Space Transfer Vehicle
Concepts and Requirements Study

Volume II Final Report
Book 1 STV Concept Definition and Evaluation

April 1, 1991
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SPACE TRANSFER VEHICLE
CONCEPTS AND REQUIREMENTS STUDY

Phase I Final Report
Volume II, Book 1
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COLOR ILLUSTRATIONS

FOREWORD

This final report of the first phase 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 1991.

This final report is organized into the following seven documents:

Volume I   EXECUTIVE SUMMARY

Volume II FINAL REPORT
        Book 1 - STV Concept Definition and Evaluation
        Book 2 - System & Program Requirements Trade Studies
        Book 3 - STV System Interfaces
        Book 4 - Integrated Advanced Technology Development

Volume III PROGRAM COSTS ESTIMATES
        Book 1 - Program Cost Estimates (DR-6)
        Book 2 - WBS and Dictionary (DR-5)

The following appendices were delivered to the MSFC COTR and contain the raw data and notes generated over the course of the study:

Appendix A   90 day "Skunkworks" Study Support
Appendix B   Architecture Study Mission Scenarios
Appendix C   Interface Operations Flows
Appendix D   Phase C/D & Aerobrake Tech. Schedule Networks

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

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ACRONYMS

AC	attitude control
ACS	attitude control system
ALS	Advanced Launch System
APU	auxiliary power unit
ASIC	application-specific integrated circuit
ATC	active thermal control
ATDRSS	advanced TDRSS
BIT	built-in test
BOLT	Boeing Lunar Trajectory Program
CASE	computer-aided software engineering
CNDB	civil needs database
CNSR	comet nucleus sample return
CT	communications and tracking
CTE	coefficient of thermal expansion
DAK	double aluminized Kapton
DDT&E	design, development, test, and evaluation
(delta) T	change in event duration
(delta) V	change in velocity
DoD	Department of Defense
DMR	design reference missions
DRS	design reference scenario
DSN	deep space network
ECLSS	environmental control and life support system
EOS	Earth observing system
EPS	electrical power system
ESA	European Space Agency
ETO	Earth to orbit
EVA	extravehicular activity
FC	fluid control
FEPC	flight equipment processing center
FOG	fiber-optic gyro
FSD	full-scale development
GB	ground based
GC	guidance control
GEO	geosynchronous orbit
GLOW	gross liftoff weight
GNC	guidance, navigation, and control
GO	ground based, on orbit
GPS	global positioning system
GSE	ground support equipment
HEI	Human Exploration Initiative
HEO	high Earth orbit
HESR	Human Exploration Study Requirements
HLLV	heavy lift launch vehicle

ICI	Integrated Systems Incorporated
ILD	injection laser diode
IMU	inertial measurement unit
IUS	Inertial Upper Stage
IVA	intravehicular activity
JPL	Jet Propulsion Laboratory
JSC	Johnson Space Center
KSC	Kennedy Space Center
LAD	liquid acquisition device
LAN	local area network
LCC	life cycle cost
LCD	liquid crystal display
L/D	lift to drag
LECM	lunar excursion crew module
LED	light-emitting diode
LEO	low Earth orbit
LES	launch escape system
LEV	lunar excursion vehicle
LLO	low lunar orbit
LMS	lunar mission survey
LO	lunar orbiter
LOD	lunar orbit direct
LOI	lunar orbit injection
LOR	lunar orbit rendezvous
LOX/LH	liquid oxygen/liquid hydrogen
LTS	lunar transportation system
LTV	lunar transfer vehicle
MEOP	maximum expected operating pressure
MET	mission elapsed time
MEV	Mars excursion vehicle
MLI	multilayer insulation
MPS	main propulsion system
MSFC	Marshall Space Flight Center
MTPE	mission to planet Earth
MTV	Mars transfer vehicle
NEP	nuclear energy propulsion
NPSH	net positive suction head
NTR	nuclear thermal rocket
ORU	orbit replaceable unit
P/A	propulsion/avionics
PC	propulsion control
PCM	parametric cost model
PDT	product development team
PODS	passive orbital disconnect strut
PSS	planet surface system
PVT	pressure-volume-temperature

RCS	reaction control subsystem
RFP	request for proposal
RLG	ring laser gyros
RMS	remote manipulator system
RTV	room temperature vulcanizing
SB	space based
SEI	Space Exploration Initiative
SEP	solar energy propulsion
SEU	single-event upset
SG	space/ground
SIP	strain isolation pad
SIRF	spaceborne imaging radar facility
SIRTF	Space IR Telescope Facility
SLAR	side-looking aperture radar
SOS	silicon on sapphire
SRM	solid rocket motor
SSF	Space Station Freedom
STIS	Space Transportation Infrastructure Study
STS	space transportation system
STV	Space Transfer Vehicle
TDRSS	tracking and data relay satellite system
TEI	trans-Earth injection
TLI	translunar injection
TMI	trans-Mars injection
TPS	thermal protection system
TVC	thrust vector control
TVS	thermodynamic vent system
USRS	Upper Stage Responsiveness Study
VHM	vehicle health monitoring
VHMS	vehicle health management system
ZLG	zero lock gyro

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1-1.0   MISSION ANALYSIS

1-1.1   GOALS AND OBJECTIVES

The Space Transfer Vehicle Concepts and Requirements study was conducted to define a flexible, high-performance, cost-effective, evolutionary upper stage program for NASA and the United States and to provide a database necessary to proceed with system definition and planning. The study was purposely set up with few groundrules (Figure 1-1.1-1) that might drive the system design toward a particular solution.

Study Groundrules. The Space Transfer Vehicle (STV) is intended to use cryogenic (liquid oxygen/liquid hydrogen (LOX/LH)) propellants. The high-performance characteristics of LOX/LH are required because of the large mass of the lunar transfer vehicle in low Earth orbit (LEO). There are other mission requirements that could benefit from consideration of other propellants types. A tradeoff exists between cryogenic propellant and bipropellant for an ascent stage from the lunar surface. While this study analyzed architectures that included two-stage lunar landing vehicles, it was assumed that the second stage would use LOX/LH.

The interface analysis assumed launch from the Kennedy Space Center (KSC) as a groundrule. All facility requirements and modifications assume use of and upgrades as required to KSC facilities. A specific launch vehicle was not given as a basis for the STV study trades. The study parameterized the Earth to orbit (ETO) capability and used Shuttle-C, Advanced Launch System (ALS), and Titan IV characteristics as general guidelines. Very large heavy lift launch vehicles (HLLV) with about a 260-ton capability were also assumed for the ground-based single-launch options.

Space basing, reusability, and use of an aerobrake for return to LEO were all considered to be desirable for the evolutionary STV system but were not requirements for the initial vehicle capability. The architectural trade study covered all three of these system-level tradeoffs.

