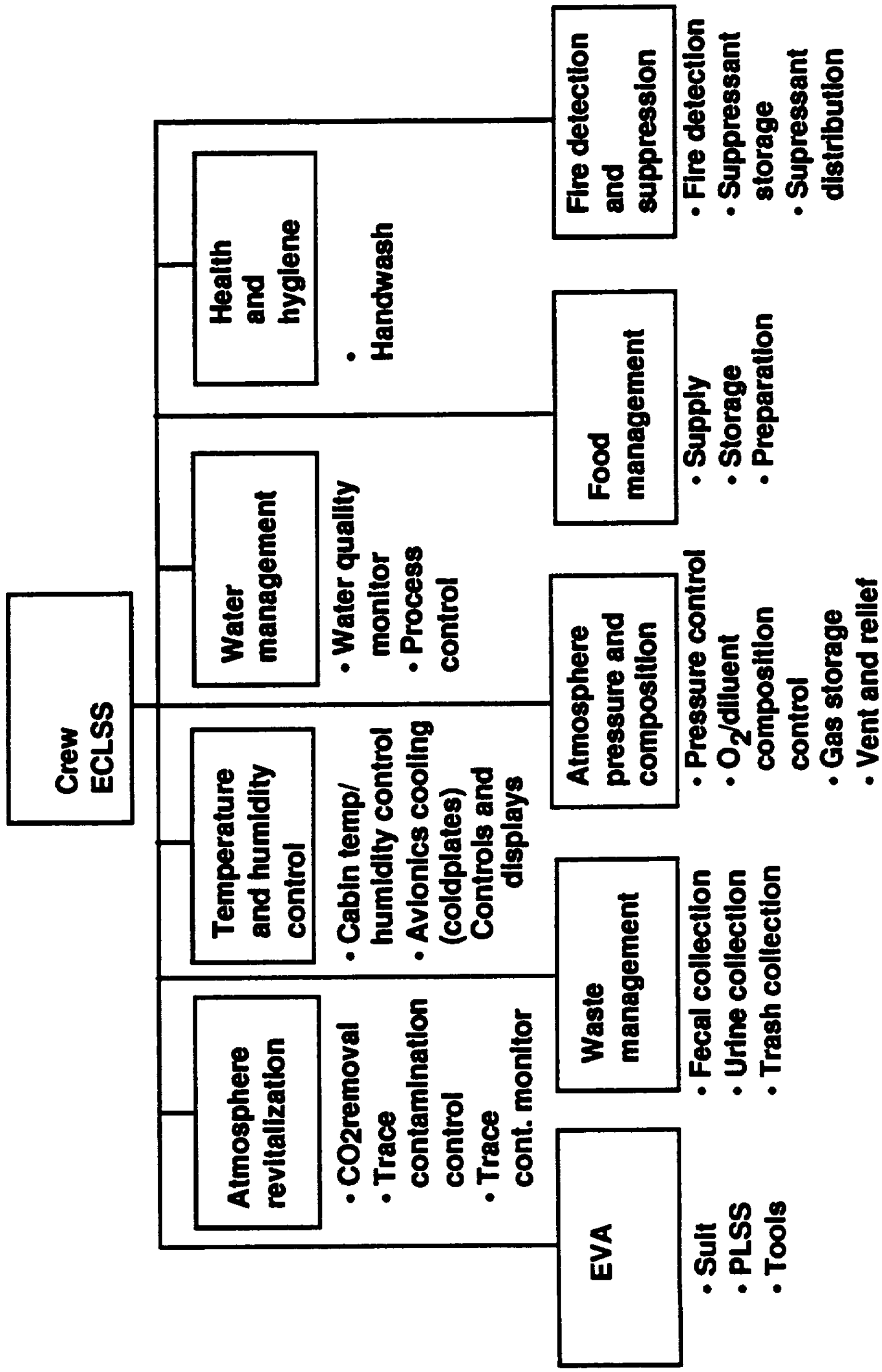


Figure 3.2.4-10 ECLSS Functions

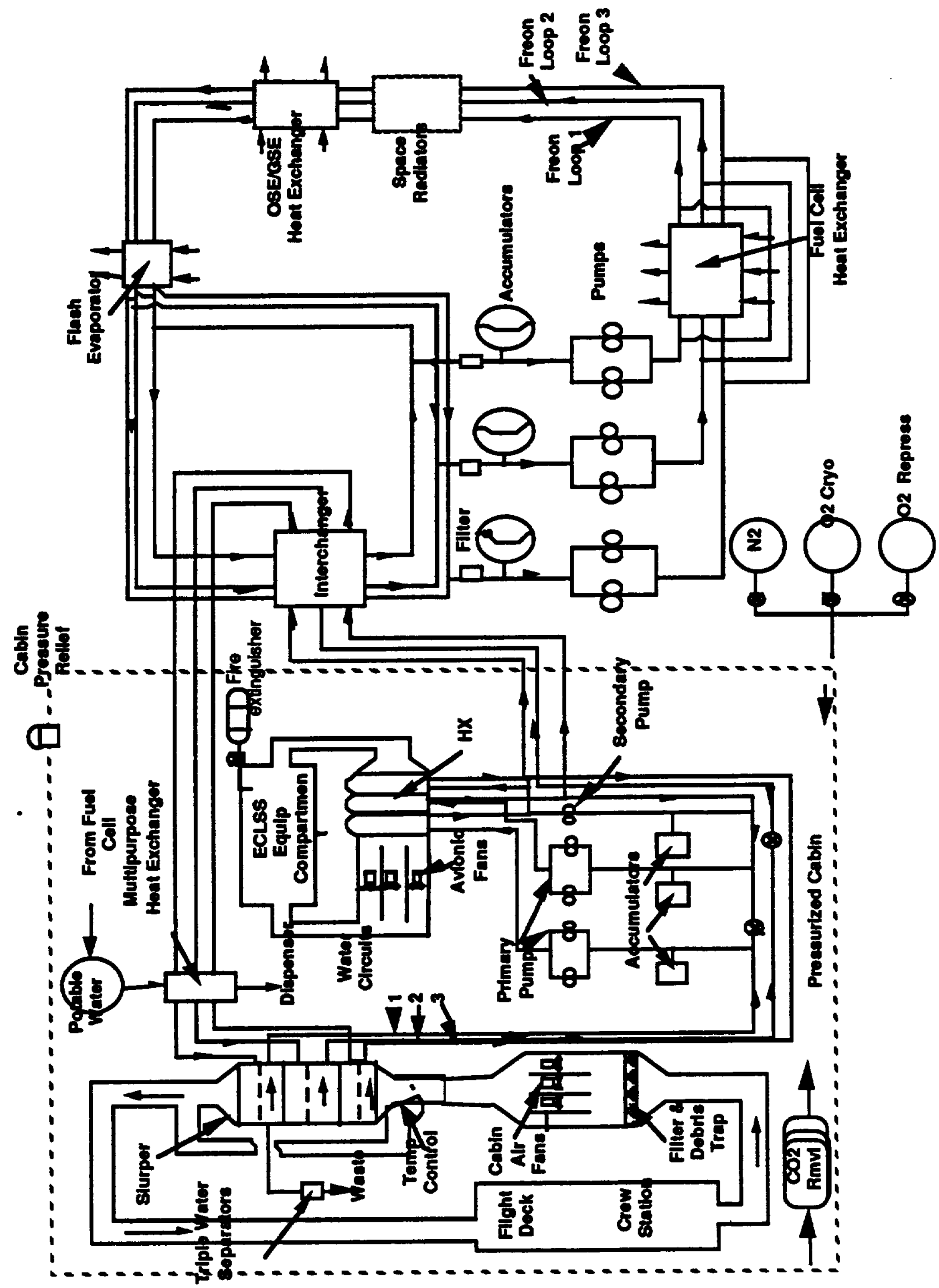


Figure 3.2.4-11 ECLSS Schematic

The pressurization and revitalization equipment maintains the crew module internal atmosphere and provides a shirtsleeve environment. The cabin pressure is 14.7 psi, with a composition of 21% oxygen and 79% nitrogen. The system is open-loop, with all gases supplied from bottles or accumulators.

For atmosphere pressurization, enough gaseous O2 and N2 is stored for two complete repressurizations of the crew module in case of atmosphere contamination. Metabolic O2 is drawn from the fuel cell reactant accumulators, where it is drawn from the main propellant tanks as liquid, heated, and stored at supercritical pressure. Cabin air is forced through filter canisters for contaminant removal and through LiOH canisters for CO2 removal. The LiOH carbon dioxide removal system is mostly passive structure with replaceable absorbent cartridges and does not require backup. The replaceable LiOH cartridges provide the necessary degree of redundancy, with additional cartridges provided for an abort mission (14.4 days). All consumable stores are sized to provide for the abort mission.

An active thermal control (ATC) loop is incorporated into the environmental control system, with coldplates for electronic equipment cooling, a cooling water loop for cabin thermal control, a Freon loop to cool vehicle heat loads, various equipment heat exchangers, and a variety of heat rejection devices designed for specific mission phases. Cabin heat loads are rejected to the water loop by the cabin heat exchanger, the avionics heat exchanger, the potable water heat exchanger, and the EVA/IVA heat exchangers. The water loop in turn rejects heat to the Freon loop by the Freon/water heat exchanger, and the fuel cells reject heat to the Freon loop through the fuel cell heat exchanger.

Heat rejection devices include ground support equipment (GSE) heat exchangers, water and ammonia flash evaporators, and space radiators. Prior to launch, heat is rejected through a GSE heat exchanger. During launch, passive thermal sink for initial liftoff and a water spray boiler above 140,000 ft are employed until the vehicle separates from the launch vehicle, after which radiators are deployed to reject heat. The water spray boilers may also be used to supplement the radiators during peak in-space heat load periods. During ground-based crew module reentry, the water spray boilers are used down to 140,000 ft, after which ammonia boilers are used for landing and post-landing. The radiators used for these vehicles are deployable triple-loop metallic radiators covered with a high reflectivity, high emittance coating. The radiators are jettisoned with the core stage prior to atmosphere reentry.

f. Personnel Provisions – Food, water, and waste management systems, as well as fire detection and crew furnishings on the crew module.

The fire detection and suppression system includes smoke detectors in the cabin and behind cover panels, as well as a central fire extinguisher, with ports in instrument panels and closed areas. Because fire poses a significant hazard in an enclosed pressurized environment, careful selection of internal materials will be essential to avoid toxic combustion by-products in the case of fire.

The food management system provides for the storage, preparation, and preservation of food for the crew. STV crew module food will be shelf stabilized, such as the type used aboard the shuttle. This food has a shelf life of about six months without refrigeration using the current Flight Equipment Processing Center (FEPC) packaging techniques. Shelf life can be extended by modifying the packaging approach, such as sealing the food in a controlled atmosphere, high in carbon dioxide and low in oxygen.

The water management system provides for potable water during the mission duration, and includes a water storage tank with water drawn from the fuel cell by-products, water dispenser, as well as tanks with a contingency water supply.

The waste management system includes urine and fecal waste collection bags with a partition for privacy and pre-moistened wipes for personal hygiene. It is believed that the exclusion of any kind of private facilities for the elimination of body wastes will be unacceptable to the crew, given the duration of the mission and the possibility of mixed-gender crews.

Crew furnishings include flight seats, emergency medical / health provisions, and personal equipment storage provisions. The flight seats are similar to those on the

STS Orbiter and provide restraint and impact attenuation for all phases of flight. They can be removed and stowed during flight and include a personal emergency air supply, similar to the Orbiter. The medical / health kit is provided for emergencies and health monitoring enroute to or from the lunar surface.

3.2.5 LTS Performance

Parametric payload performance of the LTS concepts previously described is shown in figure 3.2.5-1 for the Campsite mission. The variation in delivered lunar cargo with initial mass in LEO (IMLEO) is shown for dual-launch piloted or cargo missions (single-launch would be about the same) using either storable or cryogenically fueled ascent stages. The performance variation between RL10A-4 and RL10B-2 powered piloted cryo ascent stage is also shown.

These results indicate that a viable lunar exploration program could go forward even if only the first model of NLS Heavy Lift Launch Vehicle (HLLV) were available, since it is capable of putting almost 200 tons in LEO using two launches with suborbital burns of the TLI stage and lunar lander. A better approach, however, may be to build a new larger HLLV which could accomplish the piloted lunar mission with one launch and reduce the burden on ground facilities. A new modular ETO launch system with a degree of reusability should have launch costs comparable to, or less than, today's National Space Transportation System (NSTS). Since ETO launch costs are approximately half the total program Life Cycle Cost (LCC), significantly reducing the number of launches with a larger HLLV should significantly reduce LCC.

3.2.6 Program Cost Estimates

Since relative cost data for the leading candidate HLLVs is not available at the time this document goes to print, costs quoted in this section will be based on missions utilizing two launches of the reference NLS HLLV which has an adequate definition and costing basis.

Total LCC of the transportation portion of the "moon to stay" exploration scenario was estimated. This scenario was first derived during the NASA 90-Day Space Exploration Initiative Study and is very similar to Architecture III in reference 1. It requires approximately 418 tons of cargo delivered to the lunar surface and 17 visits by a team

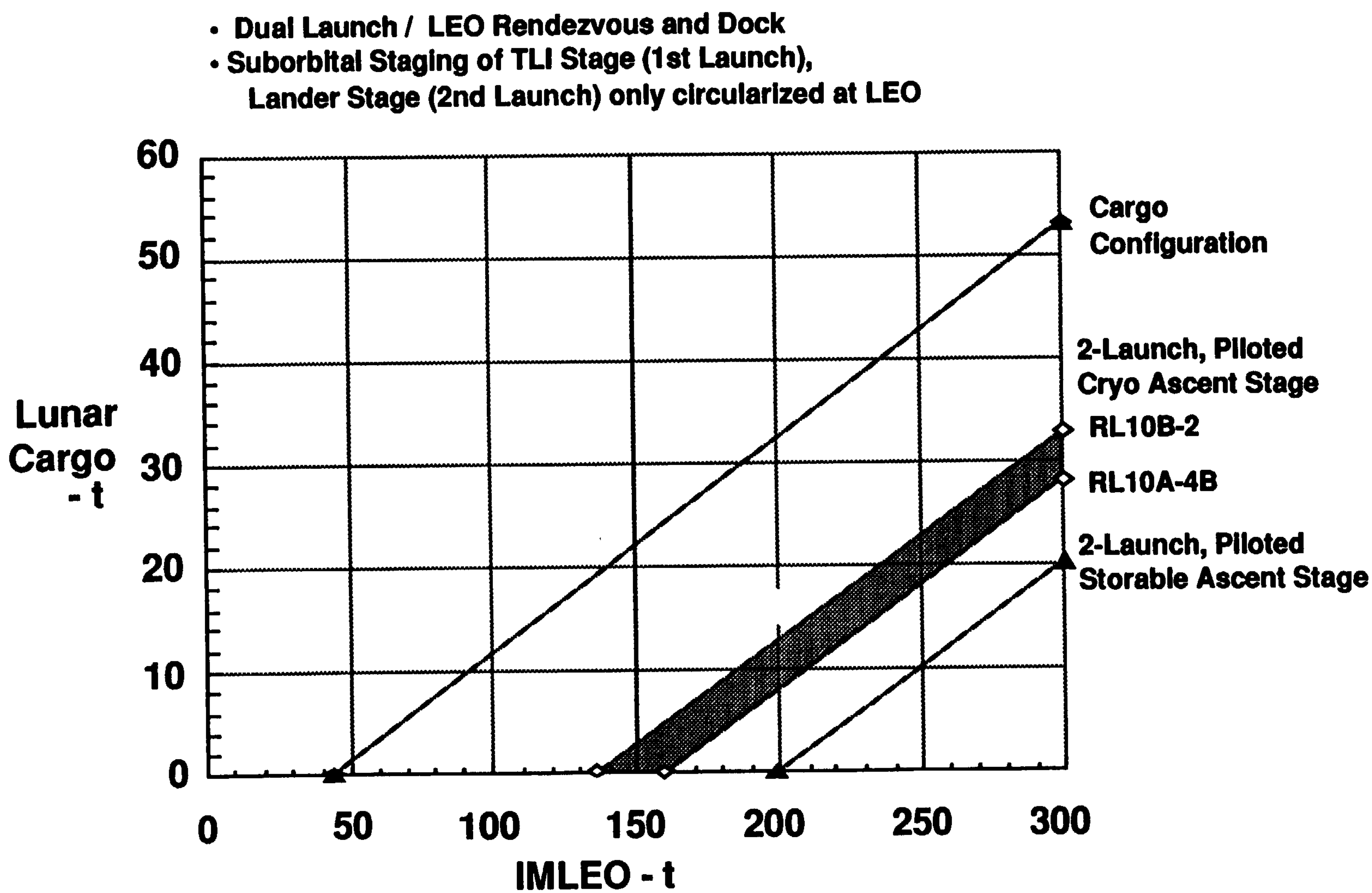


Figure 3.2.5-1 Performance Summary - Campsite Concepts

of four astronauts to explore and assemble first a lunar outpost, then a permanent lunar base.

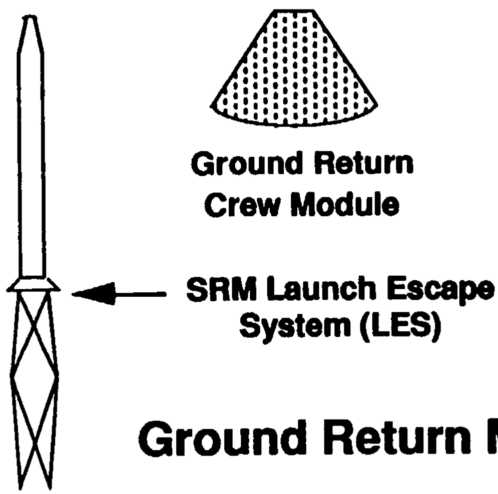
The assumed schedule was; authority to proceed with exploration program in 1995, first lunar explorer orbiter in 1999, first teleoperated mini-rover landed in late 2000, and first NLS single-launch precursor lunar cargo mission in 2002. The precursor cargo mission would deliver a pair of teleoperated exploration rovers with capability to place and maintain remotely operated science stations and conduct significant minerals prospecting. The first dual-launch missions would start in early 2004, with a dual launch then every six months. The first four launches would be to deliver and make operational the lunar outpost.

Key cost estimating assumptions were that: 1) the baseline NLS HLLV with solid boosters would be available for no additional development cost to the program (we did pay for developing the large 10m diameter shroud), 2) that the Cargo Transfer Vehicle (CTV) and NLSUS would already be developed so that appropriate subsystems, facilities, Ground Support Equipment (GSE), and test procedures would not have to be developed and qualified again, and 3) program factors would be: requirements growth = 30%, fee = 8%, government support = 15% for new manned elements and 5% for existing or unmanned elements.

Other cost estimating groundrules were that: 1) all estimates in constant-year 1991 dollars, 2) LTS launch site is KSC, Florida, 3) there are four equivalent ground test articles (one fatigue, one functional dynamic ground test, one propulsion test, and one qualification/pathfinder), and 4) there are three LTS flight tests (one TLI stage launched to LEO, a second launch of the lander assembly to demonstrate automated rendezvous and docking, and the single launch precursor cargo flight (off-loaded propellants and payload) to demonstrate automated lunar landing).

The estimated costs for the crew module and launch escape system are shown in Figure 3.2.6-1. The total DDT&E cost is $6.6B and the Theoretical First Unit (TFU) cost is $660M. Similar data for the LTS core stage is shown in Figure 3.2.6-2. DDT&E for the core stage is $1.1B and TFU is $108M. There would be a large cost savings on the core stage because it was developed from the NLSUS.

Figures are not shown for the cost estimate of the two 10m diameter fairings, but the 95 foot long cargo fairing had a DDT&E of $400M and a TFU of $84M, while the 65 foot



- Boeing Parametric Cost Model (PCM) is used to generate all flight hardware DDT&E & Theoretical First Unit (TFU) trade study estimates; schedule penalty is eliminated.
- Crew module estimate includes: operational module set(s), static/dynamic test articles, SIL articles, trainer, 1 mockup.

	Dry Mass (lbs)	DDT&E	TFU (Prod.)
Ground Return Module	14,674	$ 3,720 M	$ 372 M
Engineering & Dev.		369	
Ground & Flt. Test Hdw.		1,644	
Other Contractor Costs		1,707	
Launch Escape System	7,700	387	35
NASA Program Factors (61.5%)		2,526	250
Total Estimated Cost to NASA		$ 6,633 M	$ 657 M

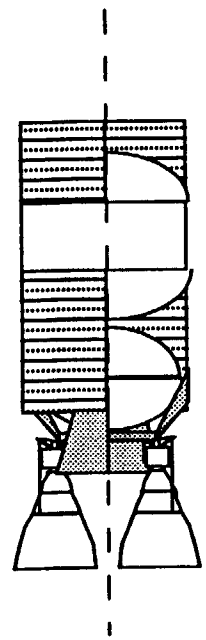
The LTS Earth return module & LES estimates are based on the previous LTS studies review cost analysis data.....

Figure 3.2.6-1 Earth Return Crew Module & LES Estimates

(Constant-year, 1991 dollars in millions)

The RL10-B2 mod. & test costs are $325M (29%) of the total DDT&E

Basic Core Stage



	DDT&E	TFU (Prod.)
Core Stage	$ 767 M	$ 75 M
Engr. & Dev.	339	
Grd. Test/Flt. Hdw.	311	68
Systems Test	62	
Tooling & STE	8	
Other Costs	47	7
Prog. Factors (47.4%)	363	33
Total Est. Cost to NASA	$ 1,130 M	$ 108 M

Estimated Dry Mass (lbs.) = 5,968

Excludes: drop tanks, service module avionics/RCS, & lander hardware.

Figure 3.2.6-2 Boeing LTS Core Stage Estimates long core/lander fairing which mounted the LES had a DDT&E of $366M and a TFU of $71M.

The LTS mission-peculiar hardware, such as the drop tanks, lander assemblies, service modules, and launch adapters were estimated separately and are shown in Figure 3.2.6-3. Note that the drop tanks and service modules vary between piloted and cargo delivery flights, and are costed by mission type.

Cost estimates for the final piece of the LTS, the TLI stage, are shown in Figure 3.2.6-4. This stage, like the core stage, also benefits from the earlier development of the CTV and NLSUS. The total DDT&E estimate is $1.1B and the TFU is $220M.

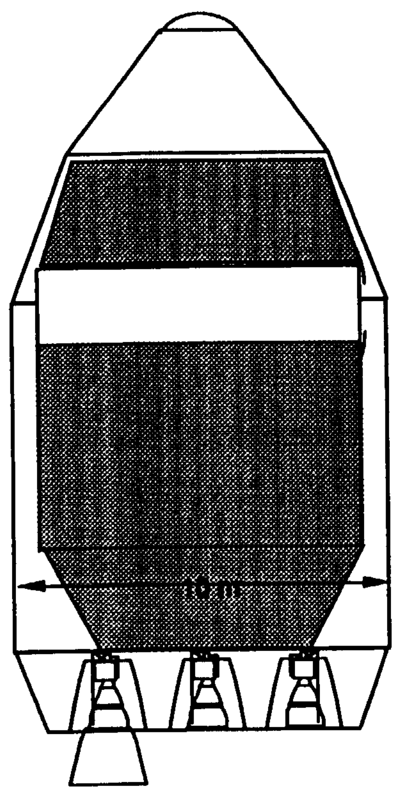
The total LTS acquisition cost is summarized in Figure 3.2.6-5. The procurement costs are for 32 sets of LTS hardware, enough for 30 lunar missions and two flight tests during full scale development.

The total cost of operations and support over 22 years of the program were estimated to be $16.6B. This breaks down to be: $10.4B for 64 launches of the NLS derivative vehicle, $2.1B for mission operations, $1.5B for LTS processing, plus crew module recovery and refurbishment, $930M for software maintenance, $800M for government program support, $500M for facilities maintenance, and $320M for spares and repair parts management.

Combining the acquisition cost with the operating and support costs we get just under $51B as the total transportation LCC over the 22 years of the lunar exploration program. This is for 17 piloted missions lasting up to six months and 13 cargo missions to set up and service a permanent lunar base over a 17 year period. This compares to $36B (in 91$) for the Apollo program which placed 10 men on the moon for very brief stays and returned 800 lbs of rocks. Unfortunately, the $51B does not include the cost of the surface systems, which will be the order of $5B to $10B, and even the $3B to $4B per year peak funding to support LTS development appears to be beyond NASA resources about the turn of the century. Accordingly, we looked at ways to reduce costs even further, and ways to improve public awareness and support for the lunar exploration program in general.