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• Propellant type:	Cryogenic (LOX/LH)
• Launch site:	KSC
• System capable of evolution* to:	
Base location:	Space
Vehicle reusability	Reusable
LEO return:	Aerobrake

* but not preclude the possibility of including one or more of these items in the initial vehicle concept

Figure 1-1.1-1.   Study Groundrules

• First cargo flight	2002
• First piloted flight	2004
• Crew size	4
• Payload delivery capability (manned steady state mission)	13 t
• Payload return capability	500 kg
• Crew support after Lunar landing	48 hours
• Capability to utilize LLOX	
• First Lunar landing on unprepared surface	
• Power and thermal support from Lunar base after 30 days	
• Payload capability for other missions derived from manned steady state capability (13 t)	

Figure 1-1.1-2.   Lunar Mission Groundrules (Option 5)

At the time of the RFP and proposal, human exploration of the Moon and Mars was an evolutionary requirement for the STV. The primary focus of the study was to provide an initial vehicle capable of supporting geosynchronous and Earth escape missions with a long-term objective of evolving the system to support human exploration of the Moon and Mars. During the fall of 1989, the focus of the study turned to the Lunar Exploration Initiative. After working on the Lunar Transportation System (LTS) as the primary mission for STV during the 90-day skunkworks activity, the STV study groundrules were changed to reflect NASA's desire to first determine the best vehicle to support piloted missions to the Moon. The Space Exploration Initiative (SEI) was also coming into focus during this time period.

The LTS was then analyzed to determine the capability to evolve from early upper stage components that could be used to support geosynchronous and planetary delivery missions to the LTS. Space basing, reusability, and use of an aerobrake for return to LEO were all considered within the architecture studies for the LTS.

There are a number of different program groundrules that have been suggested for the Lunar Exploration Initiative. Five different reference approaches (A through E) were included in NASA's 90-day study report. Reference Approach E, which will be referred to as "Option 5" in this report, is used as a groundrule for all lunar vehicle analysis. The highlights of the Option 5 lunar mission groundrules appear in Figure 1-1.1-2. Flight 0, with the payload unloader and attachments, was assumed to be the first cargo flight in the year 2002. The first piloted flight departs 2 years later in 2004. The groundrule for LTS performance capability was to fly 4 crew and 13 metric tons of cargo to the lunar surface in a steady-state mode that does not require reintroduction of space-based hardware. After lunar landing, the vehicle was required to support the crew for up to 48 hours after touchdown and be capable of self support without the assistance of planet surface systems (PSS) for up to 30 days.

Evaluation Criteria. In the early part of the study cost, margins/risk, and evolutionary mission capture were established as the design and trade evaluation criteria. "Benefits to Mars" was later added to complete the set of four evaluation criteria used for the system architecture studies. For the purpose of

the architecture trade studies, weighting factors between the four evaluation criteria were selected as shown below.

Cost	Risks/Margins	Mission Capture	Benefits To Mars
50%	30%	15%	5%

Program Goals and Objectives. The overall objectives for the STV program are broken down into the primary objective and the evolutionary objectives. At the time this final report was written, the primary objective of the STV system was to provide a cost-effective transportation system capable of supporting a human exploration program resulting in a manned outpost on the surface of the Moon. The evolutionary objectives of the program are to provide a cost-effective evolvable space transportation system capable of supporting (1) high-energy upper stage missions (such as geosynchronous and planetary) beginning in 1999 and (2) eventual human exploration resulting in a manned outpost on Mars in the 2017 timeframe.

While the primary and evolutionary program goals and objectives were supplied by Marshall Space Flight Center (MSFC), additional STV program goals and objectives were developed to further guide the system design (Figure 1-1.1-3).

Introducing reusability as, at the minimum, a long-term goal may provide a low-cost, operational upper stage system. Low-risk application of technology was desired to limit front-end funding requirements. With an initial operating capability in 1999, the technology for the initial vehicle concept would be cut off in 1995. Because these goals were established the initial operating capability appears to have moved further out allowing a later technology cutoff date. In any case, the early vehicles should allow for periodic technology upgrades throughout the life of this long-term program.

Especially in the case of a space-based STV system, simple interfaces with independent autonomous verification without the requirement for human intervention is viewed as critical to low-cost operations at the LEO transportation node. In an effort to further minimize the operations costs, operational

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- Provide a reusable transportation system
- Low risk application of mid-90's technology
- Capability for periodic technology upgrades
- Simplified interfaces with autonomous verification
- Design for operations

Figure 1-1.1-3. STV Program Goals and Objectives

• Safe manned operations	• Reliable
• Low life cycle cost	• Low risk
• Evolvable (Mars)	• Simple interfaces
• Flexible (mission capture)	• Commonality with other systems
•Low or no maintenance	• Low investment cost
•Operable	

Figure 1-1.1-4. STV System Design Goals and Objectives

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considerations such as vehicle maintenance and postflight refurbishment should be taken into account during the design phase of the STV system.

Eleven design goals and objectives were established for the STV system (Figure 1-1.1-4). These design goals and objectives are intended to reduce cost and risk and increase margins and safety. The manned system aspects of the STV require that safety considerations be factored into the vehicle design from the start. Almost all the design goals and objectives can be tied to minimizing cost. Evolvability toward the Mars missions will provide the most cost-effective means for the nation to conduct manned exploration of the solar system. Commonality with other already developed space systems could provide cost savings along with reduced maintenance requirements.

Other customer inputs

Design
Reference
Missions

DoD

**Space Exploration
Initiative(SEI)**
- Lunar Initiative - Option 5
- Mars Initiative

CNDB Mission Model
- Regime 1
  - CNDB Version FY89
  - Modified (SRS)
  - Augmented Set
- Regime 2 - Lunar *
- Regime 3 - Mars *