Note, that significant reductions in estimated LCC have already occurred to reach the levels shown here. The LTS proposed in reference 1 had LCCs of around $75B for

(Constant-year, 1991 dollars in millions)



	DDT&E	TFU (Prod.)
TLI Stage	$ 738 M	$ 149 M
Engr. & Dev.	203	
Grd. Test/Flt. Hdw.	407	129
Systems Test	36	
Tooling & STE	49	
Other Costs	43	20
Prog. Factors (47.4%)	350	70
Total Est. Cost to NASA	$ 1,088 M	$ 219 M

Estimated Dry Mass (lbs.) = 18,274

(Development includes three test flight hardware vehicles.)

Figure 3.2.6-4 Boeing TLI Stage Estimates

(Constant-year, 1991 dollars in millions)

Lunar Transportation System Element	DDT&E	Production	Total
Core Stage (GR-2.5S)	$ 1,130 M	2,415	$ 3,545 M
Lander	467	1,788	2,255
Cargo Service Module	638	1,321	1,959
Manned Flight Module	(incl. above)	1,410	1,410
Cargo Drop Tanks Set	408	1,082	1,490
Manned Flt. Tanks Set	(incl. above)	1,462	1,462
Software	1,000	-	1,000
Crew Module & LES	6,633	3,002	9,635
LTV Subtotals -	$ 10,276 M	12,480	$ 22,756 M
TLI Stage	1,088	5,043	6,131
Cargo Fairing	403*	844	1,247
Manned & TLI Fairing	366	2,164	2,530
Launch Adapters	325	850	1,175
TLI & Interface Hdw. -	$ 2,182 M	8,901	$ 11,083 M
LTS Facilities (KSC & JSC)	$ 397 M	-	$ 397 M
Totals by Phase -	$ 12,855 M	21,381	$ 34,236 M

Note: All estimates shown include the NASA directed program factors.

Note *: Revised for typographical error in original presentation chart.

Figure 3.2.6-5 Baseline Acquisition Cost Summary essentially the same exploration scenario. Ground basing the LTS eliminated $4.5B of on-orbit infrastructure, and making the LTS a close derivative of the NLSUS saved almost $6B in development costs. Manrating the NLS launch vehicle eliminated another $2.5B in STS launch support. This brought us down to $62.6B prior to the configuration cost optimization trades discussed here. These trades showed that two items; combining the transit and excursion crew modules into a single reusable module, and eliminating the third stage ascent module in favor of drop tanks on the core stage eliminated $11.6B of LCC.

Logically, the next step in reducing cost is to reduce the number of launches by using a larger, more capable booster. If one of the liquid-fueled boosters shown in Figure 3.1.2-1 were available, the number of launches could in fact be reduced by half.

Assuming the lunar exploration program would pay for developing the new liquid booster and the facility upgrades involved, we get from reference 5, a delta DDT&E of $4.7B and a TFU of $870M for a 150 t NLS derived launch vehicle. Unfortunately, this very high TFU puts the 150 t vehicle launch costs at more than three times those of the 70 t launch vehicle. The 70 t vehicle has lower launch costs because it has reusable boosters (ASRMs), and to be cost competitive the advanced liquid booster derivatives need to be made partially reusable too. Previous studies have shown that putting the liquid booster engines in recoverable propulsion modules can reduce cost per flight by up to forty percent (reference 6). A trade is required to see if the savings are adequate to justify a new 200 t launch vehicle for the lunar exploration scenario.

3.3. Phase 2 Trades and Analyses

The work performed under phase 2 of the STV Concepts and Requirements Study was done in three distinct tasks, including Task 1: A single-launch LTS design Architecture Study, Task 2: Crew Module designs, and Task 3: A Cost-optimum reference vehicle.

3.3.1. Single Launch LTS Architecture Study

Design groundrules assumed for the Task 1 analyses are listed in Figure 3.3.1-1. The significant groundrules include single-launch designs, use of low lunar orbit rendezvous, and no Space Station accommodations available. This period of study also focused entirely on a lunar transportation system (LTS). The major trades and analyses performed led to the reference concept described in the previous section. A summary of the trade evaluations follows. For this phase of study, the primary measures of goodness included mission performance and life cycle cost.

Crew Launch on HLV vs. STS The options available for the launch of the crew for a piloted lunar mission include launch aboard the LTS vehicle launcher and crew delivery via the STS Shuttle. In the case of launch aboard the launch vehicle, the crew module must be located at the top of the LTS stack in order to use a launch escape system in case of a launch abort.

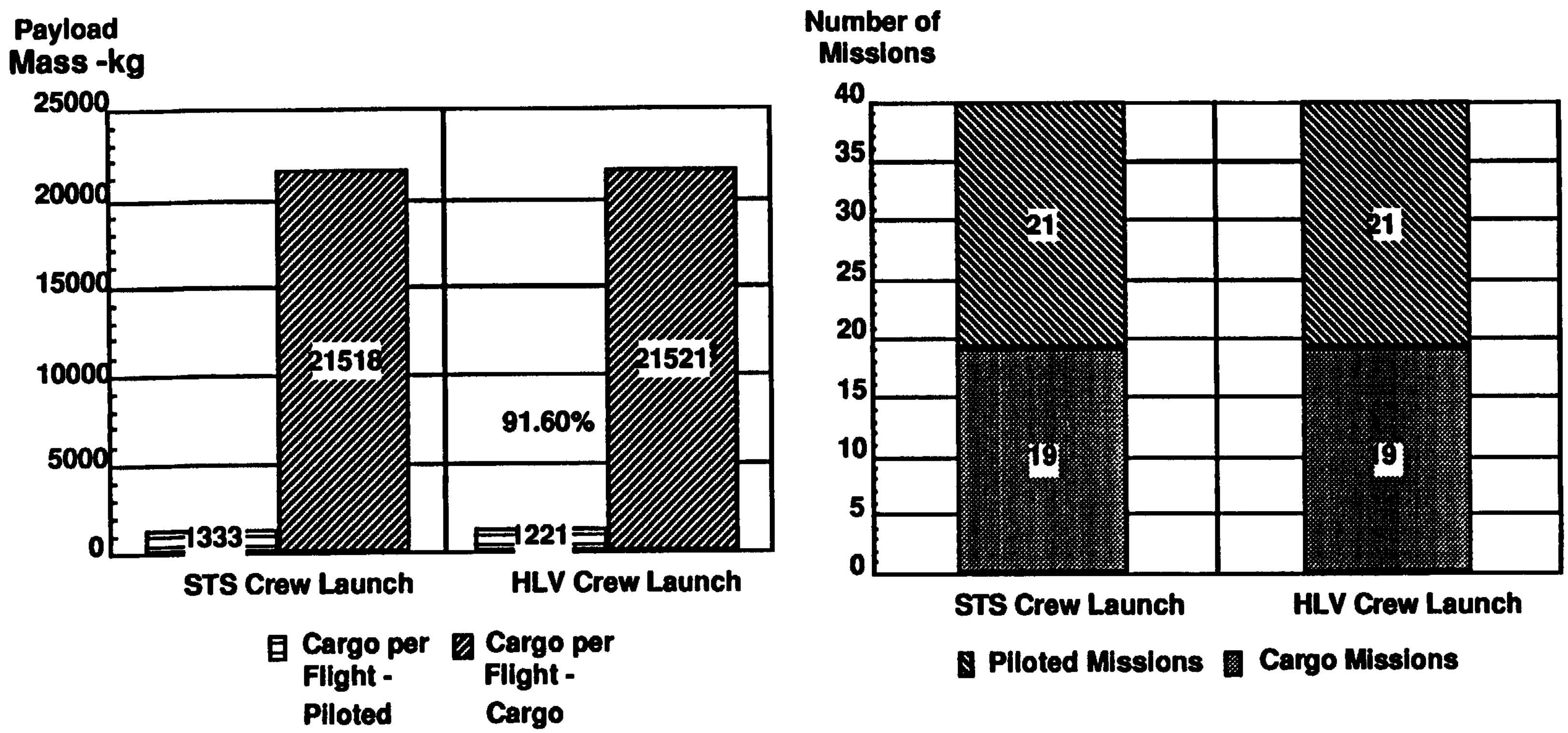
The performance and cost comparison of the crew launch options are shown in Figures 3.3.1-2 and 3.3.1-3, respectively. From the perspective of performance, there was little appreciable difference between the concepts, but from a cost perspective, the HLV launch option was 1.3% to 8.4% lower than the STS-launch option, depending on the portion of the Shuttle launch paid for.

Crew return The crew may be returned to the Earth in one of three ways. They may return directly to the ground via a direct-return crew module, or may be returned via the Shuttle following an aerobrake or all-propulsive burn maneuver and rendezvous with the Shuttle in low Earth orbit, as shown in Figure 3.3.1-4. In the direct-return case, the crew module is substantially heavier due to the necessary thermal protection required for Earth reentry, and thus mission performance suffers. In this case, an additional cargo flight is required to make up the performance difference, given similar launch vehicle capabilities, as shown in Figure 3.3.1-5. In the matter of cost, however, the

- **All vehicle concepts will require only 1 launch, except for options that bring up only the 4 crew in a Shuttle Orbiter and perform an Earth Orbit Rendezvous (EOR) for crew transfer**
- **All vehicle concepts will perform a Lunar Orbit Rendezvous (LOR), and utilize the orbit for a staging location**
- **Two failure tolerance on all safety critical components shall be required. Safety criteria used in JSC 2321 shall apply.**
- **Emphasis shall be placed on developing configuration trade data for the upper (transfer and landing) stages. Launch vehicle "optimization" to be performed inhouse. Costing and performance developed for the upper stages only.**
- **Cost is the key evaluation criteria. Performance is measured in total number of missions required to transport the total outpost cargo (418 tons). Piloted missions held constant at 21. Total missions then contribute to life cycle cost analysis. (If everything else is held equal, 1 extra cargo mission adds about 1.5% to the life cycle cost.)**

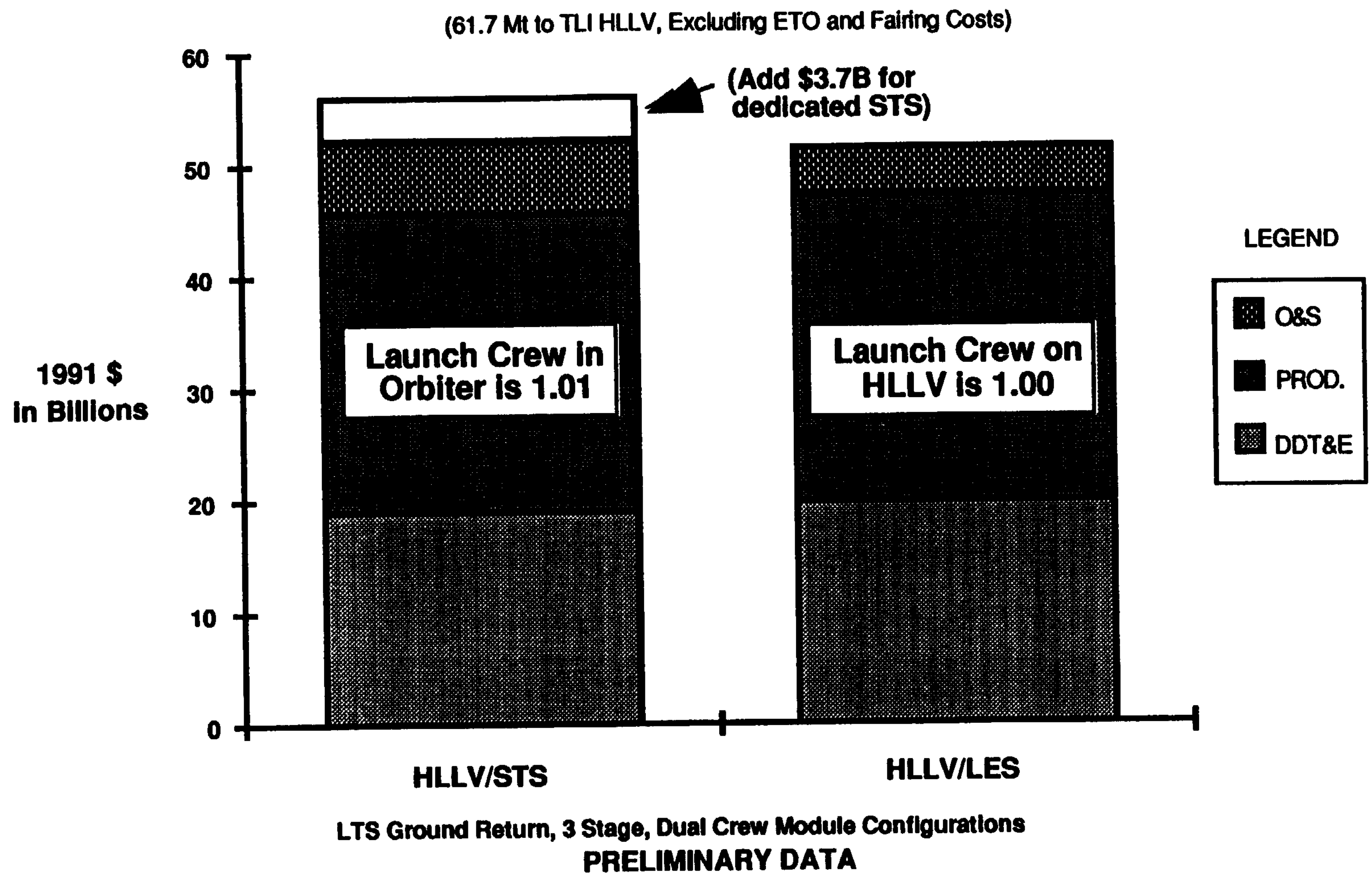
Figure 3.3.1-1 Task 1 Significant Design Groundrules

Ground-return, 3.0-stage, Dual Crew Module Vehicle Launch Vehicle Option #4 with Updated Shroud Configurations



Addition of a Launch Escape System results in no appreciable change in performance.

Figure 3.3.1-2 Crew Launch Trade Performance Results



STS Orbiter dedicated launch changes savings from 1.3% to 8.4%

Figure 3.3.1-3 Crew Launch Trade Cost Results

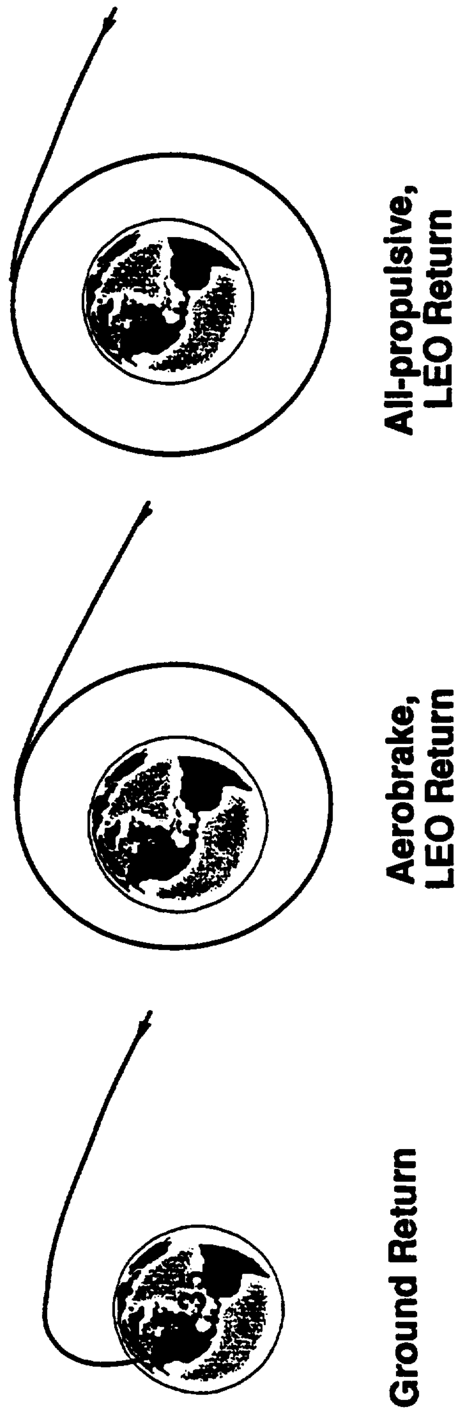
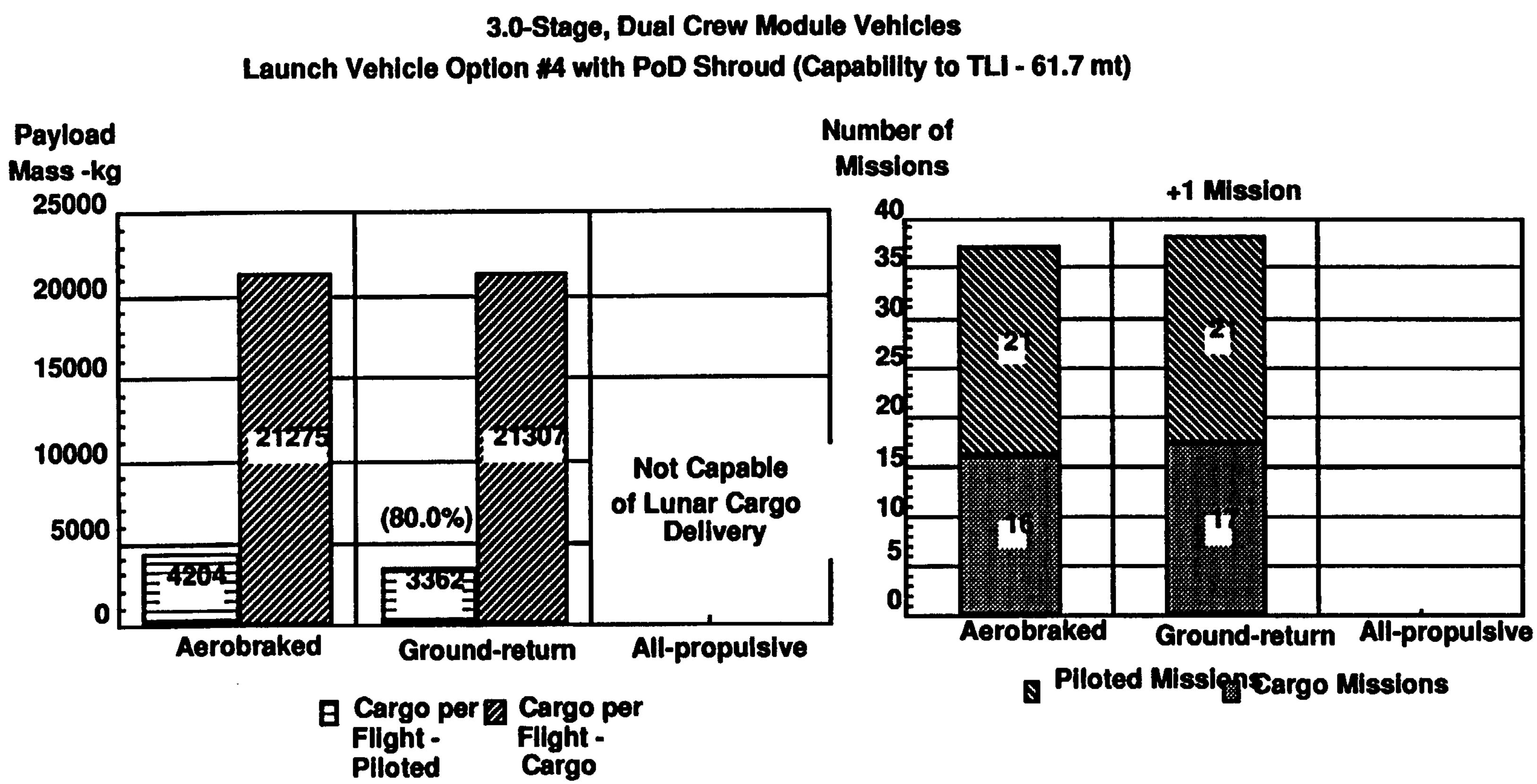


Figure 3.3.1-4 Crew Return Options



**Ground-return requires one extra cargo mission,
All-propulsive cannot perform the piloted mission.**

Figure 3.3.1-5 Crew Return Trade Performance Results ground-return system is favored by 1% to 6.3%, depending on the portion of the Shuttle flight paid for to retrieve the crew module and crew, as shown in Figure 3.3.1-6.

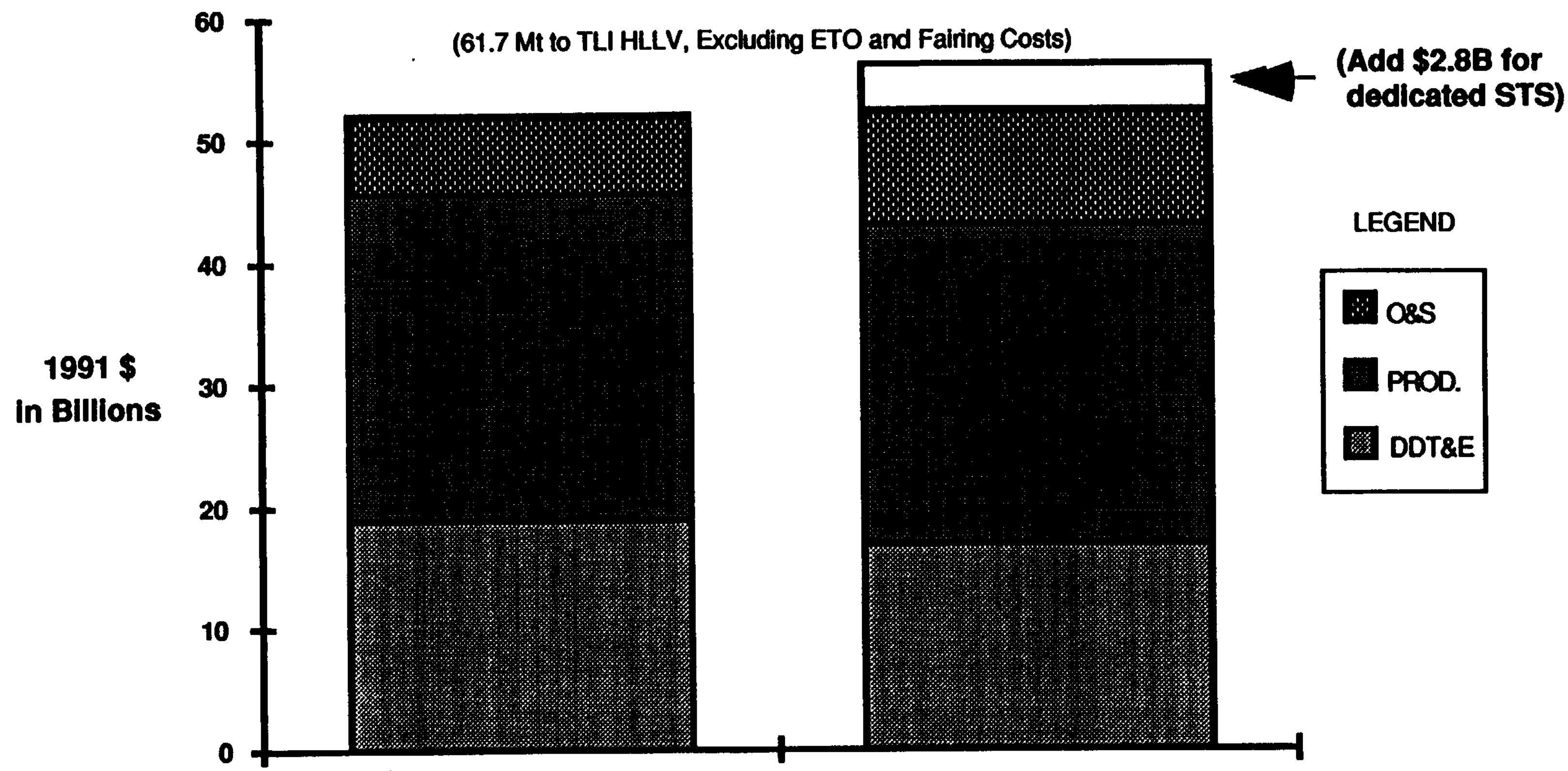
An additional issue is the matter of abort capability. If the mission is aborted early, and the crew must rendezvous with a Shuttle to return home, a rescue Shuttle flight must be made with little advance notice, or the crew module must contain sufficient supplies for a stay in LEO to await the Shuttle launch and rendezvous. With a ground-return crew module, however, the crew may return to Earth immediately in the case of an abort. A comparison of the impact of additonal crew module supplies for a 30-day and 60-day LEO stay is shown in Figure 3.3.1-7. In terms of additional cargo missions, one additional cargo mission is required for a 30-day LEO stay, and two additional missions are required for a 60-day LEO stay.

Number of Stages. The number of stages analyzed in this period of study included a 2.5 stage vehicle, a 3-stage vehicle, and a 4-stage vehicle. 1.5 stage vehicles were not considered because the TLI stage, assumed to be the launch vehicle upper stage, was included as one of the vehicle stages. In the 2.5-stage case, the LOI/TEI stage is replaced with a tankset, so that propellant for the transfer burns must be routed around the crew module to the lander stage, as shown in Figure 3.3.1-8.

From a performance point of view, the 2.5 stage concept is favored, with one less cargo mission required to deliver 418 tonnes of cargo to the lunar surface, as shown in Figure 3.3.1-9. From a cost standpoint, shown in Figure 3.3.1-10, the 2.5-stage concept is also favored, with 11% to 17% lower life cycle cost than the 3 and 4 stage concepts, repectively. The differences in cost are due to the additional propulsion systems developed and expended on each mission.

Ascent Stage Propellant. For a 6-month stay on the lunar surface, cryogenic propellants will experience some boiloff, resulting in a decrease in delivered cargo capability. With an ascent stage that uses storable propellants, this boiloff would be eliminated, but the lower performance of storable propellants would result in performance degradation. The addition of storable propellants also adds an extra stage and corresponding propulsion system.

In vehicle performance, the cryo vehicle is favored because of its higher engine specific impulse, assuming lunar surface propellant refigeration is available, as shown in Figure 3.3.1-11. An additional two missions are required for the storable propellant



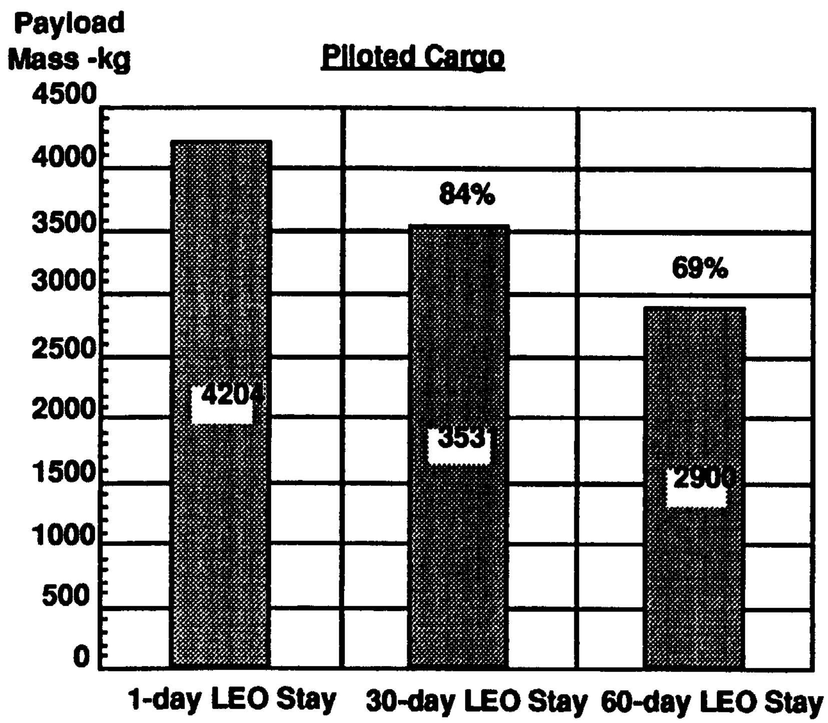
Boeing 3 Stage Configurations, with Dual Crew Modules

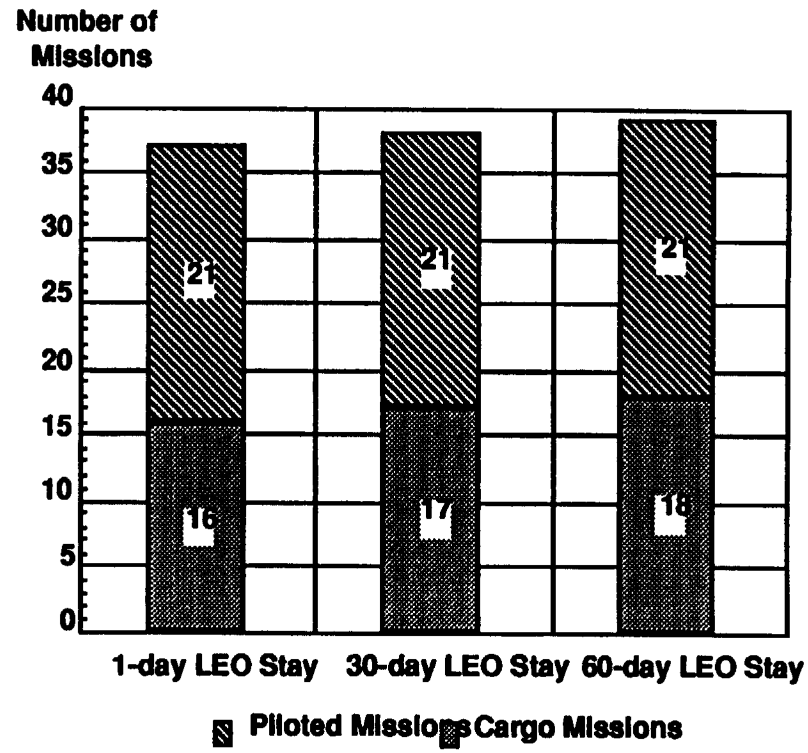
Note: Ground return modules = 4; aerobraked modules = 3 in LCC's

Ground return has minor cost savings (1% to 6.3%)

Figure 3.3.1-6 Crew Return Trade Cost Results

Aerobraked, 3.0-stage, Dual Crew Module Vehicle
Launch Vehicle Option #4 with PoD Shroud (Capability to TLI - 61.7 t)





Provisioning for a 60-day contingency stay in LEO would result in a 31 percent reduction in piloted cargo capability, adding 2 extra cargo missions.

Figure 3.3.1-7 LEO Staytime Impact on Vehicle Performance

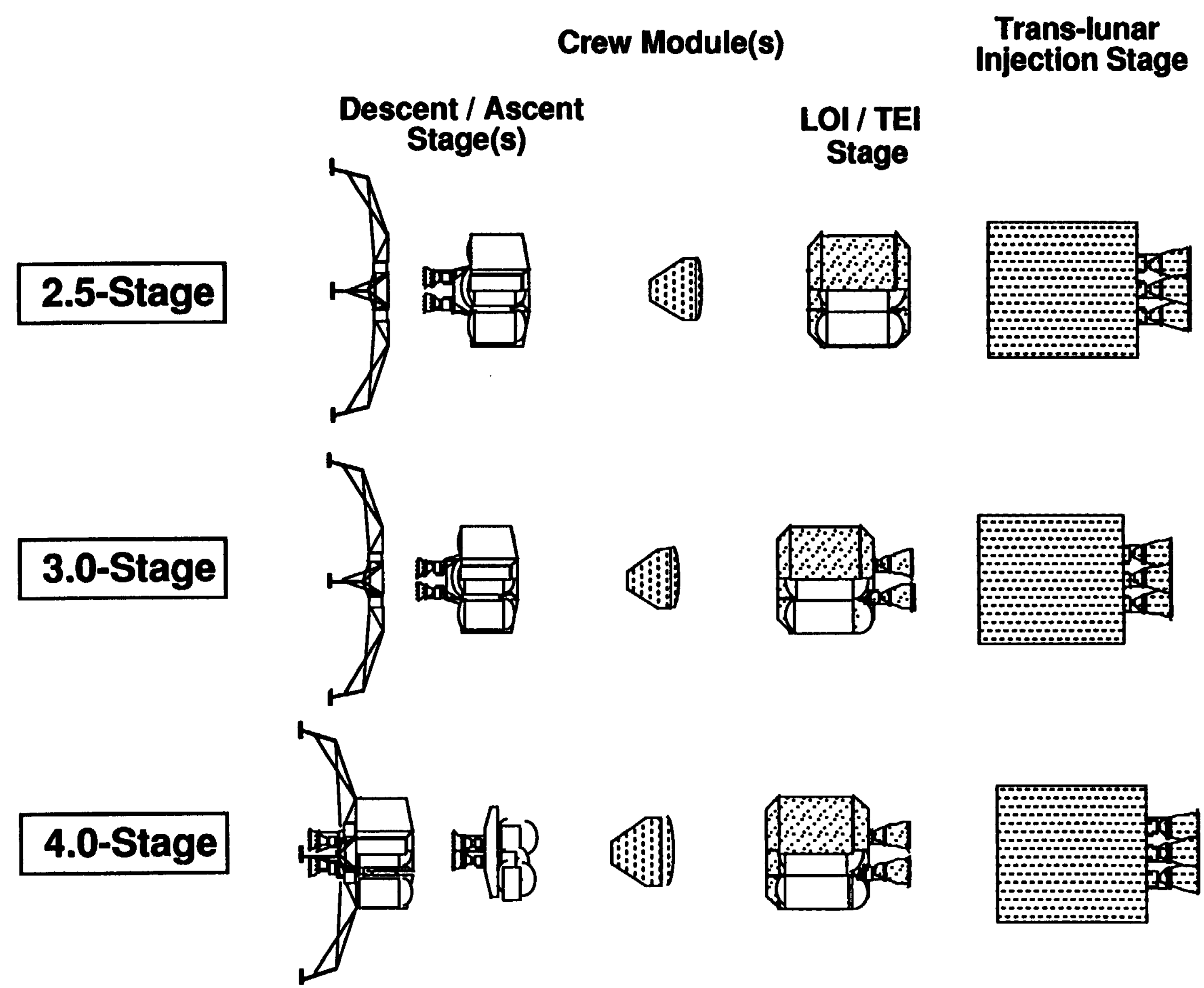
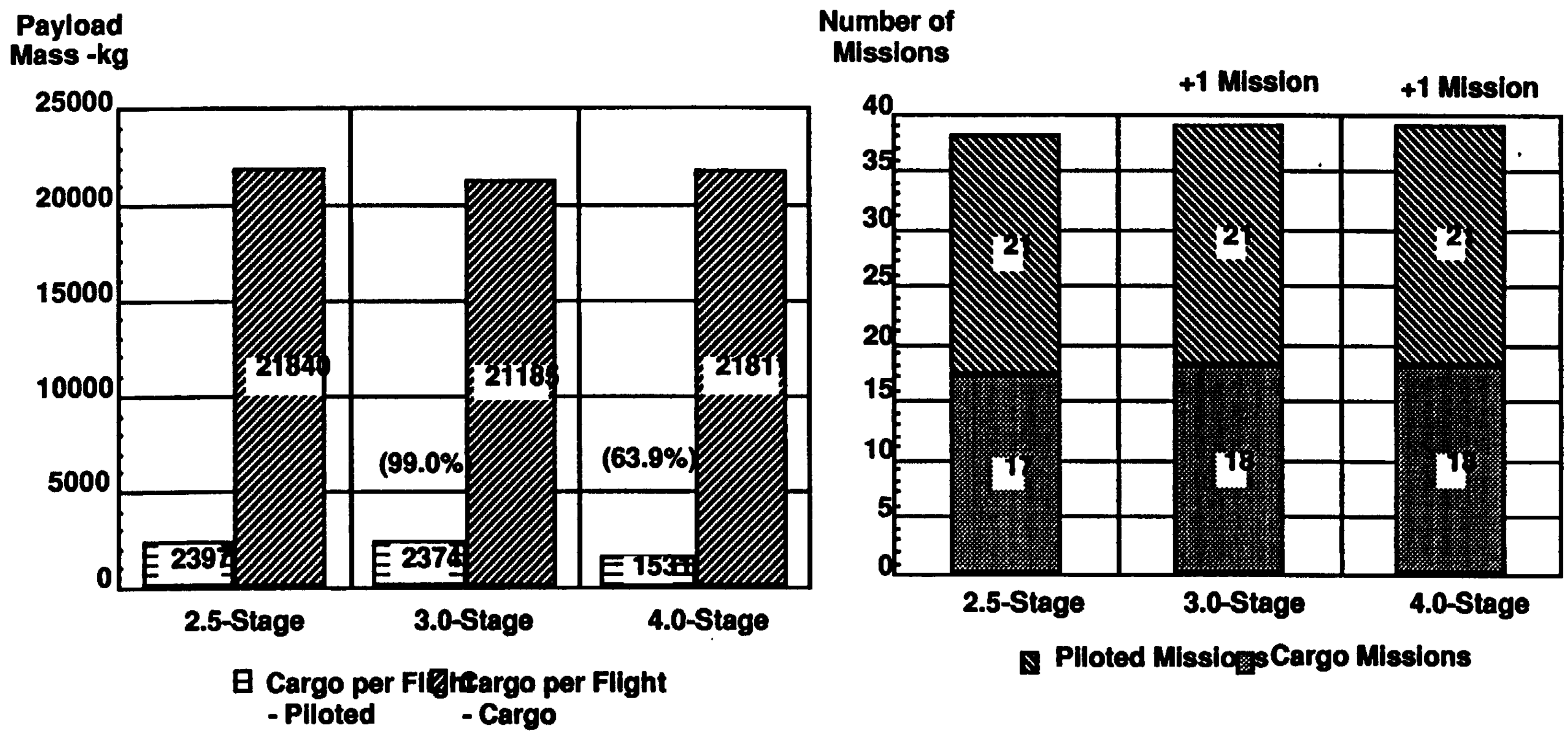


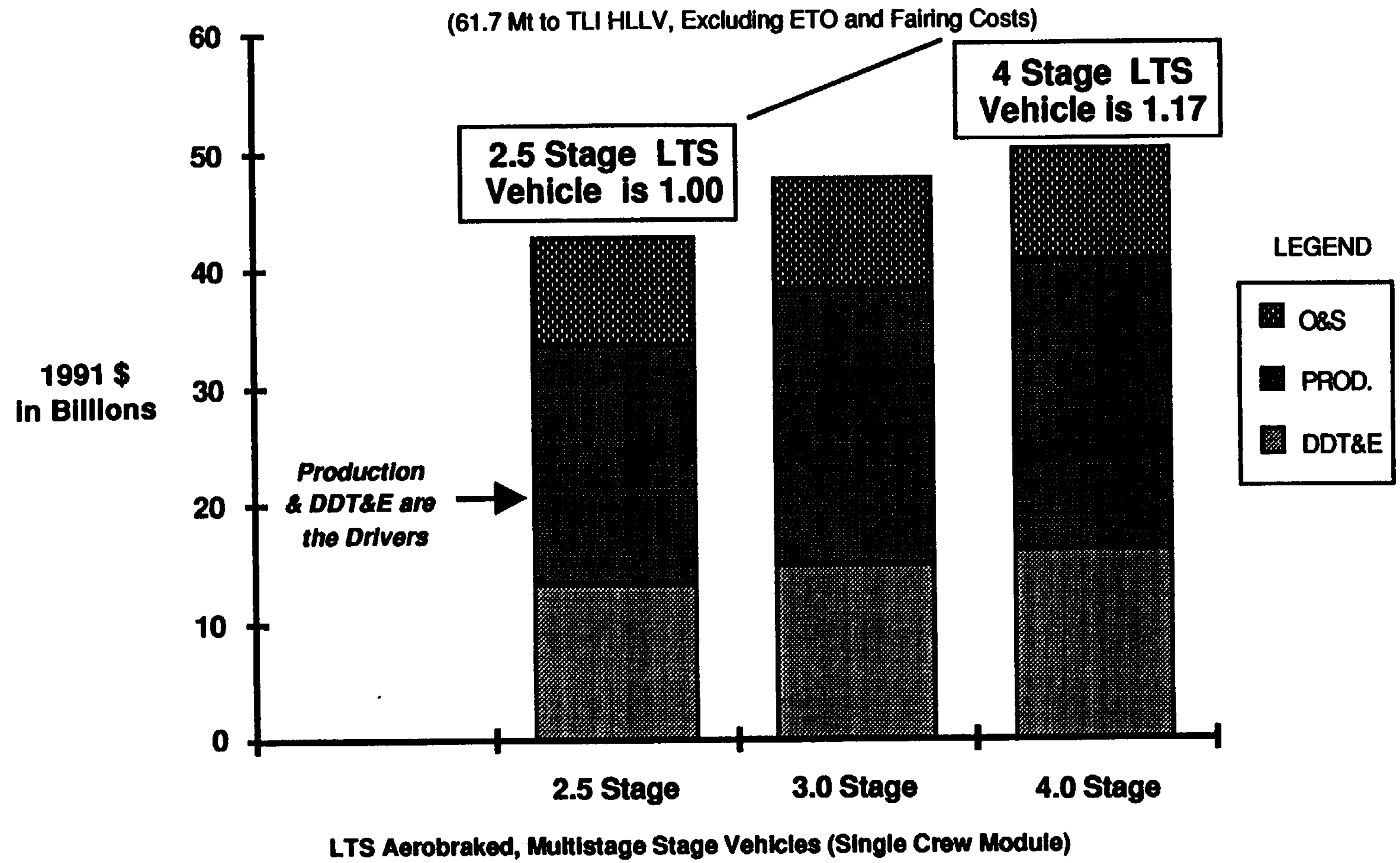
Figure 3.3.1-8 Number of Stages Options

Aerobraked, Single Crew Module Vehicles
Launch Vehicle Option #4 with PoD Shroud (Capability to TLI - 61.7 mt)



3.0- and 4.0-Stage vehicles have slightly worse performance, adding an extra cargo mission.

Figure 3.3.1-9 Number of Stages Performance Comparison

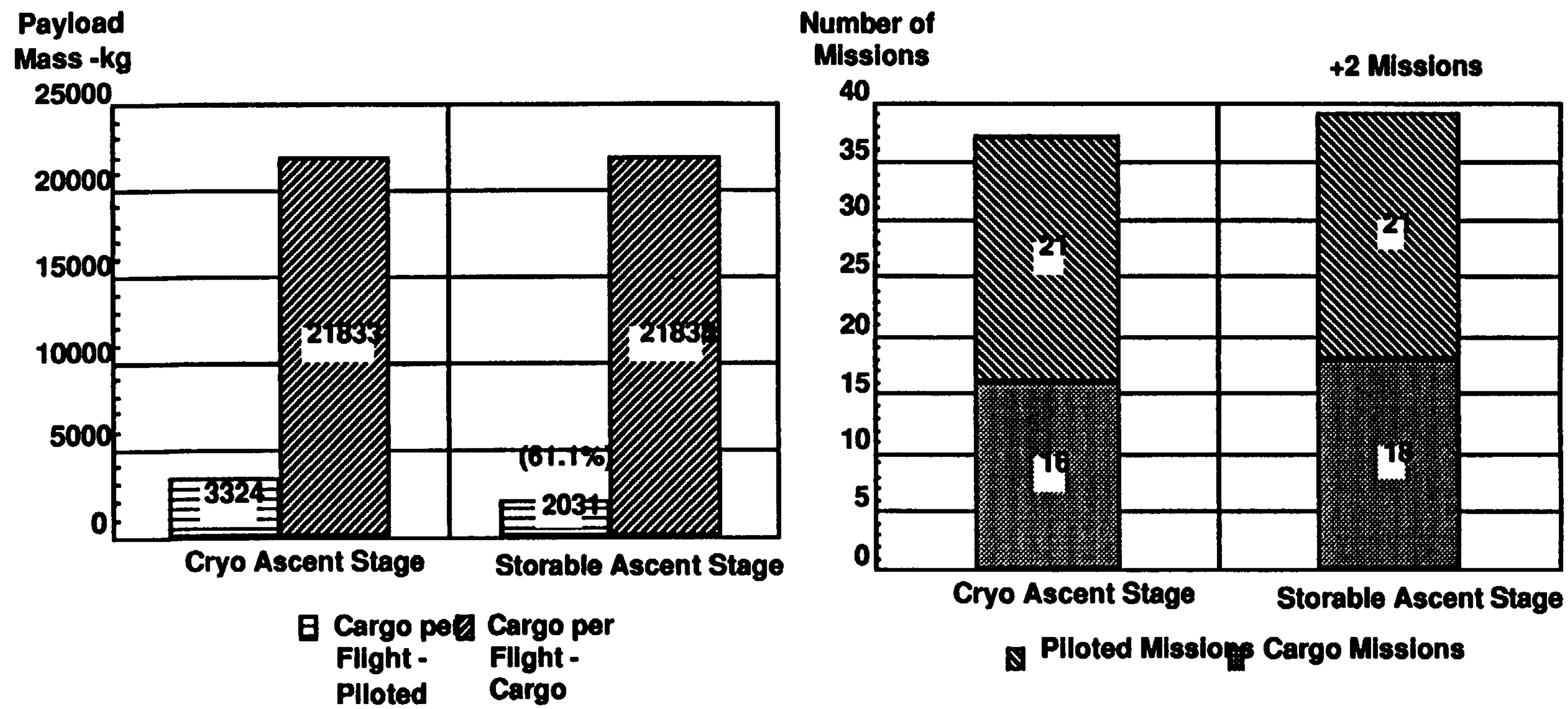


Reduced part count & integration gives 2.5 stage lower LCC (11 to 17%)

Figure 3.3.1-10 Number of Stages Cost Comparison

Aerobraked, 4.0-Stage, Dual Crew Module Vehicles

Launch Vehicle Option #4 with PoD Shroud (Capability to TLI - 61.7 mt)



A storable-propellant ascent stage vehicle requires two additional cargo missions

Figure 3.3.1-11 Ascent Propellant Performance Comparison case. With the cryo propellant vehicle, however, for an unsupported lunar staytime of 100 days, 1 extra cargo mission is required, and for 180-day stay, 2 extra cargo missions are required, as shown in Figure 3.3.1-12. From a cost standpoint, however, the cryogenic vehicle is less costly as shown in Figure 3.3.1-13, with costs ranging from 6.4% to 4.4% less than the storable concept.

Alternate Storable Vehicle Concepts (6 crew). Other vehicle concepts that make use of storable propellants included both ground-return and aerobraked return crew modules, this time designed for a crew of 6 rather than 4, in response to a Stafford Commission recommendation of more extensive use of storable propellants and a crew of six persons. These concepts were looked at only from a performance perspective, and are compared in Figure 3.3.1-14. As expected, the delivered cargo performance suffered from 5 to 10 tonnes less piloted cargo capability per mission. None of the storable propellant options were able to perform the lunar mission in a single HLV launch, given the large crew modules and vehicle sizes.

Single vs. dual crew modules. For a lunar mission utilizing lunar orbit rendezvous, non-essential crew module mass may be left in LLO during the surface staytime in order to minimize mass to the lunar surface. In this dual crew module case, a transfer crew module is used for transfer to and from the moon, and an excursion crew module is used for the lunar surface excursion. Given the design groundrules for this study phase, the excursion crew module was assumed to be expended after each mission. If the design does not utilize lunar orbit rendezvous, only a combined or single crew module may be used for the entire mission.

In a dual crew module mode, the crew has the combined volumes of both a transfer and excursion crew module during the transfer to the moon, and the excursion crew module provides a redundant system in the case of failure. On the return leg, however, only the transfer crew module is returned, so volume is more limited and no backup crew module is available.

A single or combined crew module option gives the crew added volume throughout the mission if sized for the entire mission duration, but also adds additional mass to the descent and ascent phases of the mission, due to additional TPS, consumable storage, and radiation shielding.