Automated
Geosynchronous
Planetary, etc

* Replaced by SEI models

Figure 1-1.2.1-1. STV Mission Model
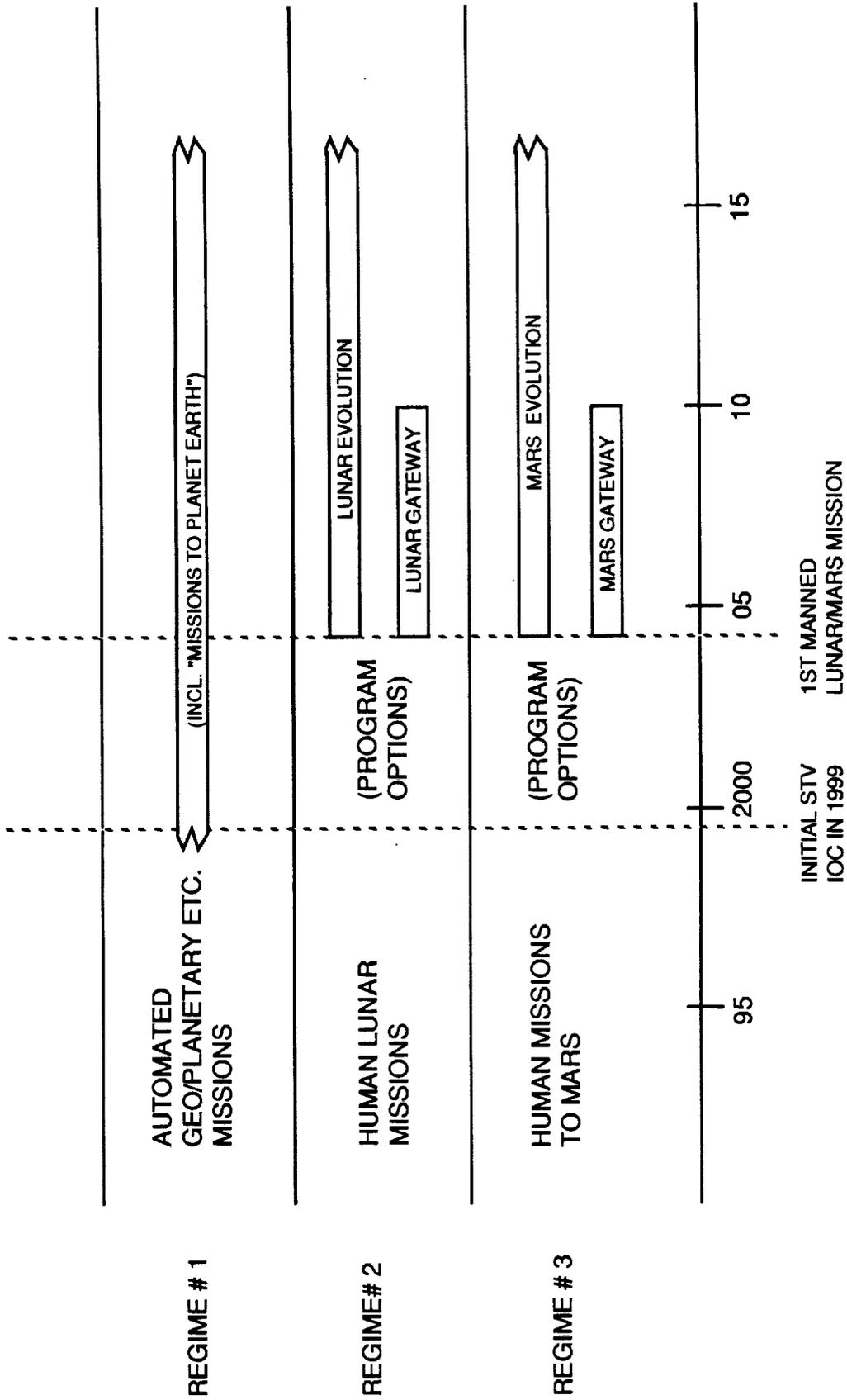
Figure 1-1.2.1-2.   CNDB   Mission   Model   Structure

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AUGMENTATION

- PAYLOAD CHARACTERISTICS DERIVED FOR CANDIDATE TRANS- PORTATION PAYLOAD/MISSIONS
  - REGIME #1: 2 PAYLOADS
    5 MISSIONS

OTHER STV TRANSPORTATION MISSIONS

STV MISSION MODEL REGIME #1

CNDB MODIFICATIONS

- PAYLOAD CHARACTERISTICS MODIFIED FOR PLANNED TRANSPORTATION SYSTEMS
  - REGIME #1: 12 PAYLOADS
    18 MISSIONS

MODIFIED

CNDB VERSION FY89

- PAYLOAD CHARACTERISTICS BASED ON EXISTING TRANSPORATION SYSTEMS
  - REGIME #1: 12 PAYLOADS
    18 MISSIONS

CURRENT

*Figure 1-1.2.1-3. Approach to Regime #1*

form category 2, referred to as the modified set in the mission model documentation. The category 2 mission set is identical to category 1 but contains expanded requirements and capabilities beyond the category 1 missions for specific missions such as the geosynchronous platform delivery for Mission to Planet Earth and Mars Rover Sample Return. Finally, this mission set was augmented by MSFC to form the third category of missions called the augmented set, which includes various manned and unmanned servicing missions, a space nuclear/debris disposal mission, and a Mars or comet sample capsule return. The augmented set missions introduce manned operations, rendezvous and docking, grappling, and retrieval requirements for the STV.

Regime #2 covers manned lunar missions, and Regime #3 includes manned missions to Mars. For the purposes of this study, Regime #2 and #3 missions were replaced by equivalent lunar and Mars missions developed for the SEI in an effort to align the study activities as close as possible to the 90-day study conducted in the fall of 1989. The CNDB lunar mission was similar to SEI Option 1 with two missions to the Moon conducted every year. SEI Option 5, with one lunar mission per year, replaces the Regime #2 lunar scenario in the STV mission model. The lunar mission was also used as the primary mission objective for the STV.

Civil Needs Database FY90. The CNDB FY90 was released toward the end of the study activity. The new version contains 476 events in the base model and 64 events in the expanded model. The expanded model contains the Option 5 lunar program but assumes a separate system architecture with separate lunar transfer vehicles and lunar excursion vehicles. The three concepts remaining at the end of this study all assume a single-stage design with one crew module. A single-stage architecture would require a different mission strategy. In addition, the CNDB assumes a space-based vehicle that drives an entirely different set of launches from the set required by the two remaining ground-based systems.

A brief analysis of the new version of the database showed that all of the non-lunar and Mars missions that drove STV requirements were eliminated. Unmanned polar platform servicing, piloted geosynchronous platform servicing, nuclear debris disposal, and piloted sample capsule return all contained in the

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FY89 model were eliminated in the FY90 version. The new model now extends out to the year 2020 to capture the introduction of the piloted Mars missions. The only traditional upper stage missions that remained in the CNDB FY90 were geosynchronous and planetary delivery (Figure 1-1.2.1-4).

A set of design reference missions were constructed based on the FY89 data (Figure 1-1.2.1-5). The rationale and analysis behind the selection of these mission appears in the Design Reference Missions section of this report. Many of the design reference missions would be eliminated with the introduction of the FY90 data (Figure 1-1.2.1-6).

DoD Missions. DoD mission were also included in establishing a broad set of design reference missions capable of meeting the overall national needs for a large cryogenic upper stage. The mission model data supplied at the beginning of the study included both a constrained and a normal growth model for DoD missions (Figure 1-1.2.1-7). All of the DoD missions were grouped together into a category called geosynchronous and mid-inclination/mid-altitude missions. The constrained model included an average of 11 missions per year between 1995 and 2010. The normal growth model would require an average of 16 mission per year. The mission rate required for either DoD model drove the STV cost analysis.

- Augmented mission set eliminated
  - No unmanned polar platform servicing
  - No nuclear debris disposal mission
  - No manned GEO platform servicing
  - No manned sample capsule return

STV Missions from Base Model	YEAR																			
	01	02	03	04	05	06	07	08	09	10	11	12	13	14	15	16	17	18	19	20
Mars Site Reconnaissance Orbiter		1																		
Mars Relay														2						
Mars Sample Return	1		1	2																
Mars Rover								1				1								
GEO Platform (MTPE)		1																		
TDRSS	1		1		1		1		1											
Lunar Relay										1										

Figure 1-1.2.1-4. Impact of CNDB FY90

DRM	Launch Year												Total
	1999	2000	2001	2002	2003	2004	2005	2006	2007	2008	2009	2010	
L1	0	0	0	1	1	0	0	1	0	0	0	1	4
L2	0	0	0	0	0	1	1	0	1	1	1	0	5
P1	0	0	3	0	1	2	0	0	0	0	0	0	6
G1	0	0	0	0	0	0	2	0	0	1	2	3	8
G2	0	0	0	0	0	0	0	1	1	1	0	0	3
S1	0	0	1	0	1	0	0	1	0	1	0	0	4
T1	0	0	4	4	5	6	6	5	4	5	4	7	50
N1	0	0	0	0	0	0	0	0	0	0	0	1	1
C1	0	0	0	1	0	1	2	0	0	0	0	0	4
Total	0	0	8	6	8	10	11	8	6	9	7	12	85

Figure 1-1.2.1-5. DRMs Derived From CNDB FY89

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DRM	Launch Year												Total
	1999	2000	2001	2002	2003	2004	2005	2006	2007	2008	2009	2010	
L1	0	0	0	1	1	0	0	1	0	0	0	1	4
L2	0	0	0	0	0	1	1	0	1	1	1	0	5
P1	0	0	1	0	2	2	0	0	1	0	0	0	6
G1	0	0	1	1	1	0	1	0	1	0	1	0	6
~~G2~~	0	0	0	0	0	0	0	1	1	1	0	0	3
~~S1~~	0	0	1	0	1	0	0	1	0	1	0	0	4
~~T1~~	0	0	4	4	5	6		5	4	5	4	7	50
~~N1~~	0	0	0	0	0	0	0	0	0	0	0	1	1
~~O1~~	0	0	0	1	0	1	2	0	0	0	0	0	4
Total	0	0	2	2	4	3	2	1	3	1	2	1	21

DRM	Launch Year										Total
	2011	2012	2013	2014	2015	2016	2017	2018	2019	2020	
L1	0	0	0	0	0	0	0	0	0	0	0
L2	1	1	1	1	1	1	1	1	1	1	10
P1	1	0	0	2	0	0	0	0	0	0	3
G1	0	0	0	0	0	0	0	0	0	0	0
Total	2	1	1	3	1	1	1	1	1	1	13

Total 2001 to 2010	21
Total 2001 to 2020	34

Figure 1-1.2.1-6. CNDB FY90 Impact to DRMs

DoD Constrained Scenario

DESCRIPTION	MODE	DESTINATION	95	96	97	98	99	00	01	02	03	04	05	06	07	08	09	10	TOTALS
DoD Constrained Scenario	DEL	GEO & MID-INCLINATION / MID-ALTITUDE	14	5	10	8	12	10	12	12	8		12	11	10	10	11	11	168

STV IOC

DESCRIPTION	MODE	DESTINATION	95	96	97	98	99	00	01	02	03	04	05	06	07	08	09	10	TOTALS
DoD Normal Growth Scenario	DEL	GEO & MID-INCLINATION / MID-ALTITUDE	16	8	14	14	19	17	19	12	13	10	14	13	14	17	18	18	236

STV IOC

DESCRIPTION	MODE	DESTINATION	95	96	97	98	99	00	01	02	03	04	05	06	07	08	09	10	TOTALS
DoD Aggressive Scenario	DEL	GEO & MID-INCLINATION / MID-ALTITUDE									TBD								TBD

STV IOC

Figure 1-1.2.1-7. DoD Mission Model

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1-1.2.2   Design Reference Mission Selection Process

The design reference missions (DRM) selected for the STV are listed in Figure 1-1.2.2-1. The DRMs are divided into two categories: primary and evolutionary missions. The primary DRMs cover a range of lunar missions, both piloted and cargo-expendable. The four lunar DRMs, L1 through L4, are intended to provide sufficient detail to define vehicle and operational concepts for the STV. The lunar DRMs were taken from Option 5 of the Lunar Initiative and were based on an informational data book written by NASA-JSC (Initial Study Period Results Summary - Planet Surface Systems - Conceptual Design and Development Requirements) defining the mission manifest and planetary surface systems to be taken as cargo by the STV. The lunar DRMs provided a basis for vehicle designs that meet the primary objective of the STV program; to provide a transportation system capable of supporting a human exploration program to the Moon.

Nine evolutionary design reference missions were selected in addition to the lunar missions. These nine missions are split between those targeted for backward and forward evolution. The initial missions required before the Lunar Initiative (2002) will be supported by an early version of the STV capable of evolving to the Lunar Transportation System (Figure 1-1.2.2-2). Examples of these sorts of missions include planetary and molniya delivery. The non-lunar missions required after LTS development will involve evolution from the lunar vehicle to a growth vehicle (or vehicle based on LTS components) capable of supporting the new mission requirements.

The DRMs were taken from a number of different sources (Figure 1-1.2.1-1). The goal in creating a set of design reference missions was to capture all the worst case requirements from the large quantity of missions included in the STV mission model in a much smaller and manageable mission set. The design reference missions are not necessarily identical to specific missions in the model but could be a mosaic composed of the driving elements of two or more missions from the model. The following section describes the process used to select the 13 design reference missions in Figure 1-1.2.2-1.

TYPE		DRM	DESCRIPTION	SOURCE
PRIMARY	Two P/A Modules	L1	Lunar-Cargo-Expendable-NASA Ref. Design	SEI
		L2	Lunar-Manned-Steady State-NASA Ref. Design	SEI
	1 P/A	L3	Lunar-Manned-Steady State-Single P/A Design	SEI
	1 Launch	L4	Lunar-Cargo-Expendable–Single Launch	MSFC/SEI
EVOLUTIONARY		P1	Planetary-Unmanned-Delivery	CNDB
		G1	Geosynchronous-Unmanned-Delivery	CNDB
		D1	Molniya-Unmanned-Delivery	DOD
		G2	Geosynchronous-Manned-Servicing	Augmented Set
		S1	LEO Polar-Unmanned-Servicing	Augmented Set
		T1	LEO Space Tug-Unmanned-Delivery/Servicing	MSFC/CNDB
		N1	Nuclear/Debris Disposal-Unmanned-Delivery	Augmented Set
		C1	Sample Capsule Capture/Recovery-Manned	Augmented Set
		M1	Mars-Manned-Steady State	SEI