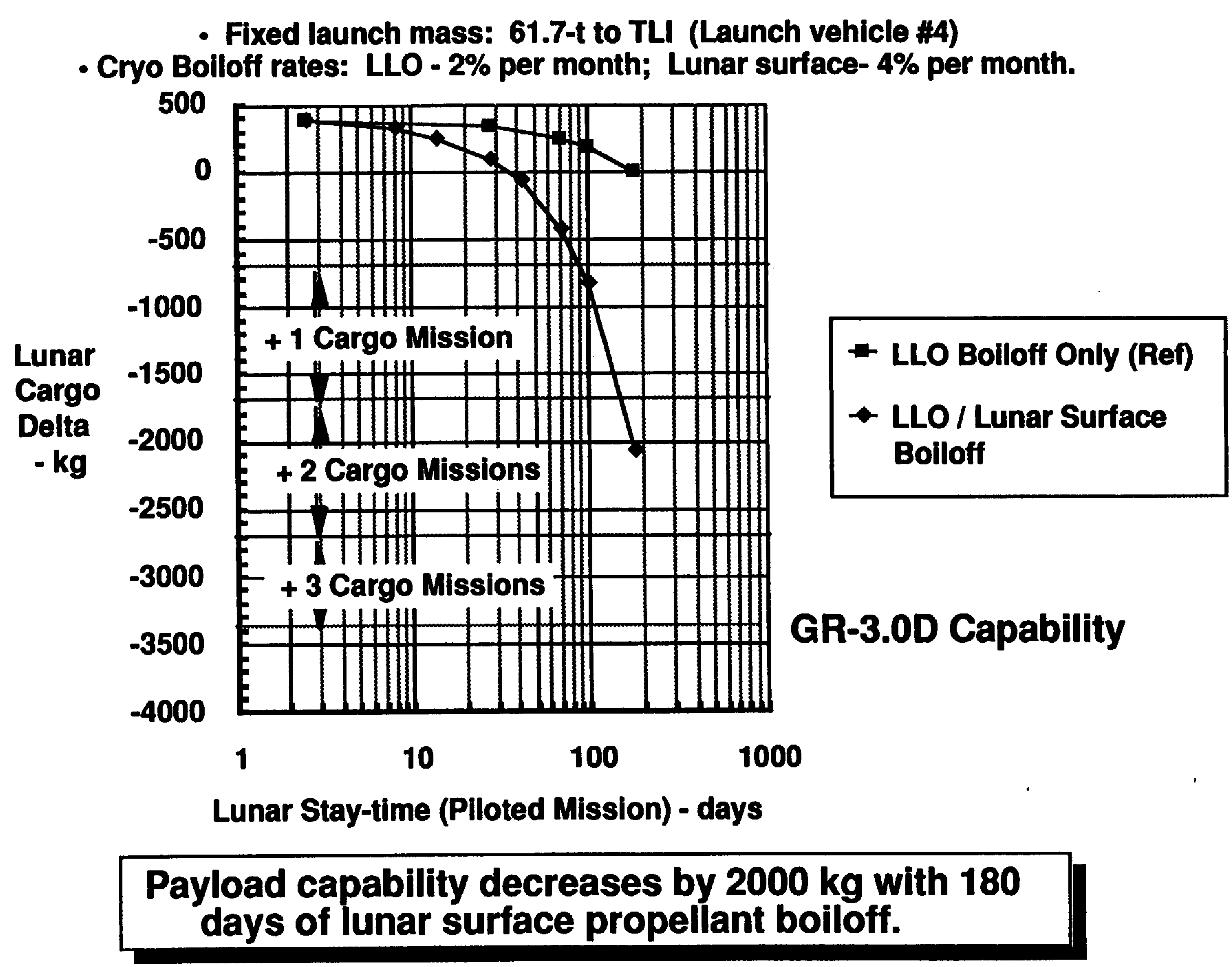
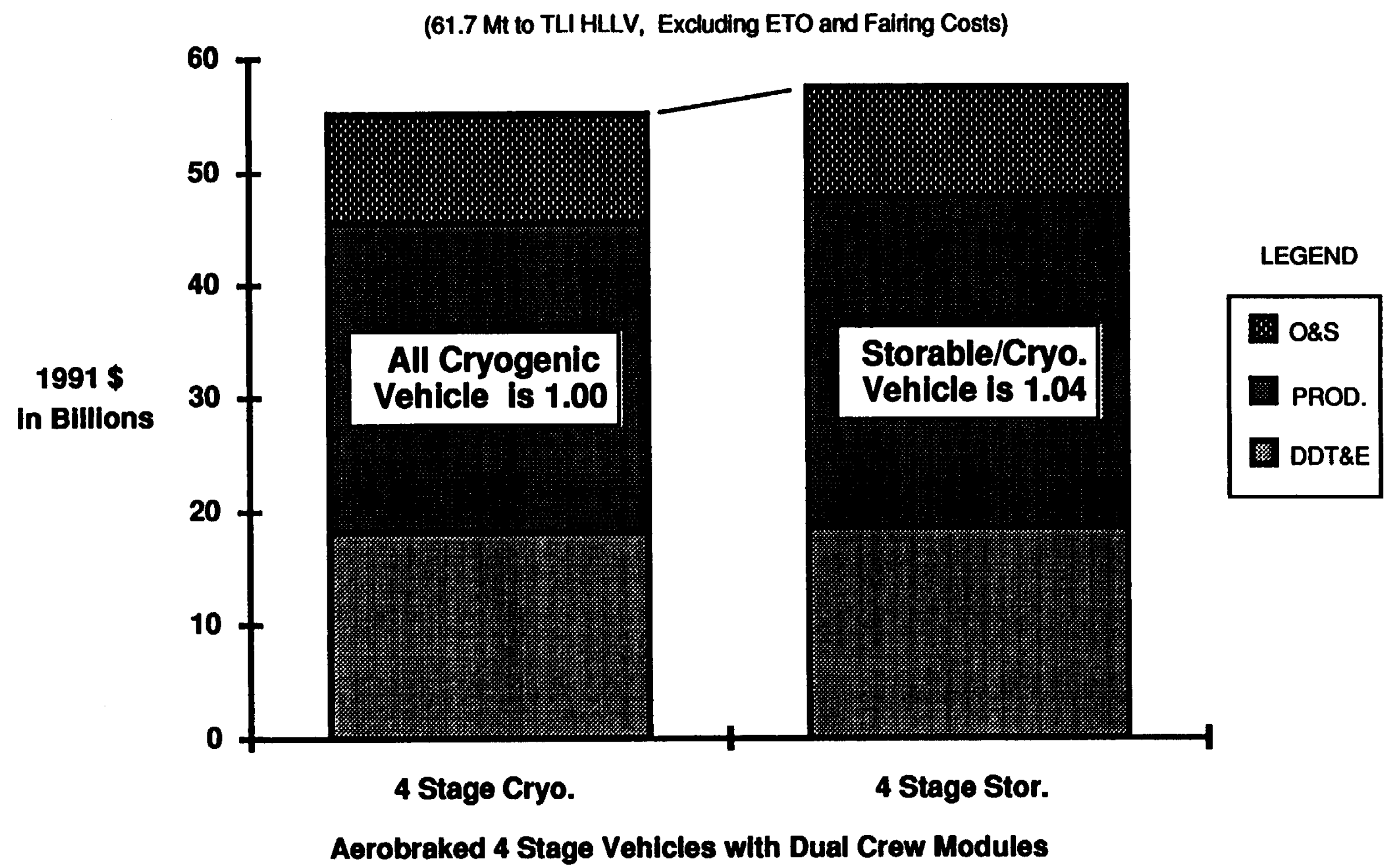


Figure 3.3.1-12 Vehicle Performance Sensitivity to Lunar Staytime



Difference in ascent propellants choice small (4.4%).

Figure 3.3.1-13 Ascent Propellant Cost Comparison

- **Six crew members, nominal six-month lunar stay**
- **Isp: RL10-IIB - 460 sec; Storable - 342 sec.**

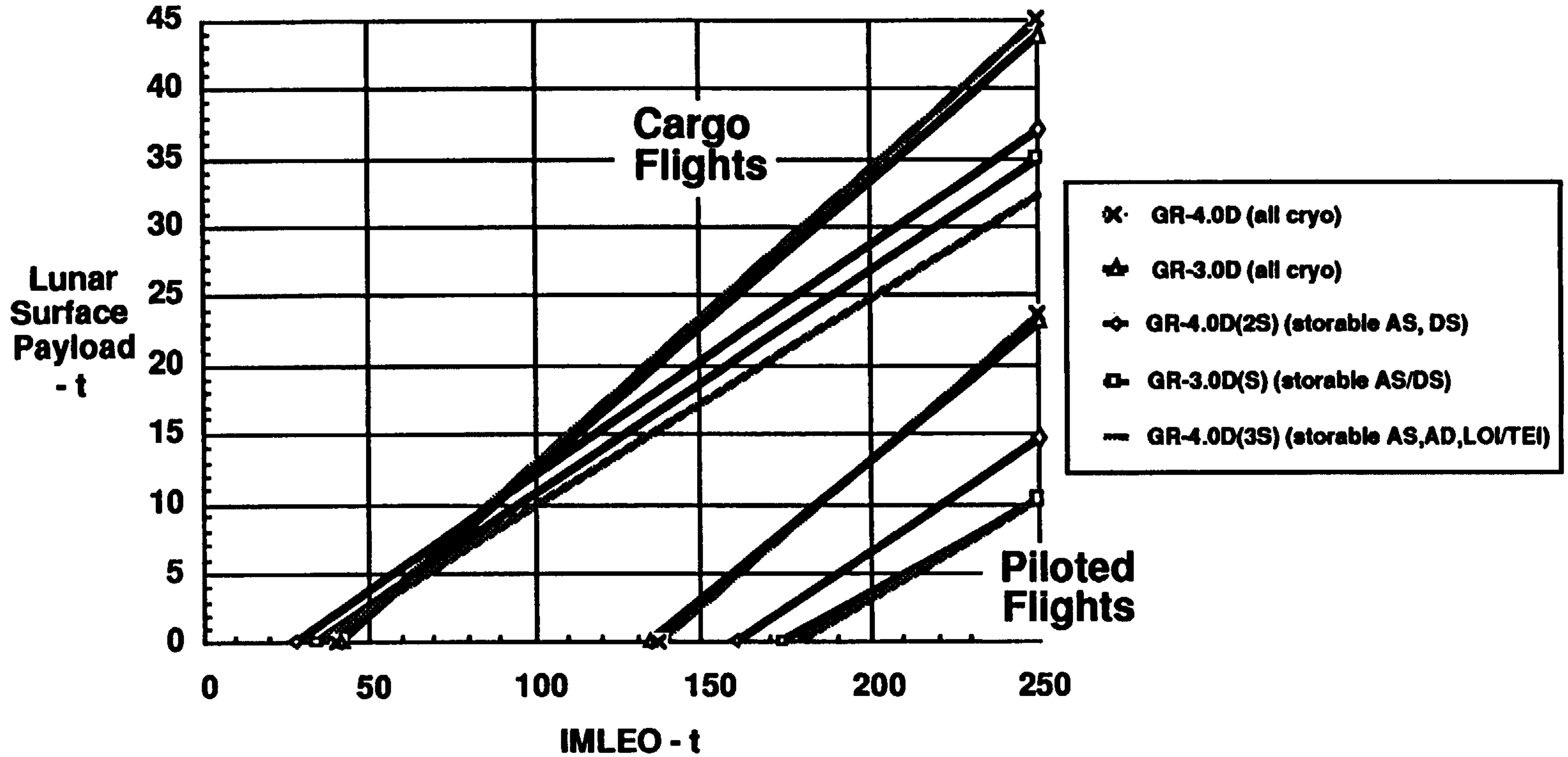


Figure 3.3.1-14 Alternate Storable Staging Concepts

From a mission performance perspective, the dual Crew Module is favored as shown in Figure 3.3.1-15, resulting in two less cargo missions over the mission model. The mass to the lunar surface is minimized, outweighing the penalty of additional mass transferred to the moon.

From a cost perspective, however, the single crew module is favored as shown in Figure 3.3.1-16, with 9.8% lower LCC. The additional module increases the hardware development cost, and adds compexity to the checkout and test development cycle. The expended excursion crew modules also add significantly to the production costs.

Lessons Learned The lessons learned from the single-launch Architecture Study are given in Figure 3.3.1-17.

3.3.2. Crew Module Concepts

The U.S. Space program has a history of manned spaceflight extending back 30 years. Each program has had a specific purpose and destination, driving crew module size, mission life, and crew size. Generally, the more extensive the mission purpose is, the larger and more accommodating the crew module is.

Recent STV studies have focused on a return to the moon, but with the purpose of going to stay. Phase 2 of the STV study focused more on limiting mass, and so crew module designs were reduced in size and accommodations from the Phase 1 crew module designs. Through the first part of this phase of study, the crew module concepts shown in Figure 3.3.2-1 were used as a point of departure to exercise subsystem trades and sensitivities. All concepts were designed for a crew of four.

Subsystem design and component distribution in the crew modules are affected by vehicle functions during the mission. Because the vehicles are used in both piloted and cargo modes and in the interest of minimizing crew module mass, some subsystems such as power and thermal rejection can be located on the stage to minimize duplication. The crew module, however, must also function for a short time on its own, during reentry or aeromaneuver, and so must have required subsystems on board. An example of the functional split between stage and crew module subsystems is shown in Figure 3.3.2-2.

Aerobraked, 3.0-Stage Vehicles

Launch Vehicle Option #4 with PoD Shroud (Capability to TLI - 61.7 mt)

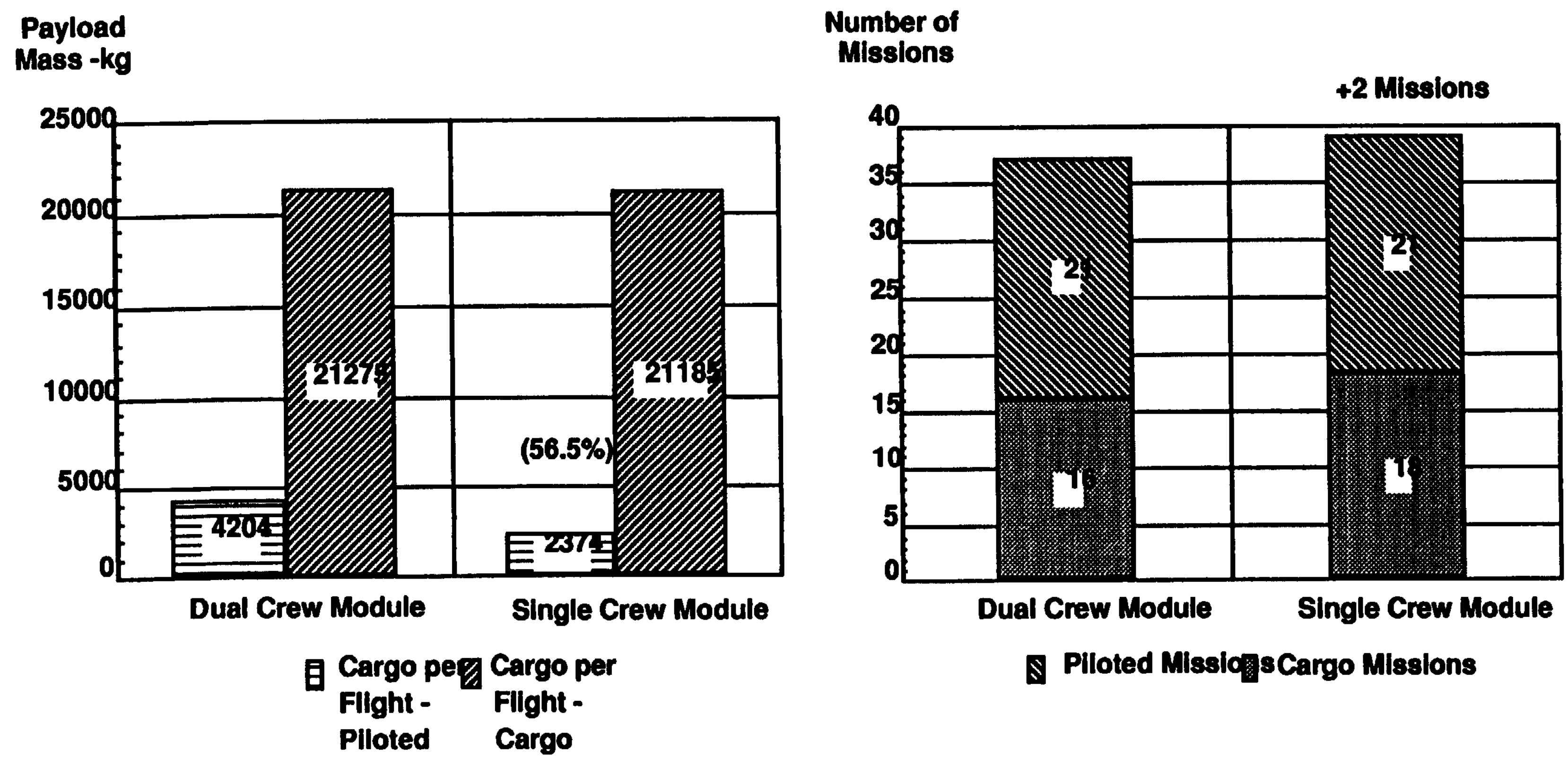
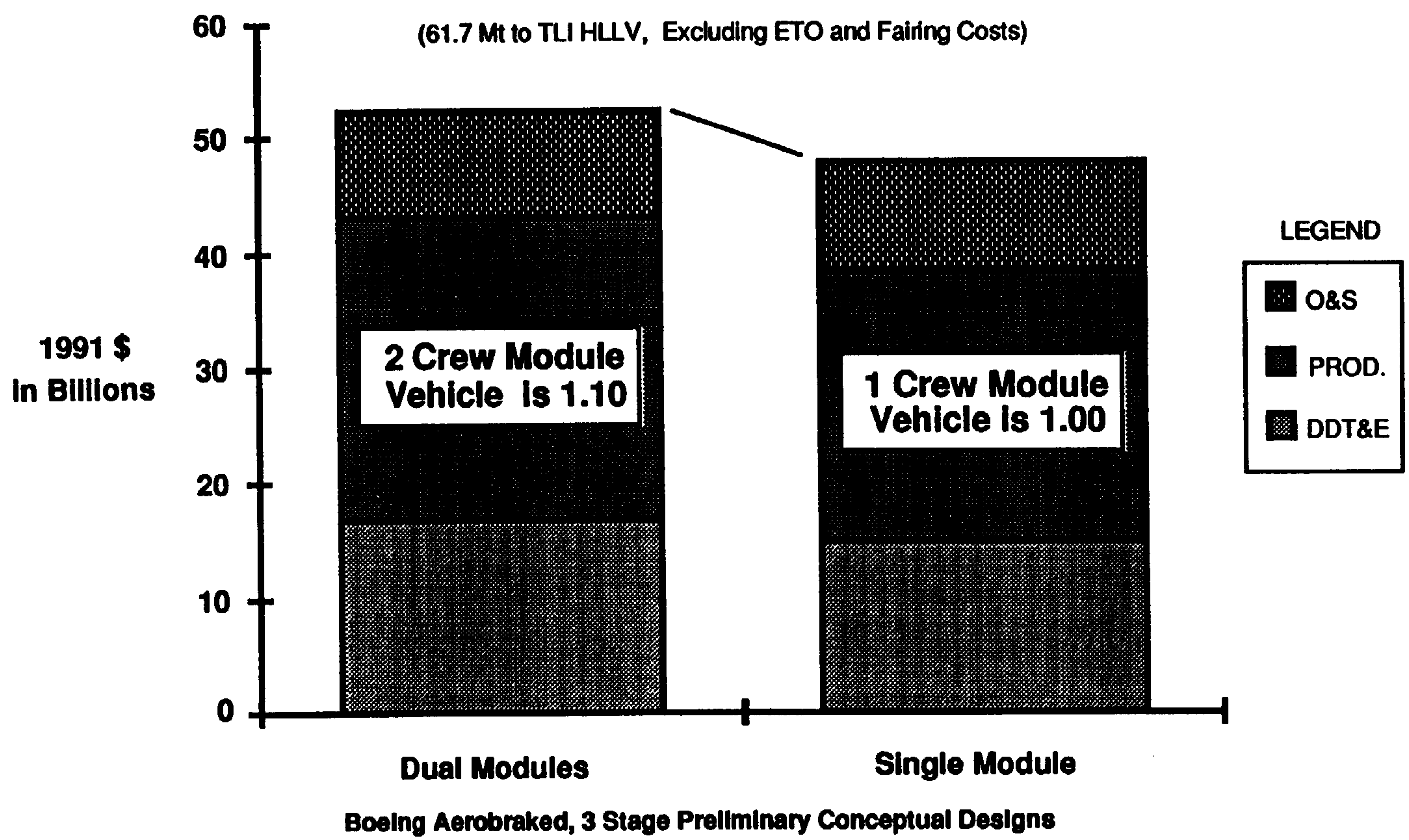


Figure 3.3.1-15 Number of Crew Modules Performance Comparison



Single crew module has lower LCC & less expended hardware (9.8%)

Figure 3.3.1-16 Number of Crew Modules Cost Comparison

Trade winner:	Δ Effect:
• HLV crew launch using a launch escape system beats STS launch	1-8 %
• Ground Return of the transfer cab beats LEO rendezvous with STS	1-6 %
• Lowest number of stages wins (2.5 for aerobkd, 3 for ground rtn)	11-17%
• Develop only 1 propellant system (all cryo), not storable and cryo	1-4%
• Single crew modules beats Dual modules where performance allows	10%

Note: these trades were performed with "apples to apples" comparisons (ie. 3 stage ground return vs. 3 stage aerobraked) There will be crossover effects between trades because of the varying "effect" of the trade. This will affect eventual "best" configuration <u>for this set of launch conditions</u>

Figure 3.3.1-17 Single Launch Architecture Study Lessons Learned

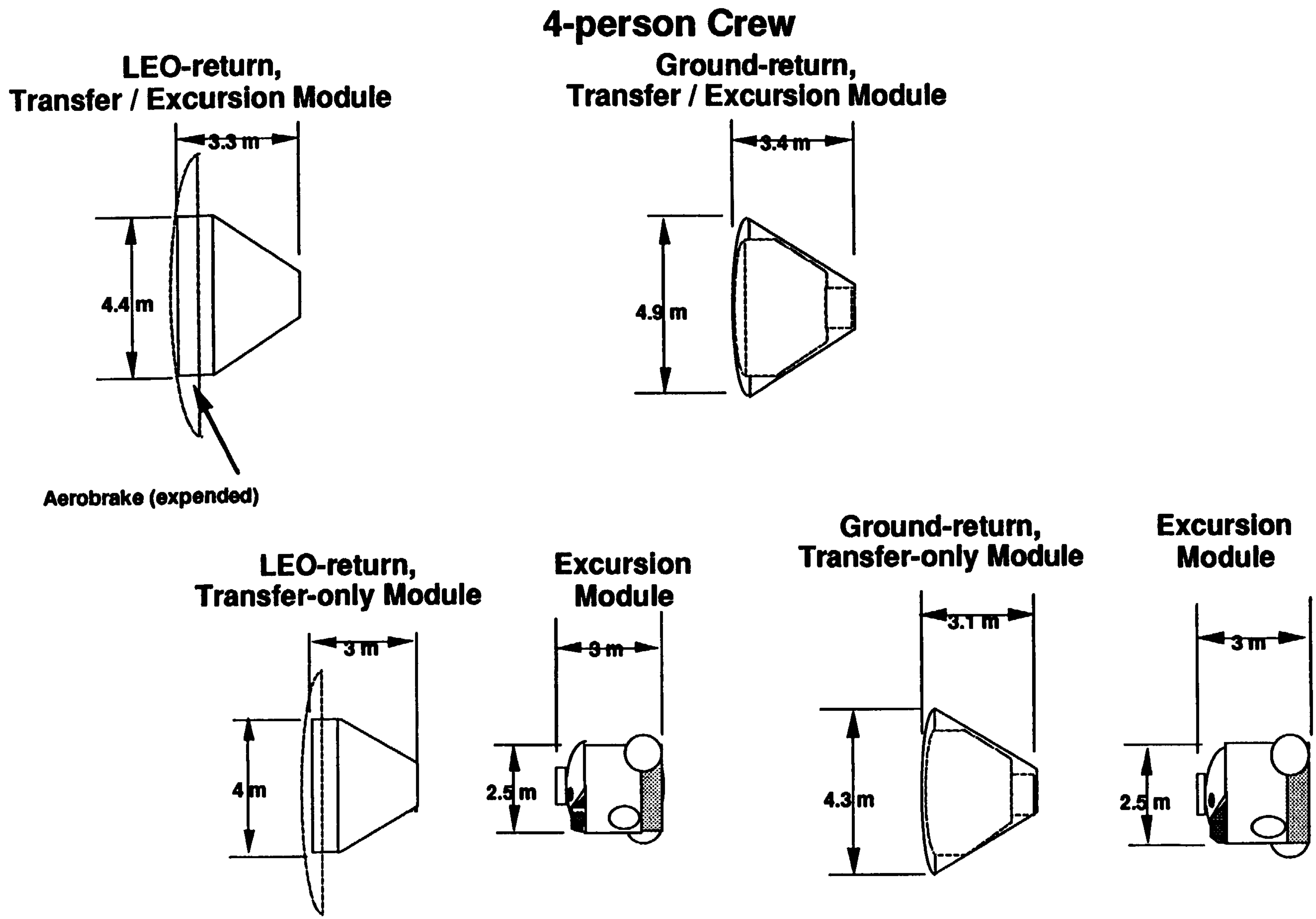


Figure 3.3.2-1 Phase 2 Crew Module Concepts

Objective: Minimize crew module subsystems (minimum reentry or aeromaneuver mass), while maintaining necessary functions and avoiding subsystem duplication in a vehicle.

Support Stage

- RCS - Orbital
- Main Propulsion System
- Power - fuel cells / solar power supply and distribution panels
- G, N, & C - orbital
- Propulsion Controllers
- Stage Instrumentation
- Comm & data - orbital
- Heat rejection - Freon loop, radiators
- Metabolic O2 (in supercritical storage tanks)

Crew Module

- TPS - Reentry / Aeromaneuver
- Exterior Insulation
- RCS - earth-return
- Power - batteries for earth return and distribution wiring
- G, N, & C - earth return.
- Controllers - RCS & mechansims (earth return)
- Communication antennas
- ECLSS Atmosphere repress - gaseous storage tanks; heat rejection - freon loop and water boilers
- Metabolic water storage

Pressurized Volume

- Internal partitions, bulkheads
- Interior insulation
- Radiation Protection provisions
- Power distribution panels and wiring
- Displays and control equipment
- Communication, data handling equipment
- ECLSS atmosphere revitalization; thermal water loop; food, water, and waste management; fire detection/ suppression
- Crew Accommodations

Figure 3.3.2-2 Transfer Crew Module Subsystem Split

Crew module concepts The dual crew module designs each provide 1.98 m^3 habitable volume per person, similar to what was provided on Apollo. On the transfer to the moon, the combined volume of 3.96 m^3 is available as habitable volume. The single-crew module designs are sized to provide 2.83 m^3 of habitable volume per person, approximately 40% more than the Apollo command module provided, due to the longer inhabited duration.