Figure 1-1.2.2-1. Design Reference Missions
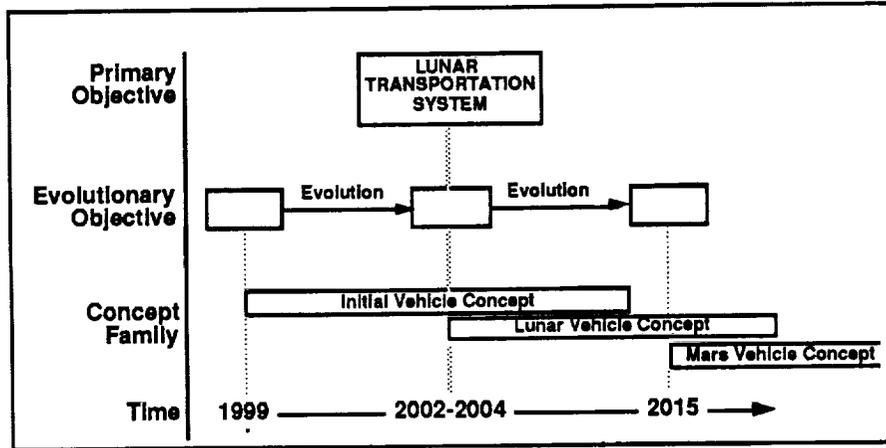
Figure 1-1.2.2-2. Primary and Evolutionary DRM Relationship
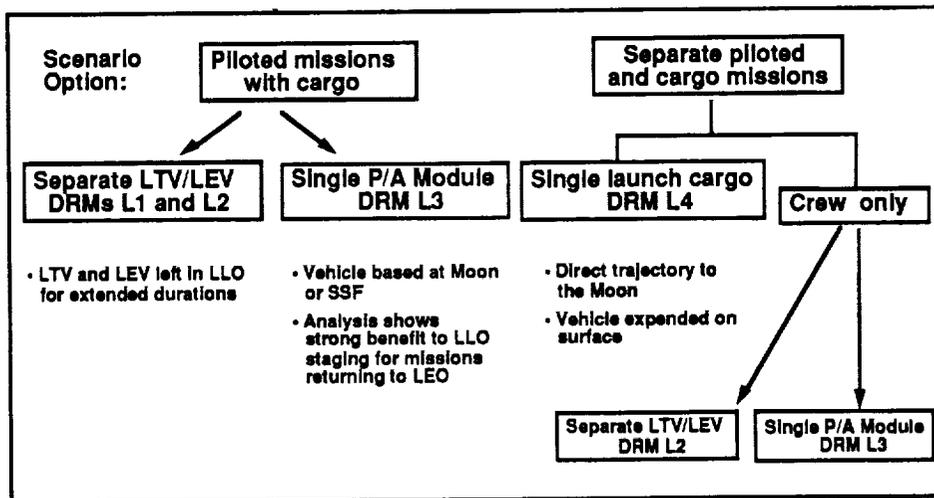
Figure 1-1.2.2-3. Lunar Design Reference Missions

Lunar Design Reference Missions. There are a wide range of missions and systems capable of supporting the Lunar Initiative. As shown in Figure 1.2.2-3 the missions can be divided between options that use a common vehicle to support both piloted and cargo missions and options that use separate vehicles for either the cargo or the piloted missions. The first option encompasses those missions and vehicle designs that include a large cargo-carrying capability while simultaneously shuttling crew to the lunar surface. The skunkworks baseline design, which required that piloted missions flown in steady-state mode are capable of also delivering 13 tons of cargo to the lunar surface, fell into this category. The second option included those missions and vehicle designs that take cargo and crew to the surface using vehicles that are optimized specifically for the piloted missions or the cargo missions.

The combined crew and cargo missions were further subdivided into missions that use separate lunar transfer vehicle (LTV) and lunar excursion vehicle (LEV) similar to the skunkworks baseline and single propulsion/avionics (P/A) module concepts that use one set of engines to provide impulse from LEO, down to the lunar surface, and back to LEO. Design reference mission L1 was specifically designed to characterize an expendable cargo missions that use separate transfer and excursion vehicles. DRM L1 would transport approximately 33 metric tons of cargo to the lunar surface. Design reference mission L2 provided the complementary steady-state manned mission using the same set of vehicle elements. DRM L2 would transport 13 metric tons of cargo along with a crew of four.

DRM 3 was specifically designed to characterize a steady-state single P/A module mission. The DRMs were set up early in the study when the primary evaluation criterion was performance. Analysis indicated that significantly better mission performance could be obtained for piloted missions that return to LEO when the vehicle elements (aerobrake and lunar transit module with storm shelter) and return propellant were staged in low lunar orbit (LLO). Later in the study it was determined that cost factors drive the tradeoff toward the lunar direct trajectory options that avoid staging in LLO. Concepts that met DRM L3 provided good comparison data to the LTV and LEV approach.

At the time the lunar design reference missions were created, the focus was on alternatives to the 90-day reference design. All the lunar DRMs assumed vehicle basing in low Earth orbit with aeroassisted Earth return. Later in the study, ground-based options were also included in the list of viable candidate approaches.

The second scenario option was broken down into cargo only and crew only missions. DRM L4 characterizes a mission using a single launch to orbit and direct transfer to the lunar surface. This option has characteristics similar to the ground-based single-launch system developed at the end of the study. The crew only missions that would accompany these cargo missions were broken down into the same two categories as piloted missions with cargo. Design reference missions L2 and L3 provide sufficient detail to provide the basis for concepts supporting crew only scenario. The DRMs could be applied with the elimination of the cargo transfer.

Mars Design Reference Mission. The Mars design reference mission provides an evolutionary target for the STV system. One typical early manned Mars delivery mission, which captures the overall mission requirements, was selected to provide a basis for the evolutionary STV design for human exploration missions. The Mars design reference mission is a piloted mission with cargo delivery.

A 2015 departure date for the Mars DRM was selected to support technology development readiness and vehicle evolution analysis. No effort was made to recreate the mission analysis already done by NASA and the Code Z STV study. Trajectory and timeline data for the Mars design reference mission were taken directly from these efforts.

CNDB-Derived Design Reference Missions. As discussed previously, the STV study mission model supplied by MSFC was largely based on the civil needs database. The lunar and Mars missions, Regimes #2 and #3 in the CNDB, highlighted above were taken from the latest Option 5 scenario from NASA and replaced the Regime #2 and #3 missions. The following is a description of the mission model analysis conducted on the Regime #1

missions. The analysis shows that all of the missions contained within the STV mission model are represented by the design reference missions.

The basic CNDB mission model (Regime #1) for STV was grouped into three different categories: (1) the CNDB version FY89, (2) the modified CNDB, and (3) the CNDB augmented set (Figure 1-1.2.1-3).

The CNDB version FY89 missions were further broken down into three types: (1) geosynchronous, (2) planetary, and (3) space tug type missions. The individual missions contained within each of these categories are indicated in the Figure 1-1.2.2-4. The geosynchronous delivery missions have been assigned a unique design reference mission designated G1. The planetary missions all fall within the design reference mission P1 description and the Space IR Telescope Facility (SIRTF) mission to boost the payload from one LEO to a slightly higher energy LEO was included in the DRM T1 - space tug mission.

The modified CNDB includes the same CNDB version FY89 missions, but contains expanded payload requirements. The missions with increased payload weight or higher energy final orbit are listed in Figure 1-1.2.2-5 and the modifications or changes are highlighted. All the expanded planetary missions fall within design reference mission P1. The characteristics and requirements of a high Earth orbit (HEO) version for the SIRTF are covered by a combination of DRM P1 to encompass the higher energy requirements and G1 to cover the operational characteristics of transfer from a low circular orbit to a higher circular orbit.

The augmented set for the CNDB includes some significantly new and different types of missions. Figure 1-1.2.2-6 highlights the new or changed missions for the augmented set. Many of the design reference missions were selected based solely on the requirements of the missions contained within the augmented set.

Servicing missions (both manned and unmanned, LEO polar, and geosynchronous orbit (GEO)) are included in the augmented set. DRM G2 represents the manned geosynchronous servicing mission, and DRM S1 covers the unmanned polar platform servicing mission. The unmanned

```
                                                     DRM #
  •  Geosynchronous-Delivery                          G1
     -  Deep Space Relay Satellite
     -  EOTS-1
     -  TD of Large GEO Satellite
     -  GEO Relay
     -  GEO Platform (Mission to planet earth)
  •  Planetary-Delivery                                P1
     -  Planetary-ESA
     -  Solar Probe
     -  Cassini
     -  Mars Rover Sample Return
     -  Comet Nucleus Sample Return
     -  Comet Rendezvous/ Asteroid Flyby
  •  LEO Space Tug                                   T1/G1
     -  Space IR Telescope Facility
```

Figure 1-1.2.2-4. Regime #1 CNDB Version FY89 Missions

```
        New or changed missions introduced by modified CNDB

                                                     DRM #
  •  Planetary-Delivery                               P1
     -  Solar Probe mass increase from 1.0 t to 1.4 t
     -  Mars Rover Sample Return
        -  Mass increased from 3.5 to 4.1 t
        -  500 kg sample return (separate propulsion)
     -  Comet Nucleus Sample Return
        -  Mass increased from 6 to 16 t
        -  Slightly increased C3
  •  High Earth Orbit (34,000 n.mi. circular)        P1/G1
     -  Space IR Telescope Facility
```

Figure 1-1.2.2-5. Regime #1 Modified CNDB Missions

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New or changed missions Introduced by augmented set	DRM #
• Geosynchronous-Servicing	
- Unmanned	G2/S1
- Manned	G2
• Unmanned Polar Platform-Servicing	S1
• Manned Sample Capsule Capture/Recovery	C1
- 3.6 mt manned capsule	
- 500 kg return payload	
- High Earth orbit (elliptical)	
- Missions	
- Comet Nucleus Sample Return	
- Mars Rover Sample Return	
• Space Nuclear/Debris Disposal	N1

Figure 1-1.2.2-6.   Regime #1 CNDB Augmented Set Missions

geosynchronous missions are covered by a combination of G2 and S1. DRM G2 includes the energy requirements required for the unmanned GEO servicing, and DRM S1 covers the unmanned servicing aspects of the mission.

The manned sample capsule capture and recovery mission would be about the same whether returning samples from Mars at the end of a MRSR mission or from a comet nucleus. Design reference mission C1 was included to cover the unique requirements of the sample return missions.

The augmented set also includes a mission to rendezvous with a failed nuclear payload or space debris for transfer out of the Earth/Moon system. Design reference mission N1 was created to be representative of these mission requirements. It is assumed that a nuclear electric propulsion payload has failed while spiraling out of LEO. The STV captures the payload in 700-km circular orbit and transfers the failed vehicle out to a 1.1 AU sun-centered orbit where the STV will be expended.

Mission to Planet Earth. The Augustine advisory committee on the future of the US space program considered the mission to planet Earth (MTPE) to be of very high importance in America's overall space strategy. MTPE is an international program designed to study and understand the physical characteristics of the Earth environment. The 15-year program consists of establishing and maintaining a system of space-based elements for global observation on a continuous basis. In addition to the space-based elements there will be a significant effort to develop in situ data to compare with the space-based measurements.

The focus of the ground systems will be to verify and calibrate the spaceborne instruments and data. This combination of space-based and ground-based observations will ensure that there is a high degree of confidence in the space-based elements. Another key aspect of the ground segment is in the cataloging and dissemination of the information to researchers and users. There will also be a significant amount of theoretical research conducted under this program.

The space-based elements of MTPE can be placed into four categories: Earth probes, the Earth observing system (EOS), geosynchronous platforms, and

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attached payloads. The Earth probes are small free-flying satellites designed for specific, generally short-term, studies. They will be in-orbits optimized to study a particular phenomenon. Launch vehicles for these payloads will be of the Scout and Delta class or multimanifest on the space shuttle. EOS is the cornerstone of the space-based elements. It consists of five polar orbiting platforms provided by NASA, ESA, and Japan. There will also be some synthetic aperture radar spacecraft such as the spaceborne imaging radar facility (SIRF). They will be launched by the shuttle, Titan, or (in the case of ESA) Ariane.

Geosynchronous platforms are also an element of the system. They will provide continuous high-resolution coverage of the Earth. There will also be five platforms in GEO. These are fairly large spacecraft and will require space transportation system (STS)-type launchers.

Attached payloads are similar to the Earth probes in that they are specialized instruments for specific observations. However, these will be attached to the space station or the shuttle and may involve instrumentation testing, calibration, or repeated observations over several years. They will be transported by the shuttle. Only the EOS and the geosynchronous platforms represent potential STV missions.

The polar platforms are the most well defined element of EOS (Figure 1-1.2.2-7). They have completed phase A studies and spacecraft construction is expected to begin in 1991. There will be two US satellites and their expected lifetime is 5 years. The current planning includes a servicing flight after 3 years of operation and a launch by a Titan IV into a sun synchronous orbit. As an STV mission they would be delivered by the shuttle to the LEO transportation node and subsequently delivered to the sun synchronous mission orbit by the STV.

The geostationary platform is in the pre-phase A study phase. Science objectives and instrument complement, along with the subsequent platform requirements are being determined. The geostationary platform (Figure 1-1.2.2-8) provides the ability to observe a specific area of the Earth on a continuous basis. However, they provide a clear view of only the mid-latitudes. They are also at a much greater distance from the surface than the polar platforms. While this orbit gives them a wider field of view it limits their ability to conduct some

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Earth Observing System (EOS)
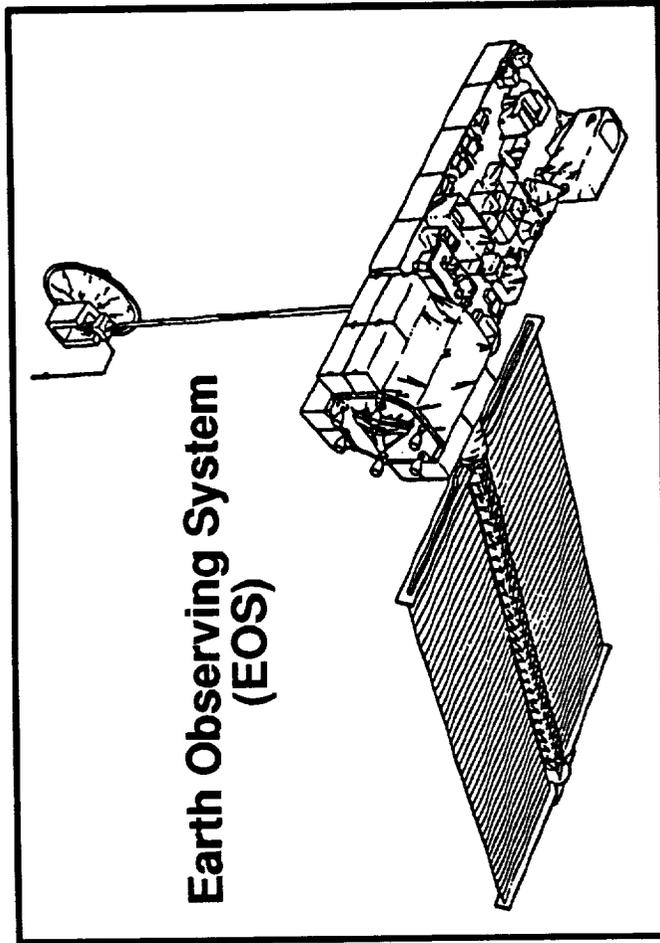
- Instrument Platform located in polar orbit
- 2 US, 2 ESA, 1 NASDA
- Phase A studies complete
- Construction to begin 1991