The transfer crew modules are similar in shape to the Apollo command module. The ground-return module requires an ablative thermal protection system and extra structural integrity to survive earth reentry and landing. The aerobraked module is attached to an expendable aerobrake and is recovered from LEO by the Orbiter. Dual hatches allow for launch pad ingress/egress and orbital EVA, as well as orbital docking ingress / egress. Windows provide viewing during docking procedures.

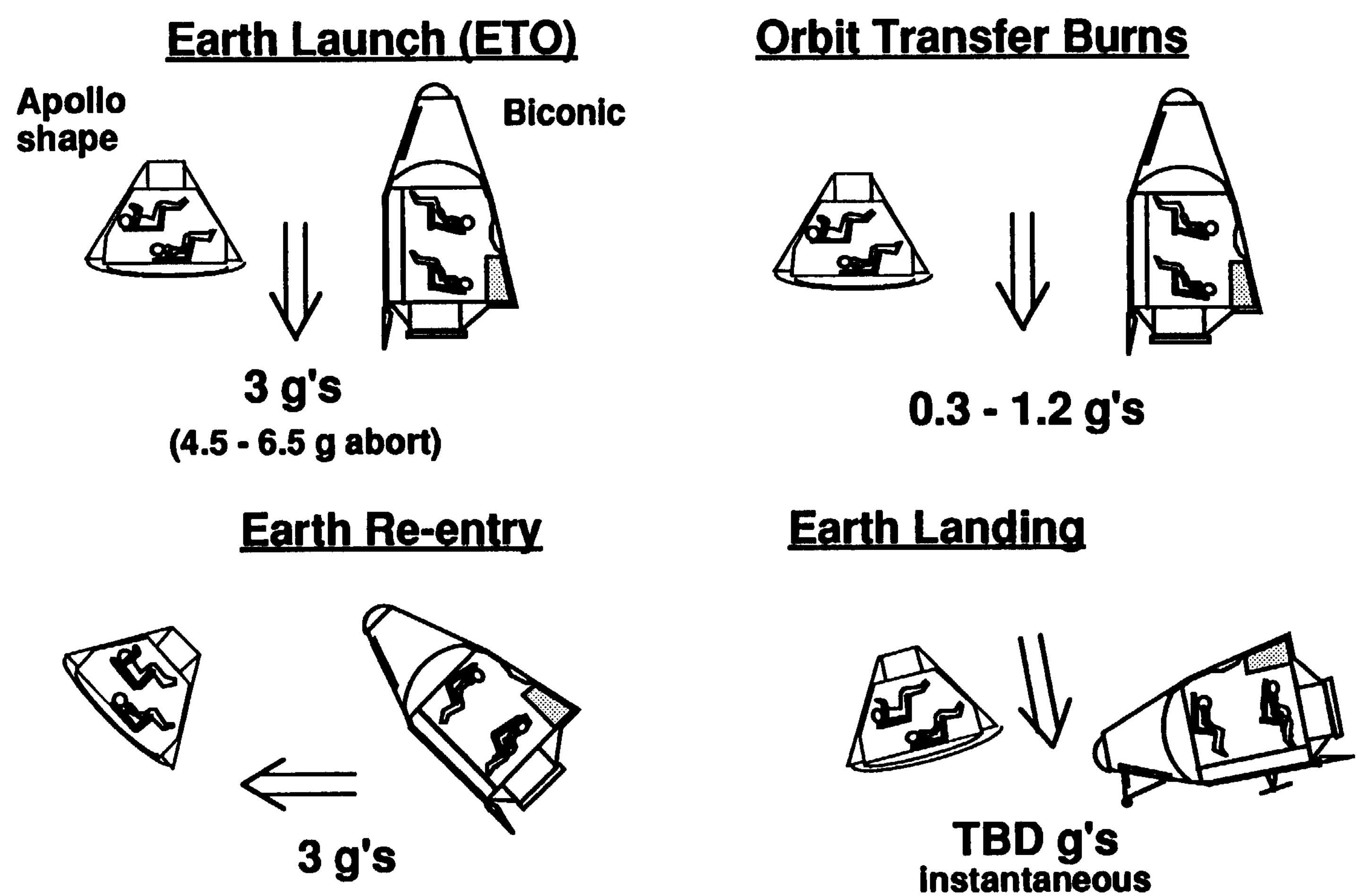
The excursion crew modules are cylindrically-shaped modules that are sized for four crewmembers. Dual hatches allow for lunar surface ingress/egress, as well as orbital docking ingress/egress. Windows provide viewing during lunar landing.

Launch / Return If the crew is launched aboard the LTS vehicle, provisions must be made for crew safety in the case of launch aborts. A launch escape system similar to that used on Apollo could be provided in the case of launch vehicle failure up to 300,000 ft. Beyond that point, the LTS stage could provide thrust for an abort to orbit.

One issue that impacts the aerobraked crew module design is launch abort. Because the aerobrake is not suited for abort reentry, the crew module would need additional thermal protection as well as added structural stiffening for recovery.

In regard to crew orientation, with an Apollo-style capsule, no significant impacts are made on the crew module design. No repositioning of crew restraints is needed, unless the crew module is also used for lunar landing, in which case the crew must be positioned for a low-g lunar landing. With a biconic shape, the crew restraints would either be repositioned, or the crew module would need to be inverted for launch and transfer, as shown in Figure 3.3.2-3.

Lunar staytime Increased lunar staytime impacts various crew module subsystems, including pressurization and atmosphere revitalization, CO2 removal, heat rejection, power supply, and personnel provisions.



Apollo shape allows single crew orientation for launch, transfer, and recovery; a biconic shape requires reorientation of crew restraints or of the crew module for launch and transfer.

Figure 3.3.2-3 Crew Orientation Comparison

Internal Pressure - The influence of crew module internal pressure was studied, based on a JSC study "Internal Atmospheric Pressure and Composition for Planet Surface Habitats and Extravehicular Mobility Units", May 13, 1991. This gave a broad approach to choosing optimal habitat internal pressures based on known interfaces and mission purpose. Preliminary findings indicate that an internal crew module pressure of 10.2 psi with 30% O_2 provides good flexibility in operating between Earth atmosphere, lunar transfer modules, lunar habitat, and EMU. Reducing the crew module pressure from 14.7 psi to 10.2 psi only reduces excursion crew module mass by 4%, and transfer module mass by less than 1%.

CO_2 Removal - Several CO_2-removal technologies are available as viable options for use on a lunar LTS. Currently, the most important factors in deciding which technolgy to use, based on mission duration and crew size, are weight and volume. Other factors that need to be accounted for, though, include power usage, heat rejection, consumable losses, and required servicing time. The CO_2 removal technologies considered in this trade are given in Figure 3.3.2-4, as well as a list of systems used on previous and currently-designed vehicles. Other than the LiOH system, all technologies shown are regenerable systems, providing CO_2 adsorption and desorption with alternating adsorption beds.

From Figure 3.3.2-5, a Lithium Hydroxide system similar to that used on the Shuttle has least mass, given a crew of four on board for less than two weeks. If the mission duration extends beyond 4 weeks, a regenerable Solid Amine, water-desorbed CO_2 removal system with LiOH backup becomes mass-effective. However, other factors such as power requirements, heat rejection, and consumable losses need to be taken into account in determining the minimum mass subsystem. Based on just the subsystem volume, the best system for the given crew size and mission is a Lithium Hydroxide system. If the mission duration extends beyond 3 weeks, a regenerable solid amine, vacuum-desorbed system or molecular sieve system (with LiOH backup in each case) become volume-effective, as shown in Figure 3.3.2-6.

Heat Rejection - In Figure 3.3.2-7, a preliminary analysis of heat rejection options shows radiators to be mass-effective beyond 1 day on the lunar surface or in transit, given the assumptions shown. Other issues that need to be addressed further include protection from solar heating during the lunar day, and the thermal contribution of

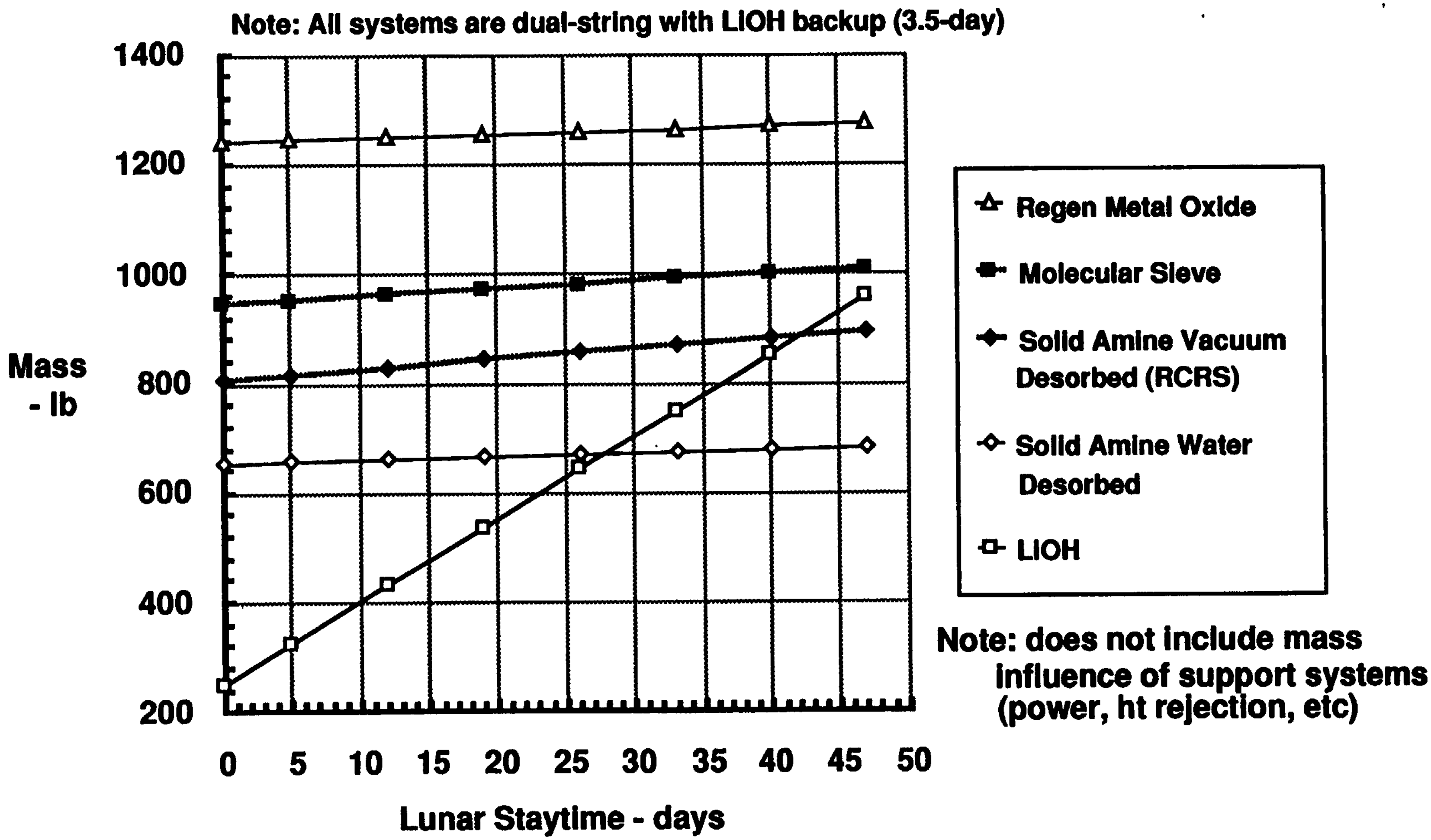
- Lithium Hydroxide - CO_2 adsorbed by expendable cartridges containing LiOH. Moisture and heat are process byproducts.
- Molecular Sieve - CO_2 is alternately adsorbed and desorbed to vacuum by two Type 5A molecular sieve beds. While one bed is desorbing, the other is adsorbing. Space vacuum and heat are used to remove CO_2 during bed regeneration.
- Solid Amine Water Desorbed - a two-bed cyclic adsorption system using steam to drive off CO_2. Moisture and heat are process byproducts.
- Solid Amine Vacuum Desorbed - a two-bed cyclic adsorption system using space vacuum to remove CO_2.
- Regenerable Metal Oxide - a two-bed cyclic adsorption system designed for EVA using space vacuum to remove CO_2.

Background:

- Apollo: LiOH CO_2 removal in all flight elements - 3 crew, 12.5 days.
- Skylab: 2-canister molecular sieve CO_2 removal - 3 crew, 84 day duration.
- STS Orbiter: LiOH CO_2 removal - 7 crew, 7 days.
- Extended Duration Orbiter: Solid Amine Vacuum Desorbed RCRS with LiOH as backup - 7 crew, 16-28 days.
- SSF: Molecular sieve CO_2 removal.

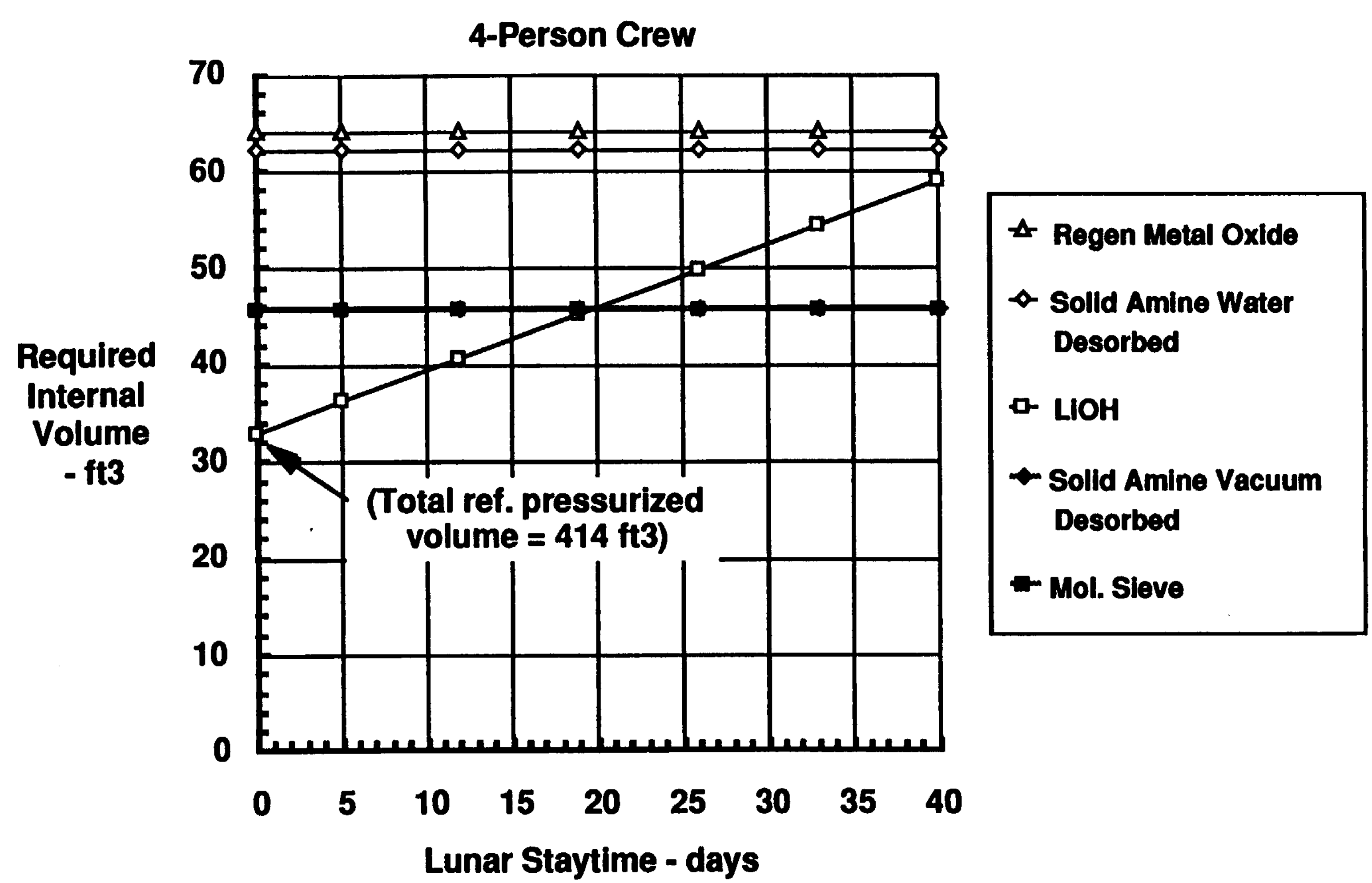
Figure 3.3.2-4 CO2 Removal System Options

- **Sized for crew of four**



Solid Amine Water-Desorbed system becomes mass-effective at 28 days with a crew of four.

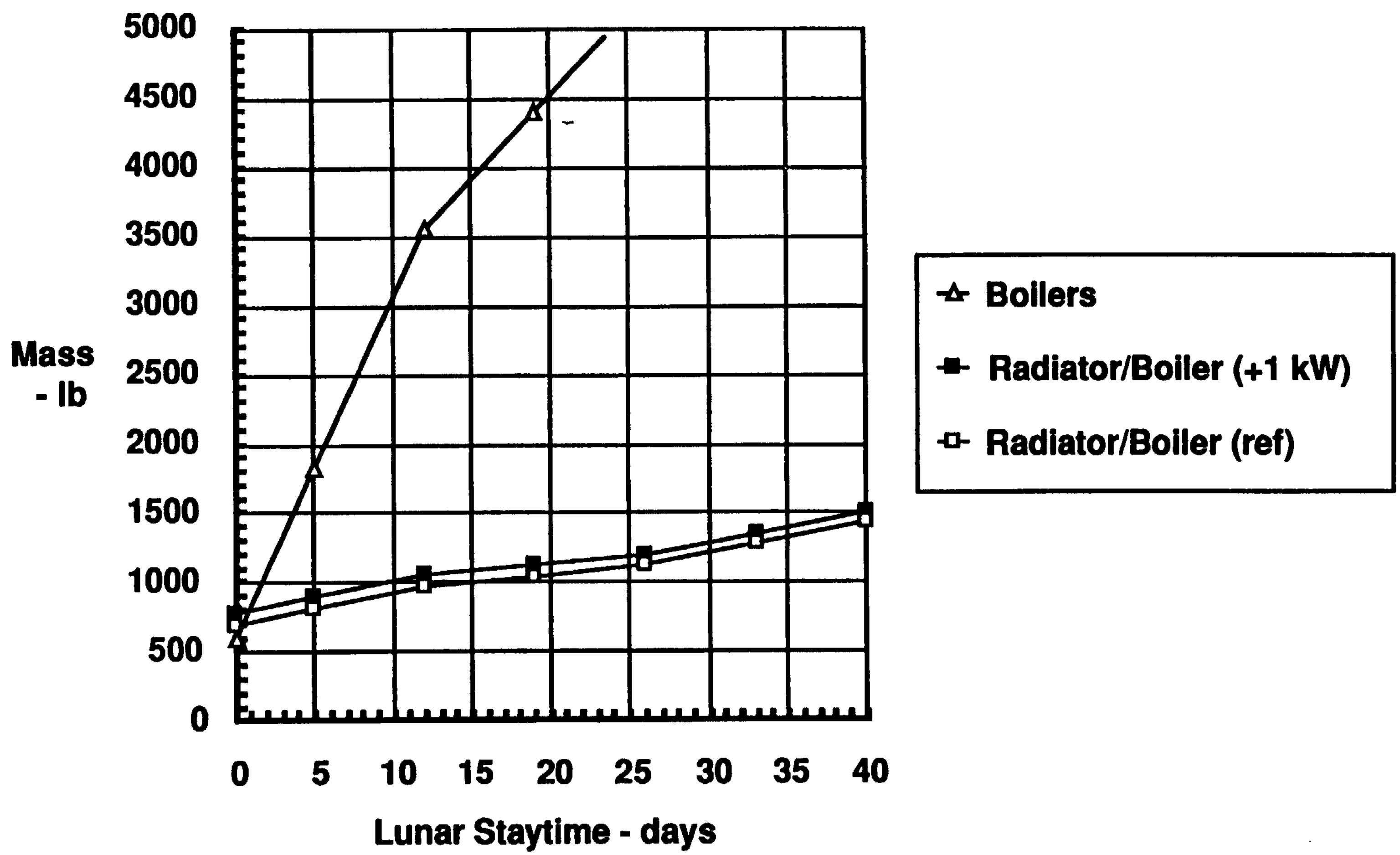
Figure 3.3.2-5 CO2 Removal System Mass Comparison



Volume crossover occurs at 20 days, with molecular sieve and SAVD systems requiring similar internal volume.

Figure 3.3.2-6 CO2 Removal System Volume Comparison

• Sized for 4 crew, lunar excursion vehicle



Radiators show significant heat rejection improvement over water boilers after only 1 day

Figure 3.3.2-7 Heat Rejection System Mass Comparison stored cryogenic propellants. Another issue is the placement of radiators on the lunar excursion vehicle, and method of deployment and use.

Power Supply - Factors that affect power system selction include the average and peak power levels, the duration of use, and for regenerable power systems, the duration between storage and discharge cycles. On the lunar surface, the storage/discharge cycle is every 14 days, so any regenerable system would need a 14-day storage capacity. In this case, batteries as a storage medium become prohibitively heavy, as do gas storage bottles for fuel cell reactants.

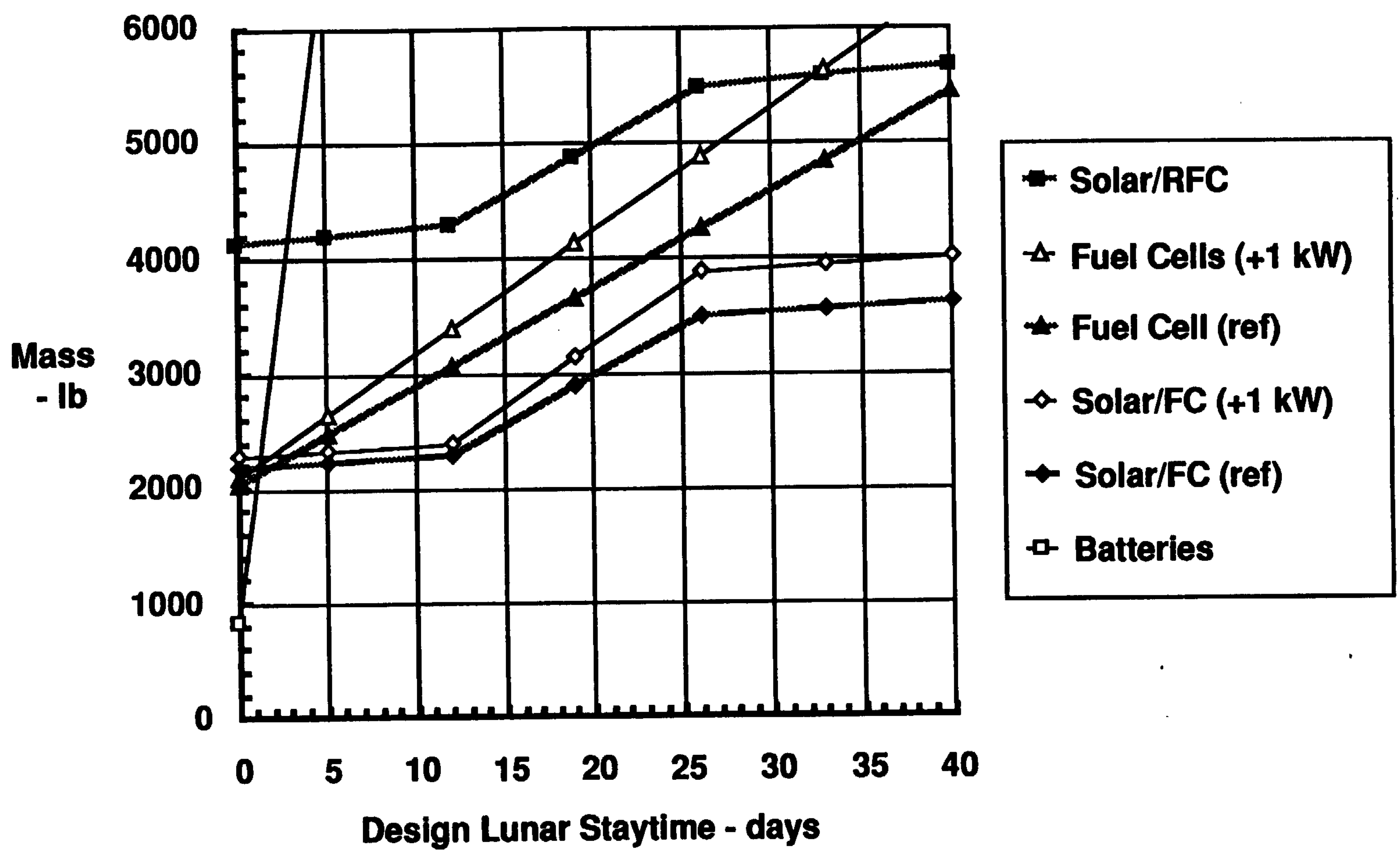
As shown in Figure 3.3.2-8, high power-density batteries, such as expendable LiSOCl batteries, are mass-effective for very short-duration use up to one day. For the current LTS excursion vehicle, with a self-powered lunar surface stay of 2 days, fuel cells remain the preferred option. Beyond two days, however, the addition of solar arrays as a lunar-day power source is the most mass-effective option. Solar arrays are assumed to be lightweight, hand-deployed arrays left on the lunar surface.