Flight Frequency

- 2 US satellites (1996,1998)
  - 3 year service, 5 year replace
- ESA (Ariane Launch)
- NASDA disposition unknown (assumed non-STV missions)

Transportation Requirements

- Titan IV Launch is planned
- *13400 kg (29500 pounds)*
- *14.5 ft diameter; 50 ft length–stowed*
- 53.5 ft x 83.5 ft deployed
- 705 x 705 km
- Sun synchronous (98.6°)

Figure 1-1.2.2-7. Mission to Planet Earth Polar Platform

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- **Instrument Platform located at GEO**
- **3 US, 1 ESA, 1 NASDA**
- **Pre-phase A studies**
  - Science objectives
  - Instrument complement
  - Platform requirements

Flight Frequency

- 3 US satellites (2000, 2002, 2004)
- ESA, NASDA disposition unknown (assumed non-STV missions)
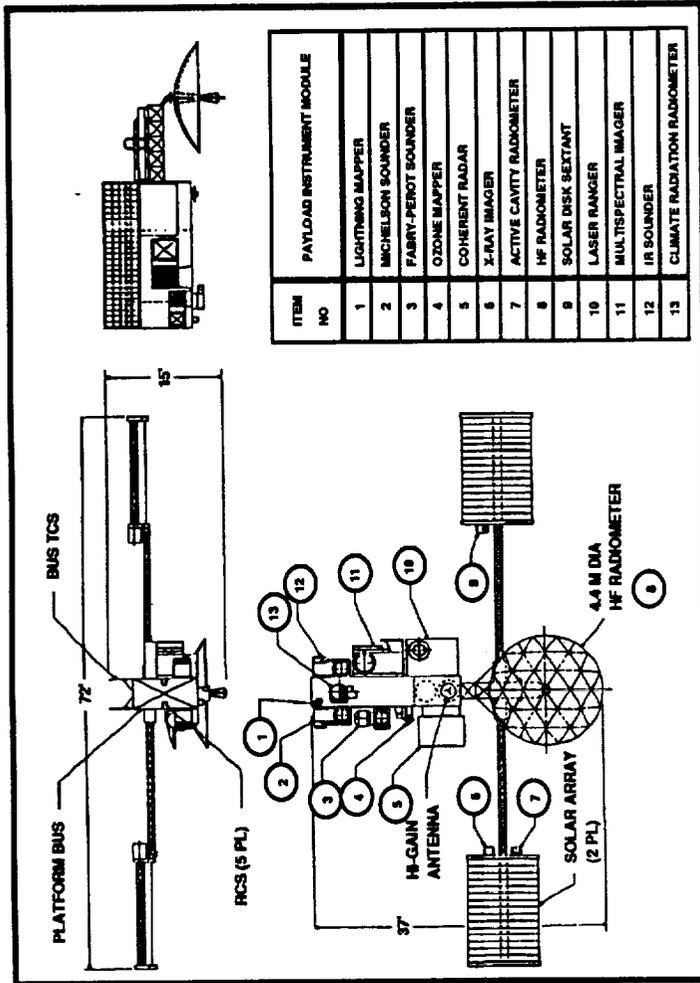
Transportation Requirements

- 10 mt (22000 pounds)*
- 15 ft Diameter x 20 ft Length
- 19,310 x 19,310 nmi altitude
- 0° inclination

* 4.5 mt (10,000 pound) spacecraft also under study–1989 CNDB

ITEM NO	PAYLOAD INSTRUMENT MODULE
1	LIGHTNING MAPPER
2	MICHELSON SOUNDER
3	FABRY-PEROT SOUNDER
4	OZONE MAPPER
5	COHERENT RADAR
6	X-RAY IMAGER
7	ACTIVE CAVITY RADIOMETER
8	HF RADIOMETER
9	SOLAR DISK SEXTANT
10	LASER RANGER
11	MULTISPECTRAL IMAGER
12	IR SOUNDER
13	CLIMATE RADIATION RADIOMETER

Figure 1-1.2.2-8. Mission to Planet Earth Geostationary Platform

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measurements from GEO. This orbit also places more stringent requirements on the instruments, such as increased transmitter power and larger antenna diameter. At the present time there is no upper stage capable of lifting more than about 10,000 pounds to GEO. Consequently, STV is an enabling system for the geostationary platform. Without the STV the platform will have to be limited in mass and, consequently, capability.

The launch schedule for the US polar and geostationary platforms is shown in Figure 1-1.2.2-8. The NASA polar orbiting platform 1 is launched in 1996 with replacement at 5-year intervals. The servicing missions are 3 years after launch of each spacecraft. Platform 2 is launched in 1998 with the same replacement and servicing schedule as platform 1. The geostationary platforms are launched at 2-year intervals beginning in 2000.

DoD Design Reference Missions. The DoD missions were broken down into five major mission categories, which are listed in Figure 1-1.2.2-9. The DoD mission model flight rate was shown in Figure 1-1.2.1-7. A number of the DoD missions are already contained in the DRMs developed to cover civil needs. Because of the smaller energy requirements and dedicated launch vehicle, the global positioning system (GPS) delivery mission was excluded from this analysis. Due to the unique nature and three-burn transfer of typical molniya orbit missions, a molniya orbit DRM was included in the DRM set and given the designation D1.

Other Design Reference Missions. The space-based lunar exploration missions require retrieval of hardware elements and propellant in LEO after boost on an expendable HLLV for transport to a LEO transportation node. Design reference mission T1 has been created to specifically address the applicability of the STV to leave a LEO transportation node, maneuver to the HLLV upper stage containing an SEI payload, grapple and restrain that payload, and return to the LEO node.

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	DRM #
• Geosynchronous delivery	G1
• Molniya delivery	D1
• Global Positioning Satellite (GPS) delivery (Tailored for MLV-II)	N/A
• Polar unmanned servicing	S1
• LEO Space Tug- unmanned-delivery/servicing	T1

Figure 1-1.2.2-9. DoD Design Reference Missions

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1-1.2.3 Design Reference Missions

In section 1-1.2.2 we discussed the selection process that ensured that the set of the 13 design reference missions provides complete coverage of the STV mission model. The following sections discuss the rationale behind the selection of the detailed definition of each design reference mission and defines the each mission in detail. Each DRM description includes discussion of (1) mission objectives, (2) payload requirements, (3) mission profile, and (4) a brief mission timeline corresponding to the numeric designators on the mission profile graphic. Section 1-1.2.4 provides more detailed timeline information.

The DRMs are intended to cover all tentpole or worst case requirements for each mission type (Figure 1-1.2.3-1). The missions are intended to be generic in nature, not tied to any particular design solution, and consistent with the STV study groundrules.