Combined ECLSS - Subsystem trade results given to this point were based solely on the given subsystem. In fact, the subsystems interact as shown in Figure 3.3.2-9, so any subsystem trade must include the interaction of other subsystems. For example, water from the fuel cell byproduct can be used as drinking water, as well as for coolant in the water boilers. Some CO_2 removal systems also reject excess heat and water vapor back into the cabin, and may require more power for operation. The interaction of these subsystems can affect the outcome of a particular subsystem trade.

Figure 3.3.2-10 shows the combined subsystems mass impact as mission duration increases. The reference system of fuel cell, LiOH CO_2 removal, and water boiler has minimum mass up to one day of support duration. From 1 to 5 days, added radiators give the minimum system, and from 5 to 40 days, added solar arrays yield the minimum-mass system. For a design for support beyond 40 days, replacement of the LiOH system with a solid amine, vacuum-desorbed system is mass-effective. The solid-amine, water-desorbed system is heavier due to water, power, and heat-rejection requirements.

Personnel Provisions - Personnel provision changes for extended mission duration may include a solar storm shelter, galley, health kit, exercise equipment, recreation/entertainment equipment, commode, sleep stations. As shown in Figure

• 4 kW Nominal Power level, 4 Crew, Lunar Excursion Vehicle



Fuel Cells with Solar arrays (for lunar day) become mass-effective after only 2 days

Figure 3.3.2-8 Power Supply System Mass Comparison

Subsystem trade results are based on the interaction of the integrated subsystems.

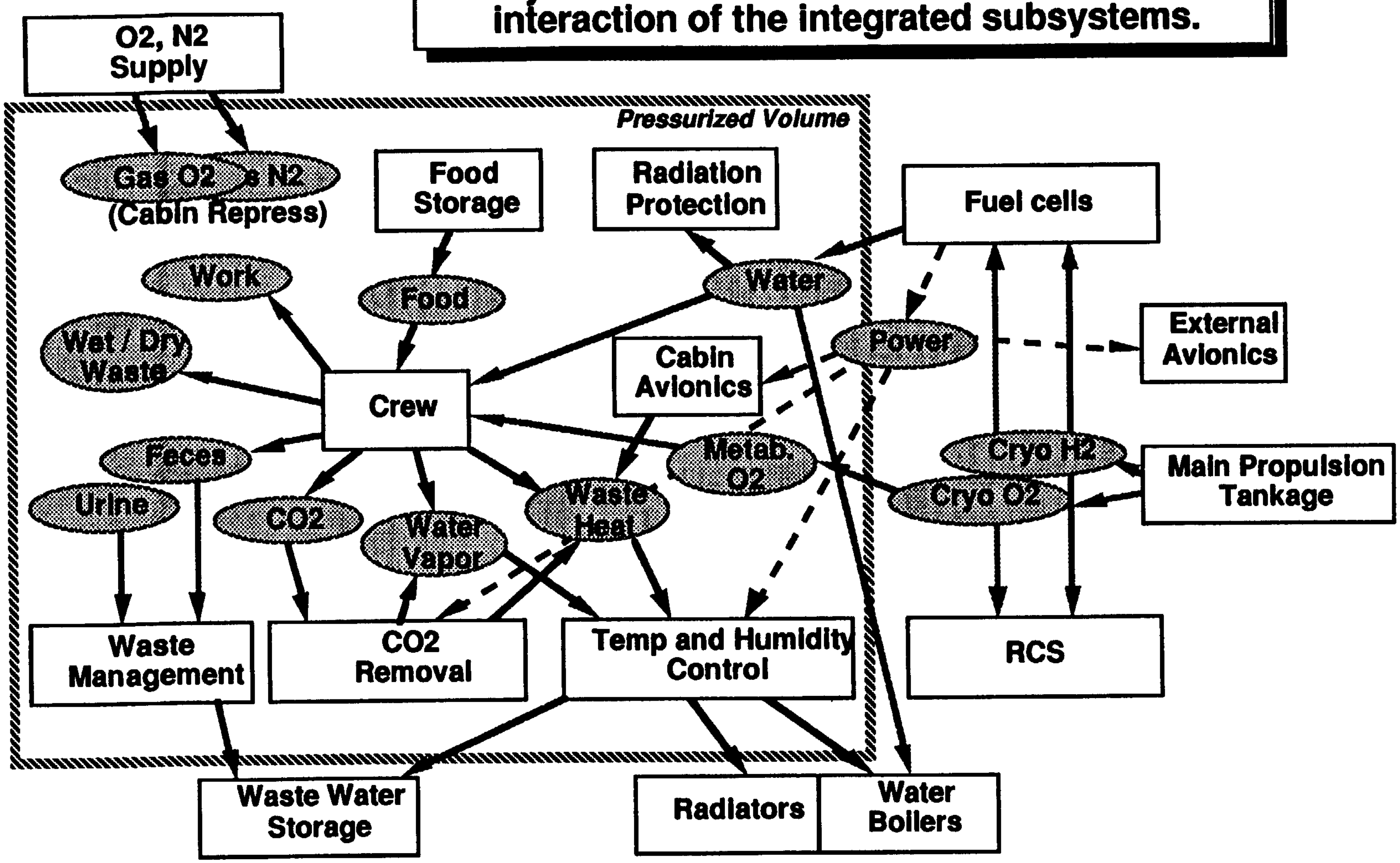
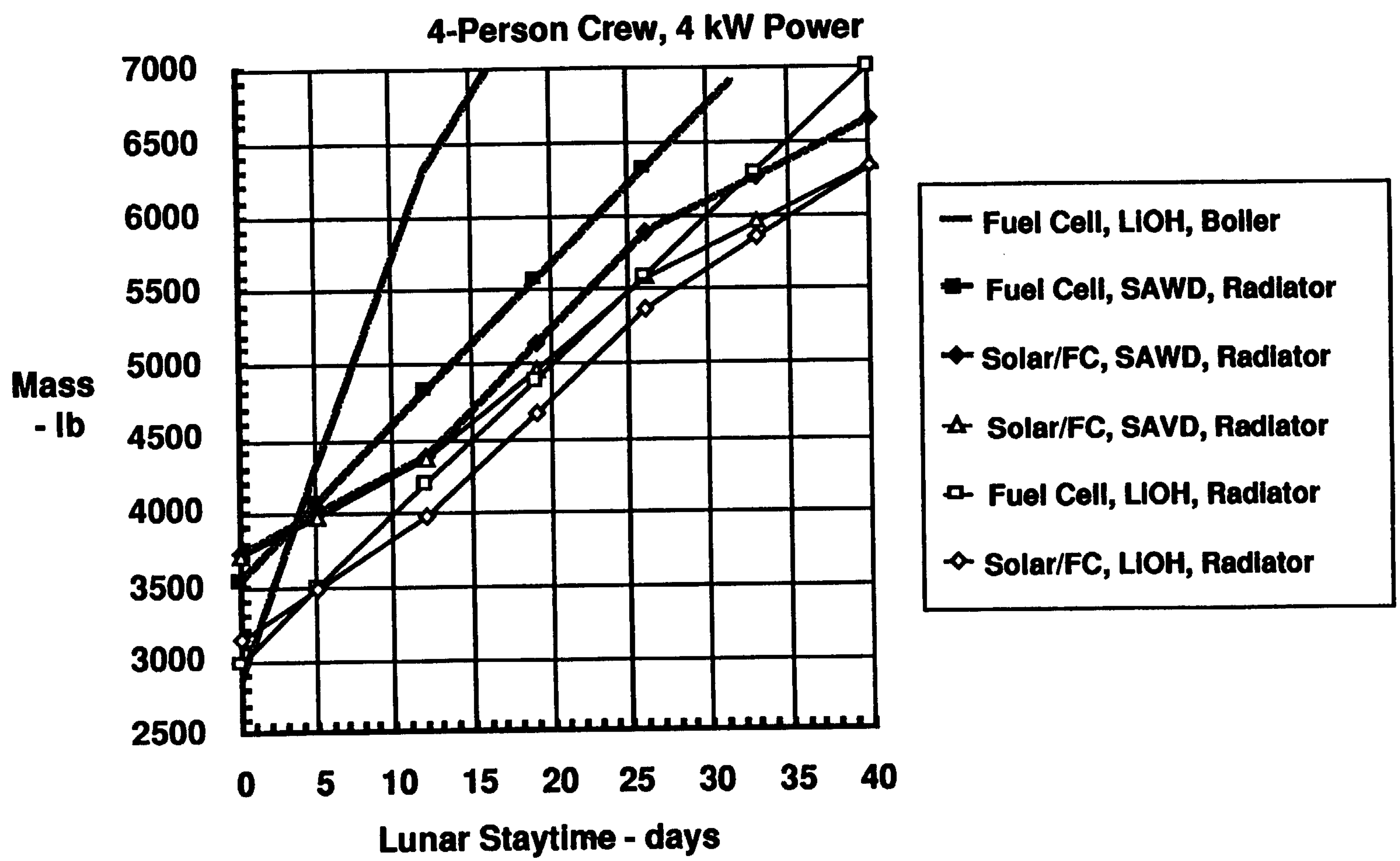


Figure 3.3.2-9 ECLSS Systems Interaction



Minimum mass systems: 0-1 days: Fuel Cell, LiOH, Boilers;
2-5 days: Fuel Cell, LiOH, Radiators;
6-40 days: Solar/Fuel Cell, LiOH, Radiators;
40+ days: Solar/Fuel Cell, SAVD, Radiator

Figure 3.3.2-10 Combined ECLSS Systems Mass Comparison

3.3.2-11 for any extended lunar stay, the initial jump in mass is for a solar storm shelter or shielding, considered a necessary addition for any extended lunar stay. Other mass increases are shown for three cases - for a design with volumes based on NASA STD-3000 volumes, a design with constant volume, and a design for a contingency stay only, with no changes in crew accommodations. These mass increases are for pre-designed changes made to optimize for a particular lunar stay, that would affect the cargo capability of the piloted vehicle.

Airlock - Because of limited use and the short habitation time on the lunar surface (48 hours after landing) the current LTS excursion module design does not include an airlock, but like the Apollo excursion module, is vented and repressurized for EVA. An external dust porch with stowage provisions for EMU outer garments can provide some exclusion of lunar dust, and suits are stowed inside the crew module. An added air shower, similar to the Skylab shower, may also be an effective means of removing dust.

If the crew is to use the excursion module for extended periods of time and do more extensive EVA's, an added airlock may become mass- and cost-effective. Options include a collapsible or inflatable airlock, and a rigid airlock.

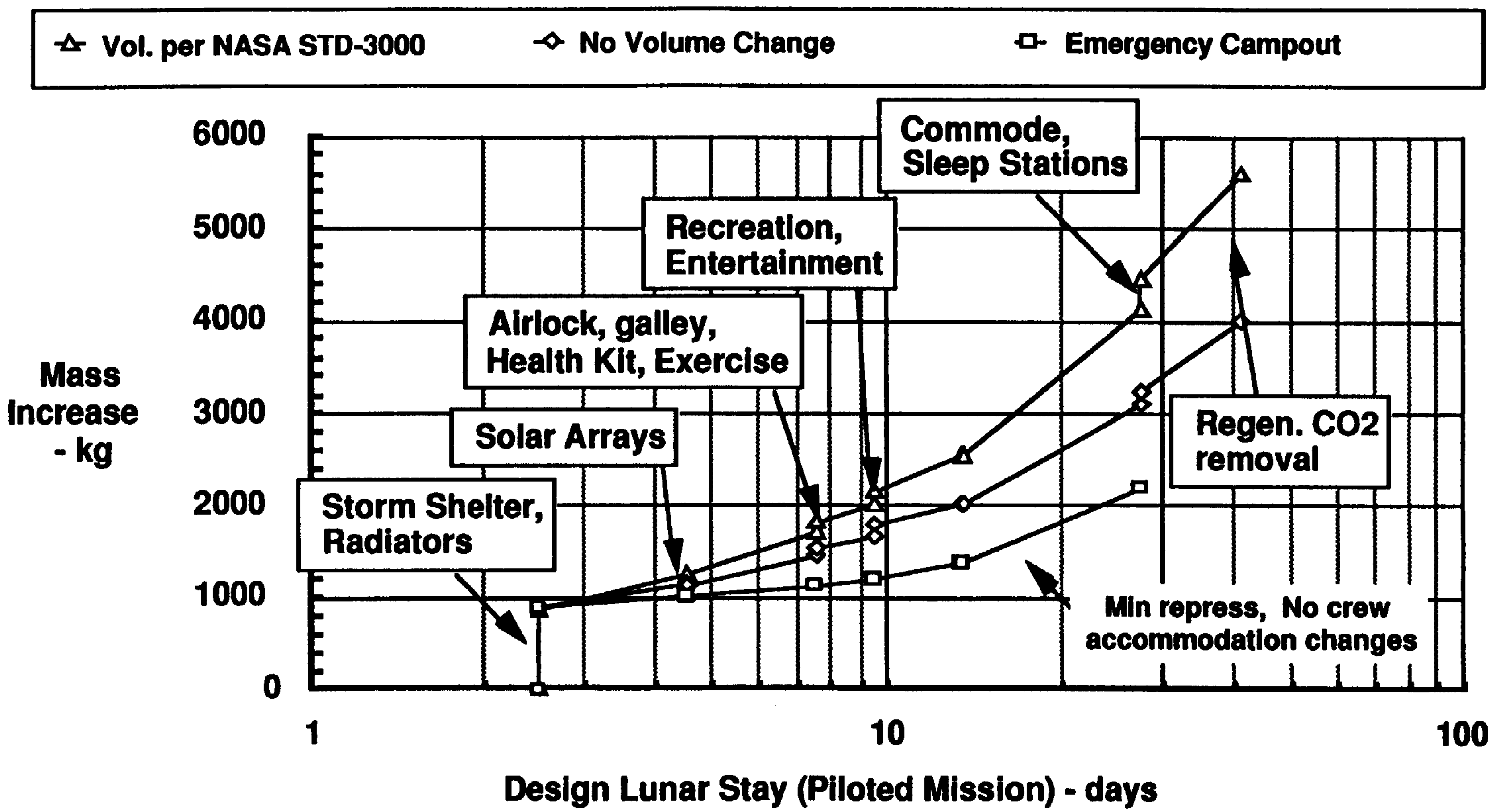
Taking into account hardware mass, added air revitalization hardware, as well as vented consumables, a rigid airlock becomes mass-effective for a 4-person crew doing 2 EVA's per day at 11 days, as shown in Figure 3.3.2-12. With a 6-person crew (additional atmosphere losses), the rigid airlock mass crossover occurs at 7 days. If the number of EVA's are increased to 3 per day, the crossover point is at 7 days for 4 crew and 5 days for 6 crew. Collapsible or inflatable airlocks could provide even better mass-efficiency, but might be more subject to damage than a rigid one.

Crew Size - Top-level mass comparisons are shown between the LTS 4-person crew modules, 6-person crew modules, and the comparable Apollo crew modules in Figure 3.3.2-13. Increase in mass from the Apollo crew modules (+45%) includes increased size, as well as a 15% weight growth allowance added to dry weights, and provision for radiation protection on board the transfer crew modules (+1100 kg).

Increasing the crew size to 6 adds 17-20% to the transfer crew modules (due to size, crew provisions, and radiation protection), and 7% to the excursion crew module mass (mainly due to size increase).

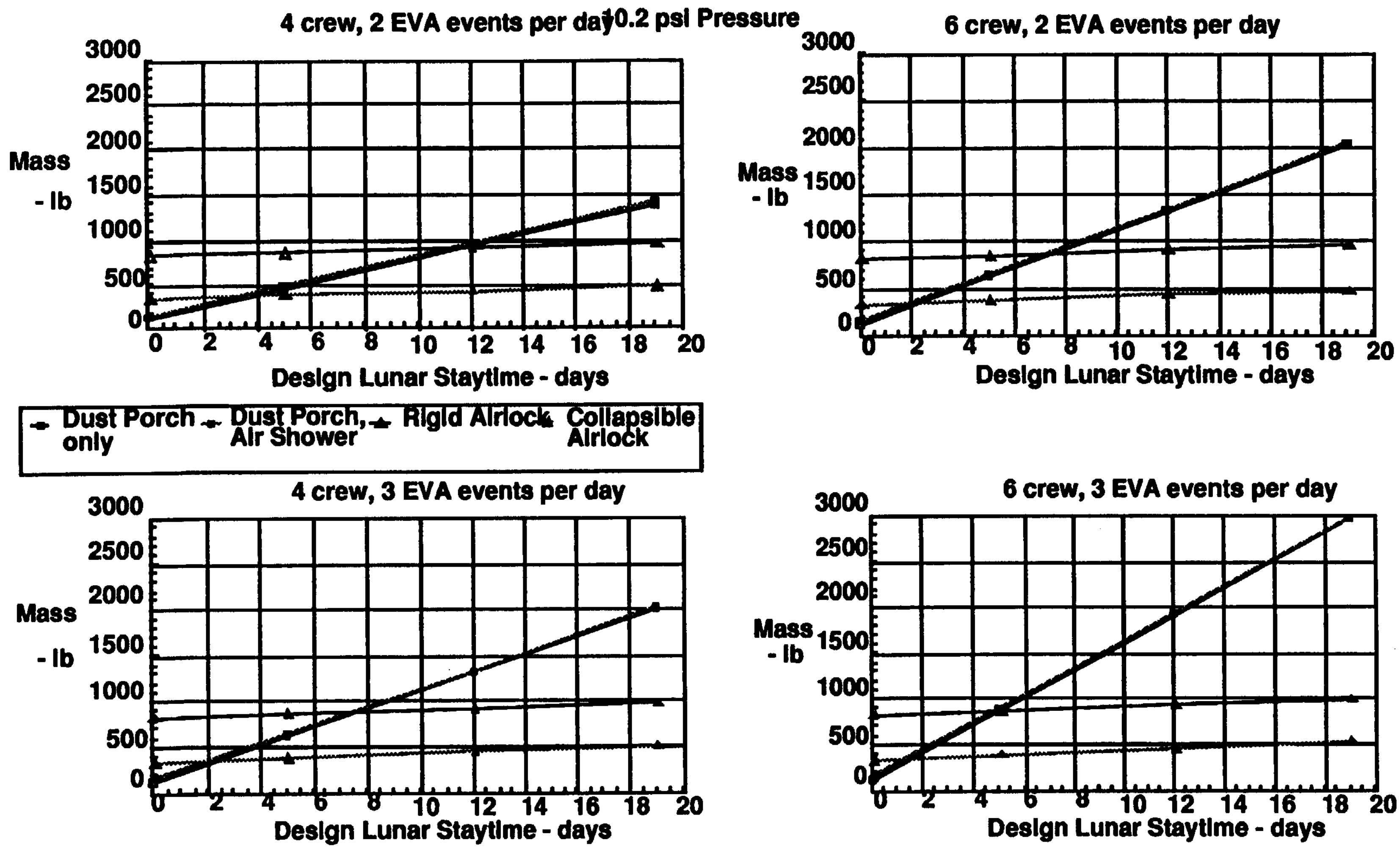
Assumptions:

- No Lunar Services Available (Power, Hab, etc)
- Power System: Fuel cells included on baseline system



These are changes made ahead of time to optimize for a lunar staytime.

Figure 3.3.2-11 Crew Module Mass Sensitivity to Lunar Staytime



A Rigid airlock becomes mass-efficient at 5 days for 6 crew/ 3 EVA events per day, and at 11 days for 4 crew / 2 EVA events per day.

Figure 3.3.2-12 Airlock Options Mass Comparison

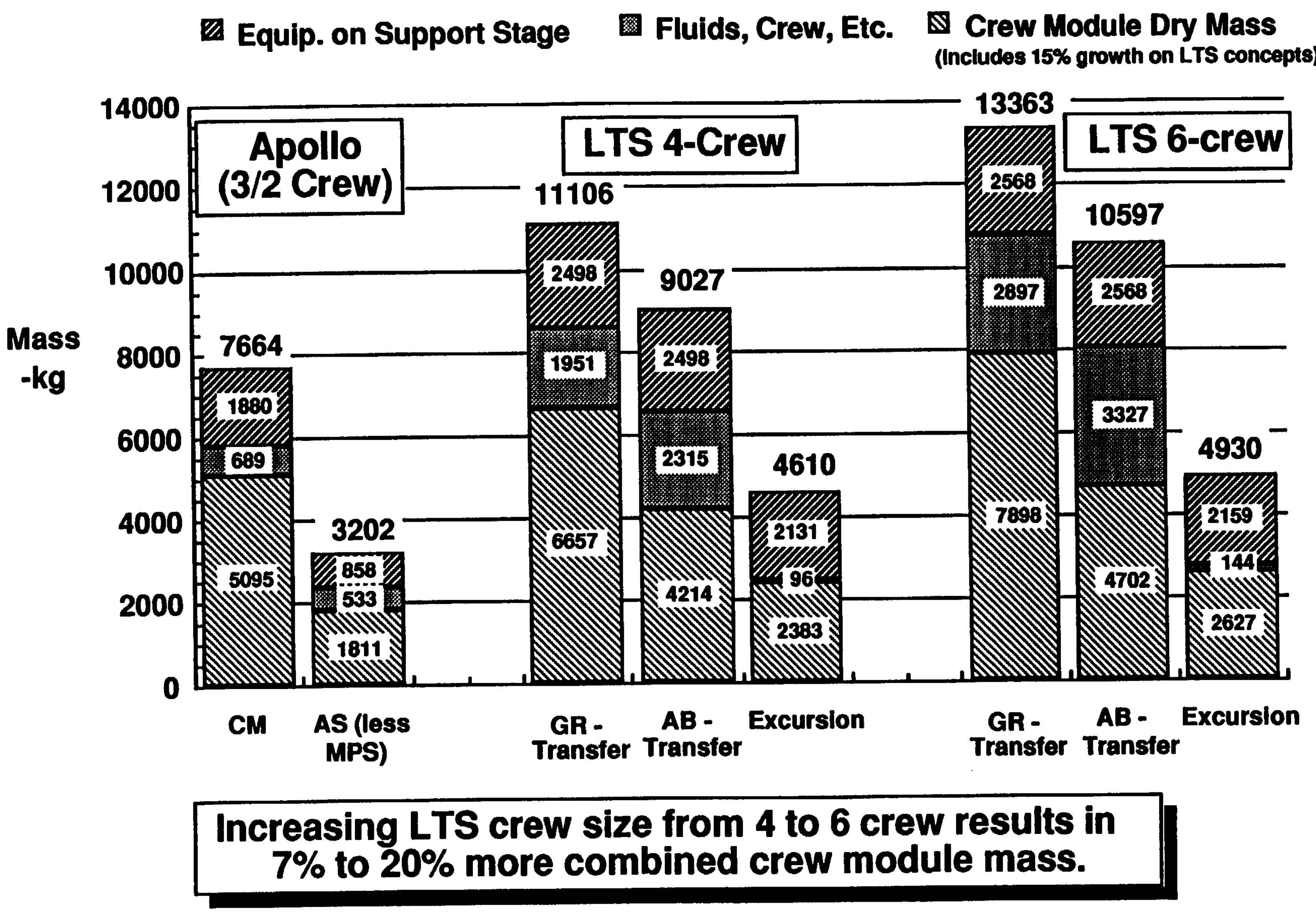


Figure 3.3.2-13 Impact of Crew Size on Crew Module Mass

3.3.3. Cost Optimum LTS Design

The goal of this period of study was to design a lunar transportation system that has minimum cost and that operates from one or two launches of one of the NLS-derived launch vehicles. In order to decrease program cost, several issues needed to be addressed, including affordability, operability and supportability, as well as the reduction of program risk.

Based on the results of past STV design trades, a cost-optimum vehicle concept maximizes commonality with other system elements, and minimizes the number of stages and crew modules to reduce development and production costs. A possible flow of vehicle design derivation to maximize commonality is shown in figure 3.3.3-1. An unpiloted lunar lander and ascent stage can use avionics, propulsion systems, and reaction control systems developed for an NLS upper stage, which derives its systems from the NLS launch vehicles and cargo transfer vehicles. From the unpiloted vehicles, systems can be derived for a large Trans-lunar Injection stage, as well as piloted landers and ascent stages. Lunar crew modules can use systems developed for a Personnel Launch System and/or ACRV.

Further reduction of cost and program risk may be achieved by reducing the cost of lunar surface systems, and by changing the philosophy or reason for going to the moon. A study of alternate lunar scenarios was also accomplished in this task.

a. NLS Upper Stage Characterisitics – Current NLS upper stage designs call for a vehicle that will deliver 15,000 lb to GEO using an NLS 1.5-stage vehicle, or in a two-stage mode, where the upper stage is staged suborbitally. The Air Force-sponsored Upper Stage Responsiveness Study in 1989 produced a range of designs based on Titan IV and ALS launch vehicles that are consistent with the current NLS study requirements. Shown in Figure 3.3.3-2 is a concept from that study that we have used as a reference concept.

Commonality between the STV ascent stage and the NLSUS can lead to significant development cost savings. Several issues that need to be addressed in the NLS upper stage design that could impact the Evolution to an STV stage include mission requirements, growth capability (tank size), engine redundancy level, and avionics evolution.

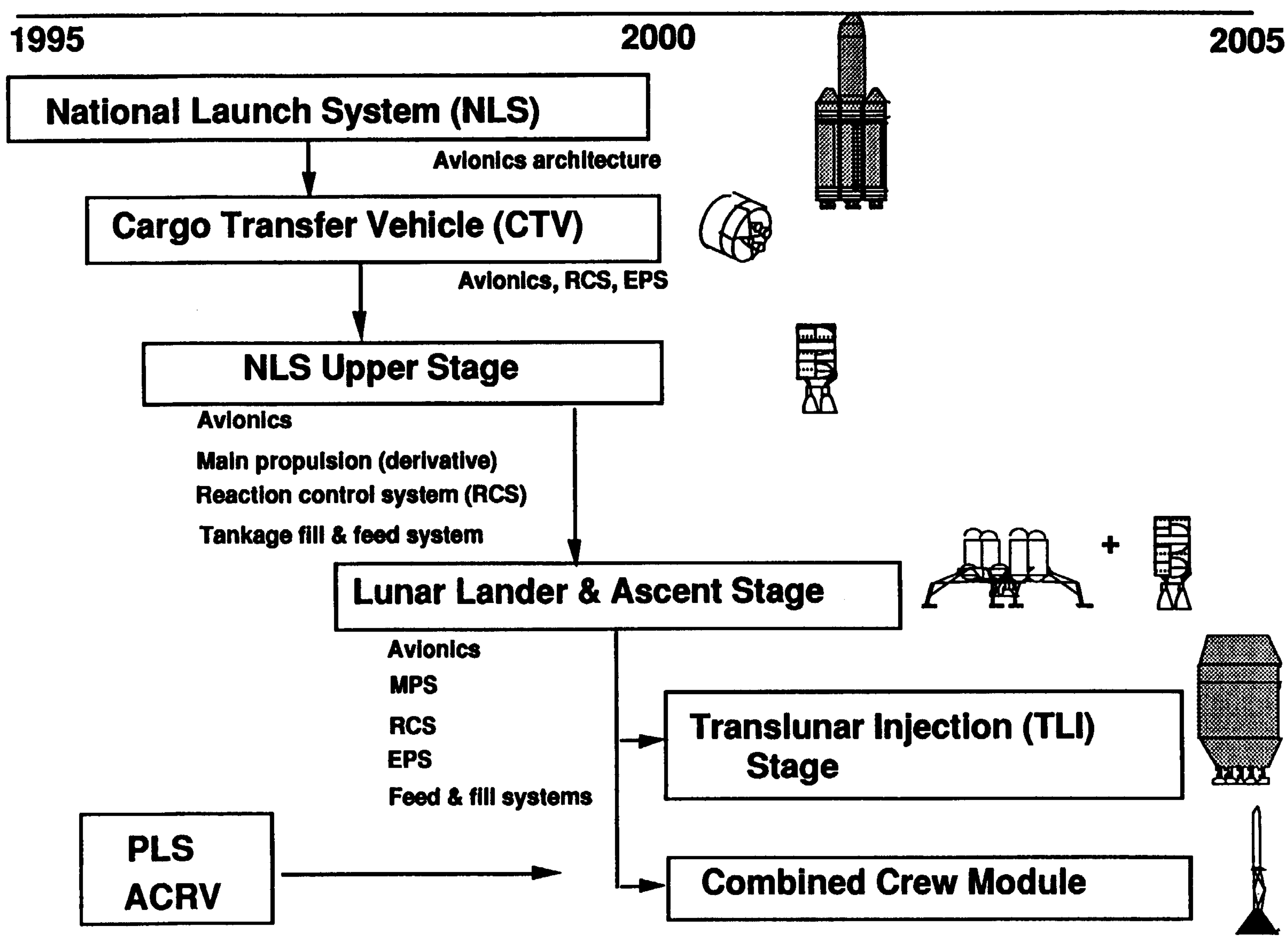
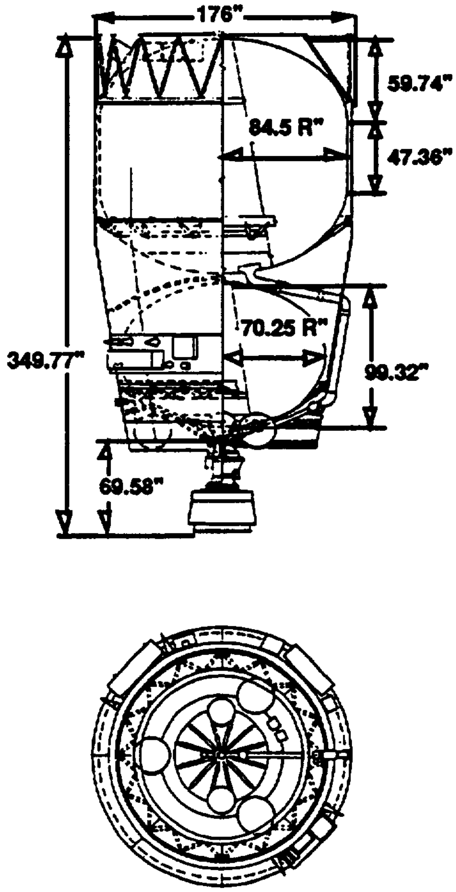


Figure 3.3.3-1 SEI Transportation Architecture Based on NLS Elements



* Payload requirement 15 klb to GEO on NLS-2
* Based on Titan IV upper stage for USRS

Features

* Single RL10-4 engine
  * 21,200 lb thrust, 447.4 sec Isp at MR 6.0
  * 90 in. length with nozzle extended
  * Expansion ration 84
* Load carrying body shell construction
* .707 elliptic dome propellant tanks
* Pallet mounted, redundant avionics
* Vehicle size (reference):
  * Propellant loaded 48,900 pounds
  * Stage dry weight 5,745 pounds

Figure 3.3.3-2 NLS Upper Stage Reference Configuratior.

b. Suborbital Staging Benefits – With only circularization to a 160 nm orbit provided by the upper stage, the HLV#1 delivers only 55700 kg to 160 nm, with a staged mass margin of 6030 kg. By staging suborbitally, the LEO-delivered mass increases to 84533 kg for the core stage launch (44 klb stage thrust) and 94058 kg for the TLI stage launch (110 klb stage thrust). The increase in delivered mass corresponds to a large increase in stage size due to the propellant required for ascent and LEO circularization.

The impact of reducing the size and thrust level (i.e. engine number) of the TLI stage is shown in Figure 3.3.3-3, with the current reference shown with 5 RL10's. It can be seen that the delivered cargo is relatively insensitive to the TLI stage mass, with a 82 t increase in TLI stage size (inert + propellant) resulting in only a 4.5 t increase in delivered cargo. Using a single J-2 engine causes a large increase in TLI stage size, with little or no increase in delivered payload.

In summary, some benefit was seen from suborbital staging, especially if payload margins are small.

c. Core Stage Configuration – One of the issues associated with using an NLSUS-derived core stage is the landed configuration of the vehicle. With the RL10-B2 engines on the NLSUS-derived core, the crew module is situated 13.1 m above the lunar surface, as shown in Figure 3.3.3-4 for an early cost-optimum vehicle configuration. If the core stage is reconfigured placing the LO2 in saddle tanks, the height of the landed configuration is reduced to 10 m and the landing gear spread is reduced, but the crew module is now recessed in between the drop-tanksets, and the core is no longer a direct derivative of the NLS upper stage.

The benefit of reduced height off the lunar surface was not deemed sufficient reason at this point to alter the vehicle design. Later analysis of the ground-based vehicle configuration, and the use of RL10-A4 engines further reduced the crew module height to 10 m.

d. Lunar Surface Boiloff Reduction – One of the disadvantages of high-performance cryogenic propellants is the associated boiloff during long-duration missions, especially on the lunar surface during a six-month stay. A basic groundrule of the initial performance analyses, assuming an established lunar base, was that some form

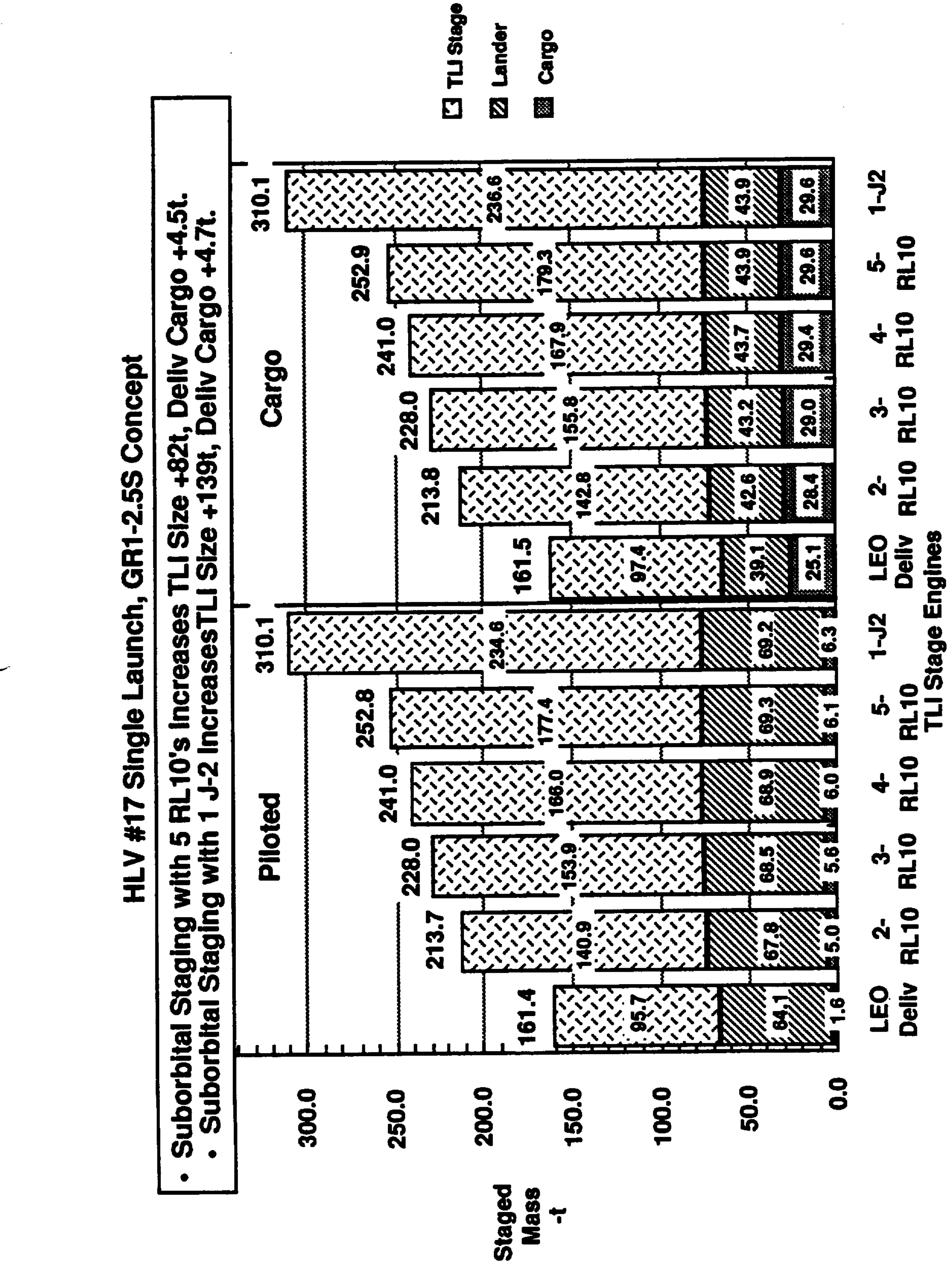
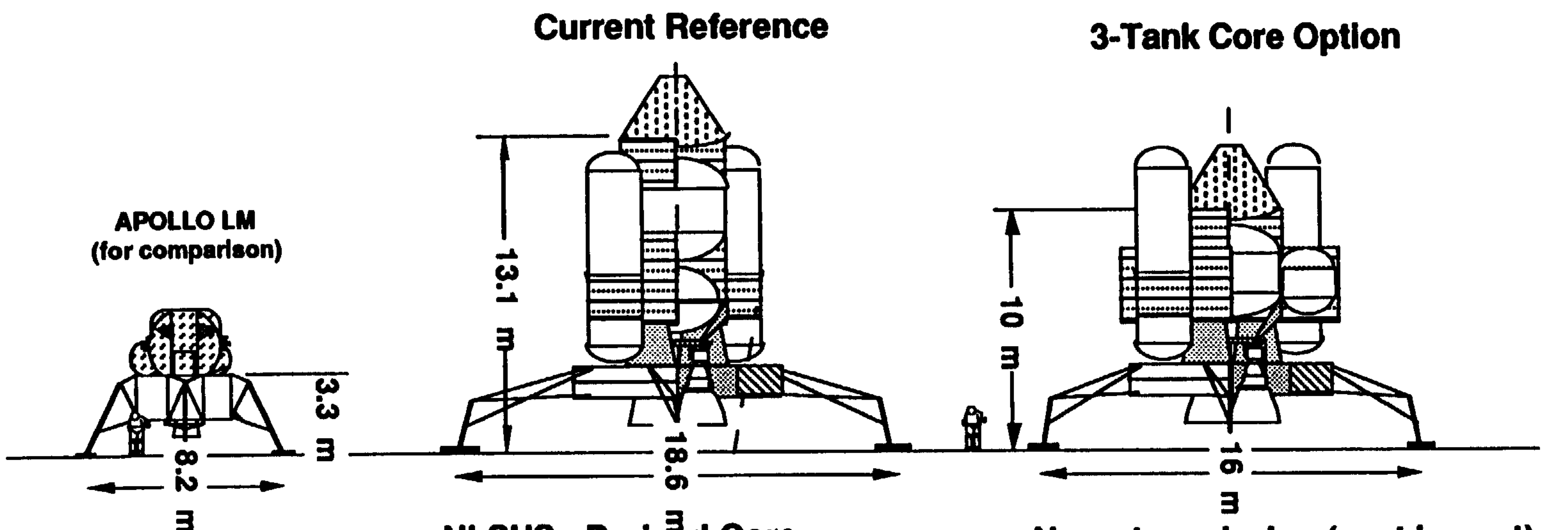


Figure 3.3.3-3 Suborbital Staging Impact on Cargo

Issues:

- **Landed configuration - distance to lunar surface and landing gear sizing.**
- **Core design as derivative of NLS upper stage.**



- **NLSUS - Derived Core (cost-savings)**
- **Landing characteristics poor. (performance impact)**

- **New stage design (cost impact)**
- **Better landing characteristics (performance benefit).**

Figure 3.3.3-4 Core Configuration Trade Summary of cryogenic propellant refrigeration would be provided on the lunar surface and that boiloff was controlled.

In the absence of such a refrigerator, several options have been proposed to minimize the impact of propellant boiloff. Those include improved thermodynamic vent systems, reduced heat leak paths, and additional insulation (all currently included on the reference vehicle), vapor-cooled shields, a reflective lunar surface 'tarp' to minimize heat reflection from the surface, on-board cryo refrigeration, or the use of storable propellants on an ascent/return stage.

The effect of vapor-cooled shields on boiloff rate can be seen in Figure 3.3.3-5, especially for LH2. At the current design MLI thickness of 3.3 cm, the lunar day average LH2 boiloff rate is about 2.7% per month. With vapor-cooled shields, the rate drops to about 0.6% per month. These rates are given for the current reference vehicle concept, with 13.8 t of propellant remaining in the core stage at lunar landing.

Combinations of MLI, TVS, refrigeration, and reflective surface cover have been compared with the reference case to arrive at a minimum mass solution to minimize surface boiloff. That solution can then be traded against a storable propellant ascent stage option.

The reference design has 80-layer MLI, a mixing thermodynamic vent system for on-orbit thermal control, optimized tank support struts, and no reflective surface cover. Because the reflective surface cover is a large mass item and would probably not be reusable, that option was not considered further.

The various boiloff-control options are shown plotted in Figure 3.3.3-6 as cargo mass impact versus time on the lunar surface, compared to a no-boiloff reference. The minimum mass system (24-layer MLI, no TVS or refrigeration) gives the most benefit only for 2 - 5 days stay, then sharply decreases cargo capability with longer staytimes. The option with the flattest slope is the option with an on-board LO2 refrigerator, optimized thermal struts, 80-layer MLI, and a vapor-cooled shield, and reduces the cargo capability by 1100 kg (6%) for a 6-month stay. For a 45-day stay, the least-impact system is the current design, with 80-layer MLI and thermally-optimized struts.

The alternate option is to use a storable propellant ascent stage, but as was seen before, the cost and poorer performance may outweigh the benefits.

Opt	Prop.	layers MLI	Struts	VCS *	Refrig?	% Boiloff/ Mon Average O2	% Boiloff/ Mon Average H2	Mass change (1st order) MLI	Mass change (1st order) Other
1	Cryo	80	optimized	no	yes **	0	0	0	0
2	Cryo	24	LeRC	no	no	7.2	18.3	-87	0
3	Cryo	24	LeRC	yes	no	5.9	9.4	-87	+307
4	Cryo	24	optimized	yes	no	2.4	6.9	-87	+307
5	Cryo	80	LeRC	no	no	5.7	9.7	0	0
6	Cryo	80	LeRC	yes	no	4.8	5.4	0	+307
7	Cryo	80	optimized	yes	no	1.1	2.3	0	+307
8	Cryo	80	optimized	no	no	1.3	5.4	0	0
9	Cryo	80	optimized	yes	LO2 ***	0.03	2.3	0	+278

Lunar Base Reference

* Single Vapor-cooled shield
** 1718 kg; supplied by Lunar Base
*** Solar array power during lunar day.

Figure 3.3.3-5 Lunar Boiloff Trade Options

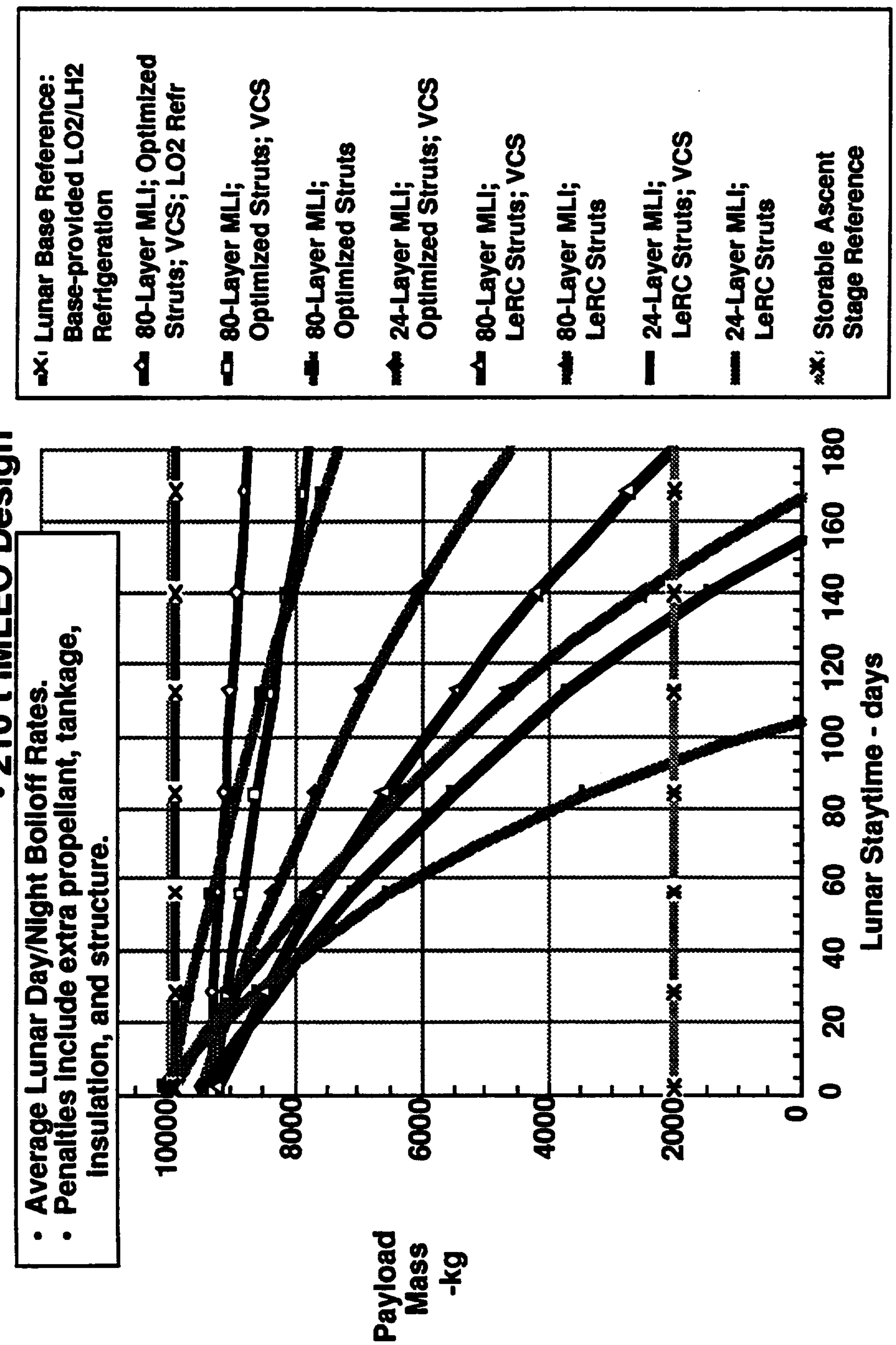


Figure 3.3.3-6 Lunar Boiloff: Piloted Payload Degradation

C-4

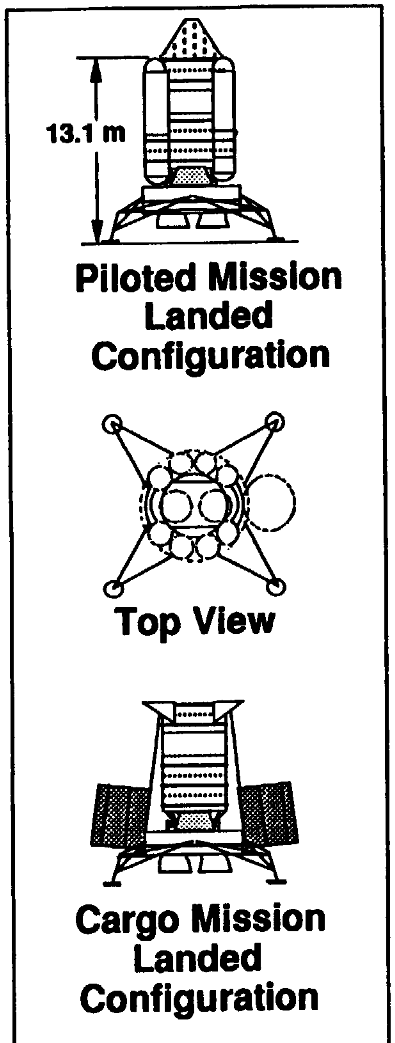
e. Shroud Diameter - 8.4 m vs. 10 m – The initial sizing of the current LTS vehicle concept was done with an 8.4 m diameter shroud in order to maintain a common diameter with the launch vehicle. For the current reference design with expendable lunar drop-tanks, however, a larger diameter is desired to reduce the vehicle complexity and to reduce the height of the crew module from the lunar surface. This trade compared an earlier vehicle concept designed for the 8.4 m shroud with one for a 10 m shroud to determine any performance or design benefits.

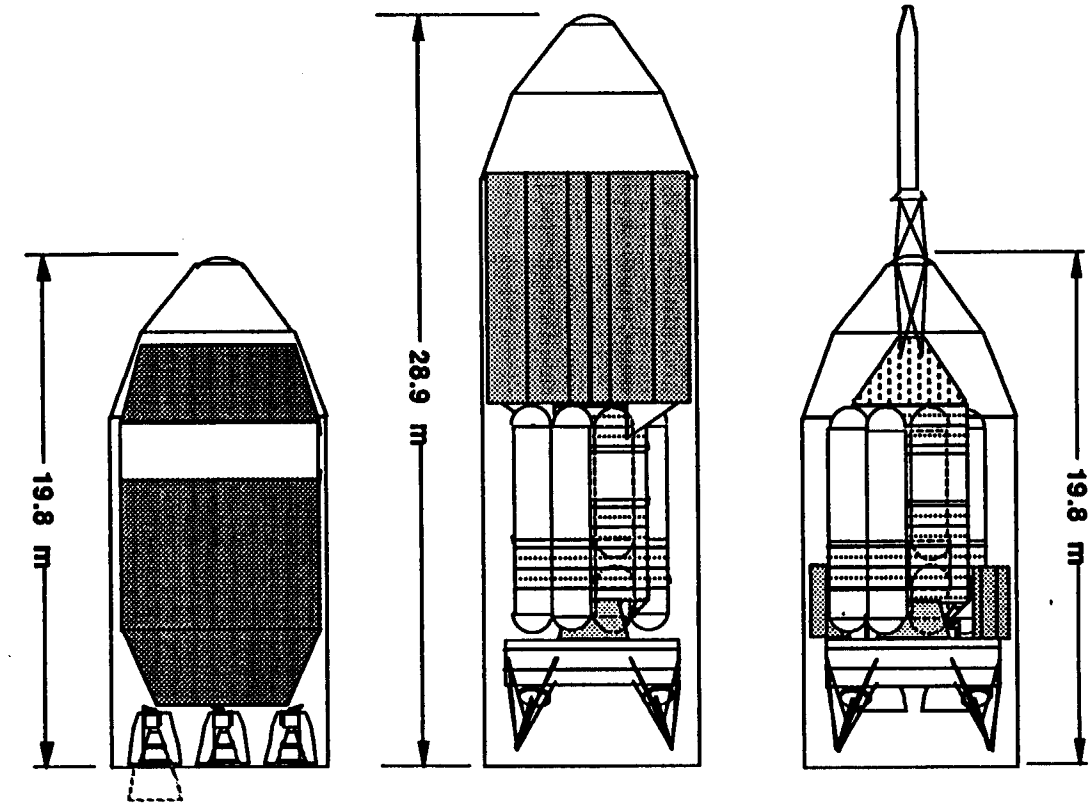
With an 8.4 m shroud, as shown in Figure 3.3.3-7, the vehicle includes a 4.4 m diameter core stage sized for 24.1 t propellant with 8 drop-tanksets, as well as a 7.8 m diameter TLI stage. The piloted mission capability is 2.6 t of lunar cargo, and a dual-launch cargo capabilityis 26.8 t cargo. In a single launch mode, the cargo capability is 9.9 t.

With a 10.0 m shroud, as shown in Figure 3.3.3-8, the vehicle includes the same 4.4 m diameter core stage sized for 24.1 t propellant, but with only 6 drop-tanksets, as well as an 8.4 m diameter TLI stage. The piloted mission capability is 3.0 t of lunar cargo, and a dual-launch cargo capability is 28.3 t cargo. In a single launch mode, the cargo capability is 11.6 t.

The 10.0 m diameter shroud size has both performance and configuration benefits and is the preferred size. It is currently assumed for the reference concept.

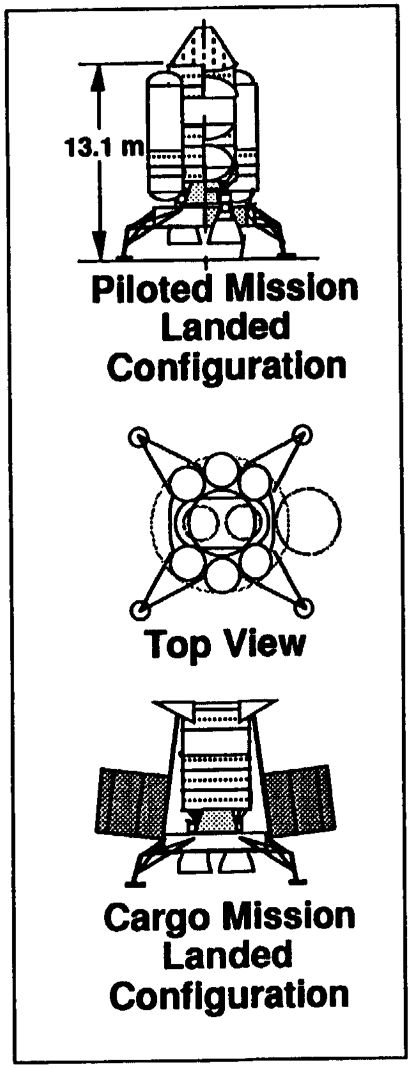
f. Alternate Exploration Scenarios – After transportation system costs and risks have been minimized, the only remaining way to reduce overall program cost and risk is by reducing the cost of surface systems, or by extending the effectiveness of the astronauts to reduce the number of manned flights required. Accordingly,we performed an architectural analysis to review the lunar exploration goals and activities, and to propose alternate goals and activities which have different emphasis areas and modified schedules. The first question we asked is "Why should we go back to the moon?" The alternatives include setting up and maintaining a Lunar base, performing geological / astronomical science, using the moon mission as a precursor to a Mars mission, and making use of lunar resources. We conducted an informal survey on that question in order to determine which exploration scenario would gather the most public support.

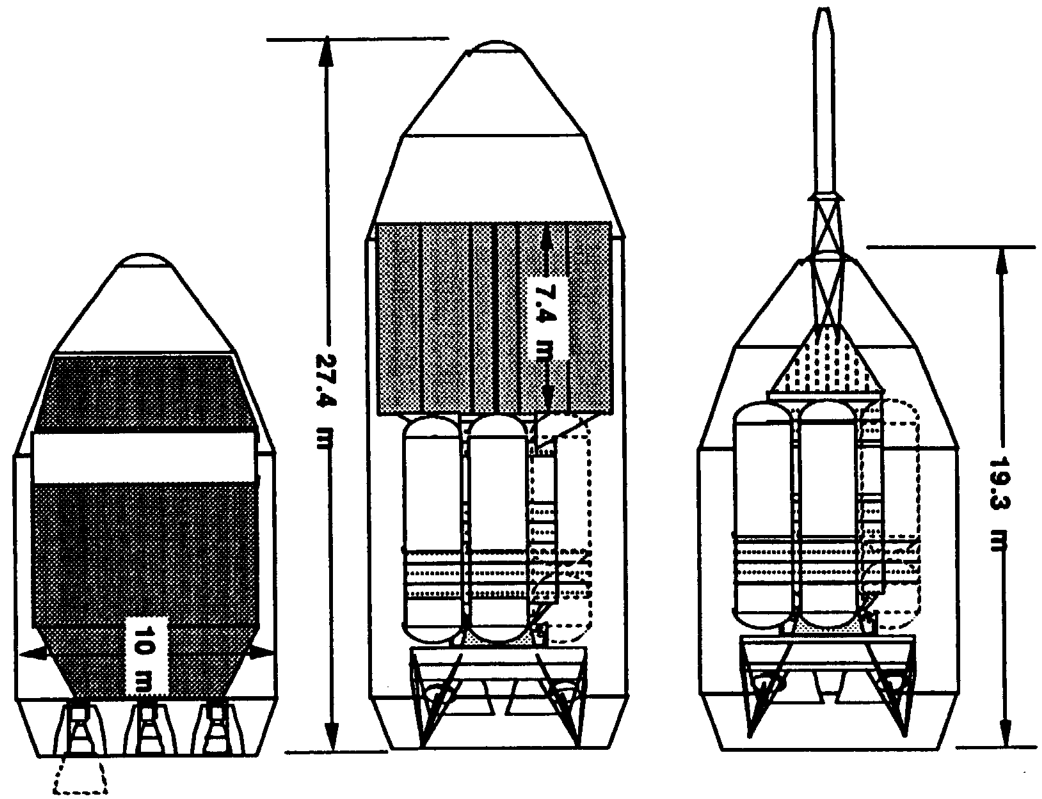




	TLI Stage	Cargo Vehicle 2-Launch	Cargo Vehicle 1- Launch	Piloted Vehicle
Shroud Mass:	8.5 t	12.4 t	12.4 t	8.5 t (+3.5 t LES)
HLV Perf. Margin:	16.7 t	9.4 t	9.6 t	9.5 t
LTS Total Mass:	150.6 t	85.0 t	86.8 t	85.1 t
Lunar Cargo Mass:	n/a	26.8	9.9 t	2.6 t

Figure 3.3.3-7 8.4 m Diameter Shroud Configuration





	TLI Stage	Cargo Vehicle 2-Launch	Cargo Vehicle 1- Launch	Piloted Vehicle
Shroud Mass:	9.0 t	13.6 t	13.6 t	9.0 t (+3.5 t LES)
HLV Perf. Margin:	16.7 t	12.4 t	12.4 t	12.4 t
LTS Total Mass:	150.5 t	113.5 t	113.5 t	113.1 t
Lunar Cargo Mass:	n/a	28.3	28.3	3.0 t

Figure 3.3.3-8 10 m Shroud Configuration

The most often stated reason is to perform lunar science and better understand formation of the solar system. That answer is acknowledged as a good reason to the average layman, but not as the sole justification for a $50+ B program. The next most stated reason is to expand the human habitat and demonstrate readiness to go on to Mars. That answer does little for the layman, because the trip to Mars is at least thirty years in the future and so is viewed as science fiction.

The proposed reason which got the most support was to discover and map valuable resources for future utilization. The average layman feels humankind will expand into the solar system someday, and it is worth an investment now to insure his/her descendents will have access to the most valuable regions in the distant future. With this knowledge in hand, we set out to design a "lunar prospector scenario".

Locating valuable resources on the moon will not be an easy task. Most of the surface is covered with several meters of semi-homogenous regolith. Exposed bedrock can be surveyed optically using spectroscopy, but vast areas must be surveyed by core drill down through the regolith. This will require surface exploration by long range rovers, which are in the baseline scenario, but happen well after the permanent outpost because they require extensive ground support systems.

If surface exploration and minerals prospecting is the primary goal of this scenario, then why not introduce long range rovers first, and build the permanent outposts later on the most lucrative sites for further development. The benefits of this approach are: probable delay in the development of the expensive surface infrastructure, a much better understanding of surface conditions when it finally becomes time to build, and the possibility that if the moon is useless for future development we will know about it before we commit major resources.

When we examined the requirements for long-range rovers, we determined the key missing "ingredient" was a lightweight, long-lived power source. We then surveyed power sources and found two capable of doing the job, and one outstanding near-term candidate. One already proposed system is the regenerative fuel cell combined with solar panels to recharge the system during lunar day. Although heavy (2.5 tons for a 30 ton rover), this system is already being space qualified for SSF so it should be available. Both the other candidates are radioisotope powered electrical generators. The Radioisotope Thermoionic Generator (RTG) is a proven, space qualified unit with ten to twenty year life. Its problem is low efficiency (<7%) which makes it heavy (6 watts/kg).

A new concept now in test, the Radio-isotope Thermophotovoltaic (RTPV) power source, promises efficiencies over 30% using the same standard radioisotope energy source. The efficiency comes from using the white-hot radioisotope as a light source for special photovoltaic cells designed to operate in the infrared part of the spectrum. An important feature of these cells and their backing plate is that most of the light they cannot convert into electricity is reflected back into the radioisotope source where it is reabsorbed and remitted. This allows high efficiency and 20 to 30 watts/kg specific power.

A design sketch of a possible habitat on wheels long-range rover configuration is shown in figure 3.3.3-9. Key features proposed are: a 30 meter coring drill, a heavy-duty manipulator arm to uncover samples and clear obstacles, a television eye mounted on top of tall antenna for surface navigation, and redundant airlocks to insure egress and ingress. The power cart towed behind the vehicle contains fifteen 1kw RTPV generators and the waste heat rejection radiators.

The man tended rover would be landed intact but unmanned using a modified standard cargo lander as shown in figure 3.3.3-10. The need to land almost 30 tons in one piece drives the low "bed height" for this type of lander, and the elevated propulsion tankage is not a large weight penalty.

The results form the rovers first architecture analysis indicated that 8 manned missions and 10 cargo flights are required to deliver 315 tons of man-tended facilities and transportable elements. The biggest savings are probably in the cost of surface elements, but they haven't been costed during this exercise.

A cost analysis of the various transportation options is shown in figure 3.3.3-11. The baseline LTS using the reference 70 ton NLS HLLV had a transportation LCC of $51B. Switching to a new 150 ton liquid-boosted launch vehicle reduced the number of launches by half, but showed no improvement in LCC, due to the increased DDT&E and cost per flight. Adding reusability to the booster elements using in-house data on recoverable engine pods, reduced program launch costs (and LCC) by about $4B.

- Supports four crew for 60 days to two years
- Prospects for critical resources, performs science studies, and services unmanned observatories
- Sized for intact delivery by a single cargo lander
- Powered by advanced radioisotope generators

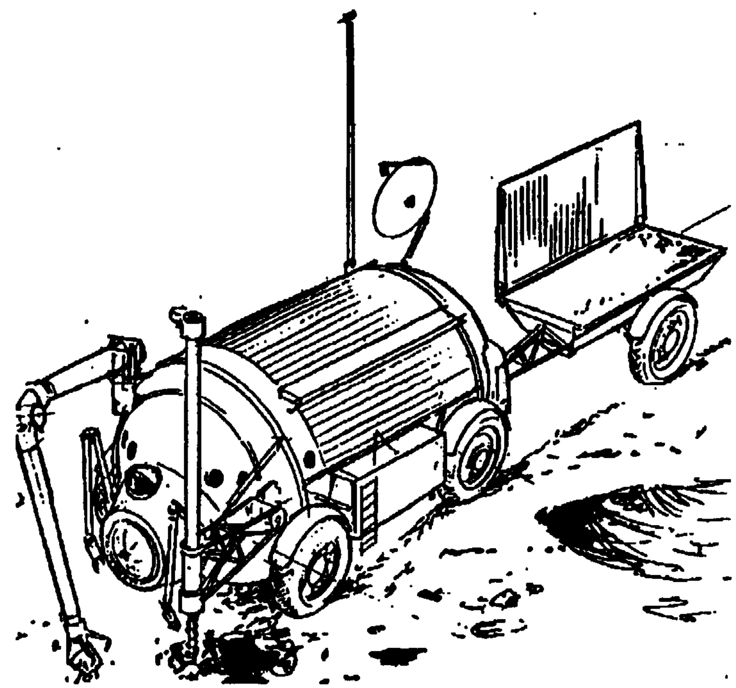


Figure 3.3.3-9. Proposed Rover First Concept

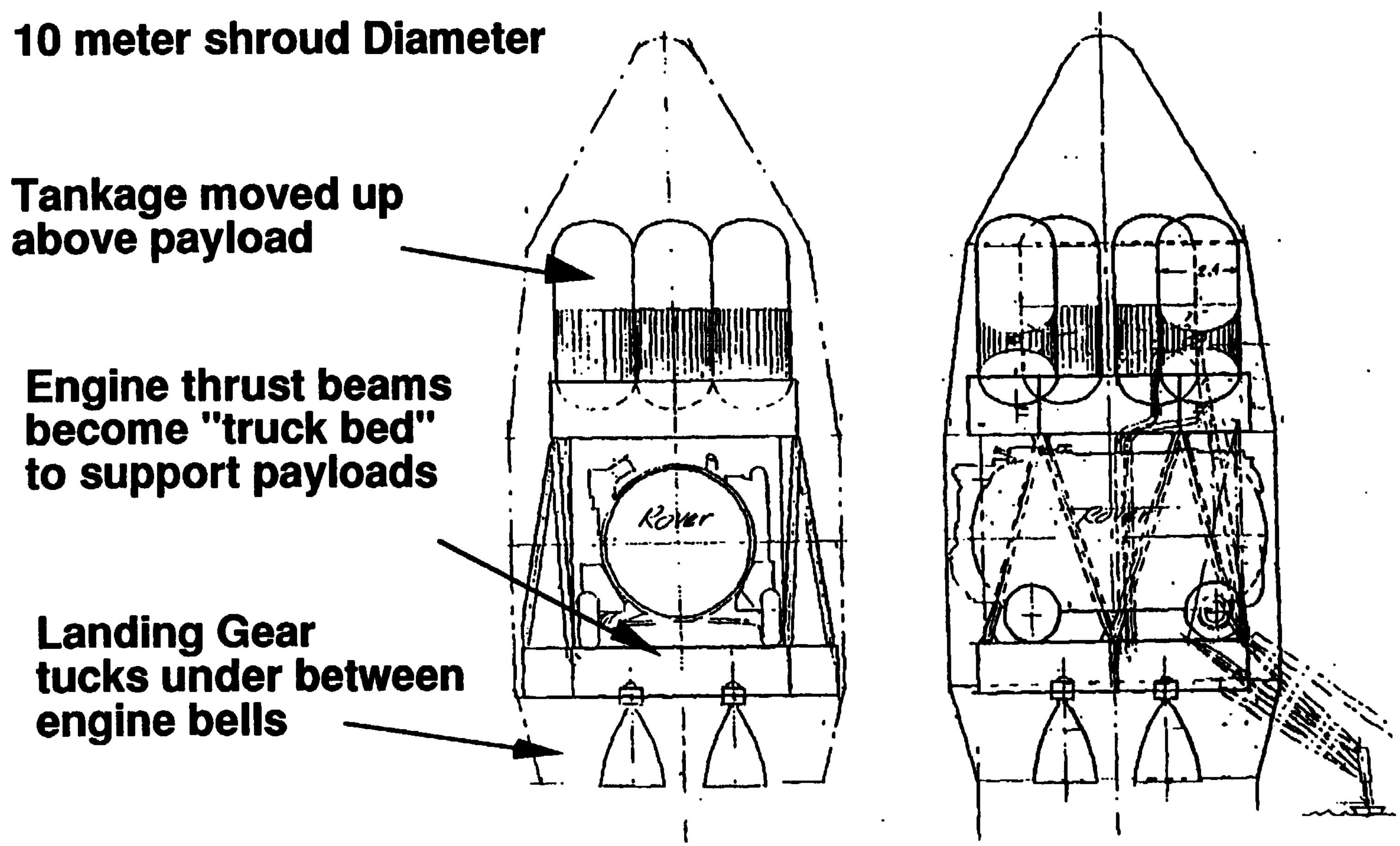


Figure 3.3.3-10. Rover Capable Lander

Program Option	STV TD-03 Baseline reported at IRR #8	ETO Architectural study update (Dec 1991)	ETO Architecture updated w/reusable P/A Modules	"Rovers First" architecture with P/A Module ETO
ETO transportation	70 t HLLV #1 using dual launch w/R&D	150 t ETO option #17 single launch	Upsized ETO option #17 with P/A Module recovery	Same
Lunar transportation system option	2.5 stage, all cryo single cab, ground based	Same	Same	Same
Mission Scenario	Moon to stay scenario (17 piloted missions, 418 tons of cargo)	Same	Same	"Rover First" scenario (8 piloted missions, 315 tons of cargo)
LTS Life Cycle Cost	$51 billion (64 launches = $10.4B0	$49 to 52B (32 launches ≐ $12.4 to 15.5B)	$44 to 48B (32 launches ≐ $8.7 to 10.9B)	$41 to 44B (20 launches ≐ $5.4 to 6.8B)

Figure 3.3.3-11. Results of LTS Cost Tra

Incorporating the new "Rovers First" mission architecture reduced the number of flights from 32 to 20, providing savings with respect to transportation elements of about $1B in production costs and $3B in launch costs. Overall advantages of the rovers first scenario are: lower transportation LCC ($4B reduction); lower technical risk since program success doesn't hinge on delivery, assembly, and operation of an integrated base complex; and lower political risk since the program requires less early development of surface infrastructure and has a more salable program goal.

g. Lunar Campsite Scenario – An adaptation of the Rovers First Scenario is a Lunar campsite scenario, where an unmanned cargo flight delivers a lunar habitat to a point of interest on the moon, followed by a piloted mission that delivers the crew and a small cargo with rover and supplies for a 45-day stay. During that time, the crew lives and works from the habitat, which remains on the cargo vehicle. Following the lunar stay, the crew returns in the piloted ascent stage and return directly to the ground in a ballistic return crew module.

This vehicle concept, shown in Figure 3.3.3-12, became our final reference concept, as it is similar to the cost-optimized concept studied in Task 3 and meets all requirements for the lunar outpost missions as well. In addition, this concept can be used for the Rovers First Scenario, with the fixed habitat replaced by the roving habitat shown in Figure 3.3.3-9. This concept could be even more versatile in that the ascent stage could be a derivative of the NLS upper stage, with improvements in long-term cryo storage capability, and could capture other non-SEI missions as well.

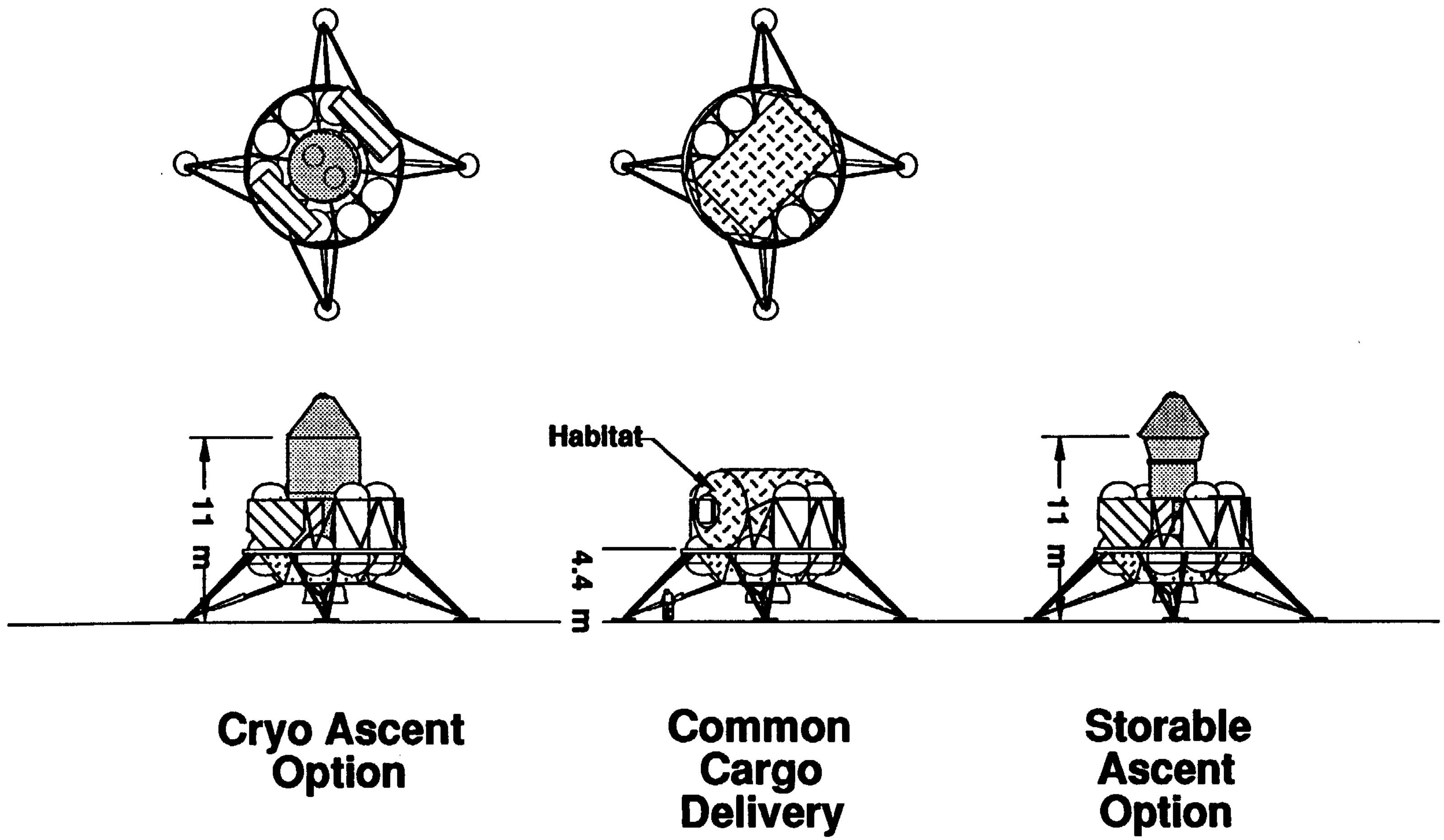


Figure 3.3.3-12. Reference Lunar Campsite Concept