Number	Name	Designator
1	Lunar Cargo - LTV/LEV	L1
2	Piloted Lunar - LTV/LEV	L2
3	Piloted Lunar - Single P/A Module	L3
4	Lunar Cargo - Single Launch.	L4
5	Planetary Delivery	P1
6	GEO Delivery	G1
7	Molniya Deliver	D1
8	Piloted GEO Servicing	G2
9	LEO Polar Servicing	S1
10	LEO Space Tug	T1
11	Nuclear/Debris Disposal	N1
12	Piloted Sample Return	C1
13	Piloted Mars	M1

Figure 1-1.2.3-1. STV Design Reference Missions

Overall Lunar Design Reference Mission Selection Rationale. Lunar Initiative Option 5 was used as a point of departure for the lunar design reference missions. The manifest (Figure 1-1.2.3-2) provided a basic framework including launch dates, cargo manifesting, and reusability assumptions to construct the lunar DRMs.

The Option 5 scenario, or Reference Approach E in the 90-day study, provides a program description of a man-tended lunar base with first flight in 2002. The first piloted flight would depart in 2004. The four cargo flights are all expended on the lunar surface. Piloted vehicles are reusable with a phased approach to increase reusability gradually over a time period covering the first 14 mission of the lunar program. The typical stay time for the the crew is 6 months. This scenario assumes one lunar mission per year. The manifest continues out to the year 2026 at which time a Mars base should be in full operation.

One of the most important interfaces for the STV system is with the lunar base or planet surface systems (PSS). At first the PSS elements are integrated with the transportation system to create a cargo interface. After offloading on the lunar surface those cargo elements become active and support the transportation elements as surface systems, thus creating a second interface.

Groundrules were taken from the MASE documents describing the Option 5 requirements (Figure 1-1.2.3-3). The Option 5 scenario defines stay times for the lunar crew on the lunar surface. The Lunar Initiative, broken down into three major phases, specifies from 30 to as much as 600 days stay time.

The detailed PSS description provided to the STV study contractors in December 1989 specifies a lunar lander servicer (LEV servicer) capable of providing up to 9,000 watts of power and 3,000 watts average heat rejection. It is also assumed that the servicer is only available after 30 days on the surface. There are a number of early flights before the servicer is flown to the lunar surface as a STV payload.

In addition to these groundrules, it is assumed that the cargo elements will have mass property characteristics consistent with the lander dynamic constraints. It

Date	Flight #	Mission Type	LTV/LEV
2002	0	Cargo	Expended
2003	1	Cargo	Returned/Expended
2004	2	Piloted	Delivery
2005	3	Piloted	2
2006	4	Cargo	Expended
2007	5	Piloted	Replacement
2008	6	Piloted	2
2009	7	Piloted	3
2010	8	Cargo	Expended
2011	9	Piloted	Replacement
2012	10	Piloted	2
2013	11	Piloted	3
2014	12	Piloted	4
2015	13	Piloted	5
2016	14	Piloted	Replacement
2017	15	Piloted	2
2018	16	Piloted	3
2019	17	Piloted	4
2020	18	Piloted	5
2021	19	Piloted	Replacement
2022	20	Piloted	2
2023	21	Piloted	3
2024	22	Piloted	4
2025	23	Piloted	5
2026	24	Piloted	Replacement

Figure 1-1.2.3-2. Reference Lunar Initiative Program

	SOURCE
• PSS shall support LEV on Lunar surface for:	
• 30 days to 3 months during emplacement phase	Option 5
• 6 months to 600 days during consolidation phase	Option 5
• 30 days to 6 months during operational phase	Option 5
• Lander support includes: (average/peak)	
- 9/12 kWe watts power	PSS-JSC
- 3/5 kWt rejection	PSS-JSC
• Each pallet will have mass properties consistent with vehicle dynamic requirements	BOEING
• PSS payloads will be capable of mounting on a standard interface pallet	BOEING

PSS - Planetary surface systems

Figure 1-1.2.3-3. PSS Support Groundrules

is also assumed that the payloads are mounted on standard pallets that provide the transportation elements with a common interface.

The items listed in Figure 1-1.2.3-4 are the major cargo elements for the Option 5 Lunar Initiative as described in the Johnson Space Center (JSC) PSS documentation. These items were selected from the total manifest because of their large size, weight, and resource requirements. The payload unloader is by far the largest payload to be carried by the STV. This payload, along with the rest of the Flight 0 manifest, was selected for the DRM - L1 payload. The largest payload identified for the piloted missions is the submillimeter infrared interferometer science experiment. The PSS payloads have evolved since the time this initial description was assembled. Figure 1-1.2.3-5 includes some of the PSS payload changes that occurred in the spring and summer of 1990 and was ultimately used as a reference payload description for the lunar vehicle designs.

A checklist of items that the lunar design reference missions must take into account is shown in Figure 1-1.2.3-6. The categories included cover the major areas that could drive the design of the STV. Each item designates the design reference mission that provides coverage of the relevant requirements and factors.

1-1.2.3.1 Lunar Cargo (LTV/LEV) DRM - L1

Mission Objectives. The overall objective of the expendable cargo lunar mission, DRM-L1, is to deliver a one-piece cargo to the lunar surface (Figure 1-1.2.3.1-1). The mission is similar to Flight 0 of lunar Option 5. This design reference mission assumes the use of a separate LTV and LEV. The LEV is expended on the lunar surface, and the LTV is expended in a 300-km circular LLO. The mission leaves Space Station Freedom in the year 2002.

The LEV must make an autonomous landing on an unprepared surface without use of any landing aids and must provide for the autonomous unloading of the payload on the lunar surface. Many self unloading concepts were developed during the study and are highlighted in the Interim Review Briefing #5 in the Payload/PSS Interfaces section.

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 1-1.2.3-4. PSS Cargo Description

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- Cargo mission assumes Flight 0 of 90 day study manifest
- Piloted mission assumes Flight 2 of 90 day study manifest

Quantity	Item	Total Mass (kg)	Volume (m**3)	Length (m)	Width (m)	Height (m)	Source/Comments
	Piloted Mission						
1	Tools & Implements	250.	1.5	(3)	(3)	(3)	A
5	Lunar Surface EMU	875.	.6 (1)	.8 (1)	.8 (1)	2.1 (1)	A
0.2	4/6 Resupply + Half Pallet	625.	7.8 (2)	(3)	(3)	(3)	A/Total mass = 3120 kg
12	Spares	3000.	37.5 (2)	(3)	(3)	(3)	A/One set of spares = 250 kg
1	Geologic Exp Equipment	100.	1.2 (2)	(3)	(3)	(3)	B
1	Geophysical Station	100.	1.2 (2)	(3)	(3)	(3)	B
1	Laser Retroreflector	50.	0.6 (2)	(3)	(3)	(3)	B
	TOTAL	5000.					
	Cargo Mission						
1	Payload Unloader	10000.	240.0	10.0	10.0	20.0	A
1	Payload Unload Attachments	5500.	32.0	4.0	4.0	2.0	A
1	Excavation Pyrotechnics	3200.	1.8	(3)	(3)	(3)	A
1	Communication Equipment	820.	15.0	(3)	(3)	(3)	A
1	Unpress. Manned/Robotic Rover	1277.	11.0	4.5	2.0	1.5	A
	TOTAL	20797.					

A - Element/Systems Data Base - Version 90.0 - 4/13/90 Release
B - PSS Initial Study Period Results Summary - Reference Mission - Option 5 - JSC - Fall 89
(1) - Characteristics of single item
(2) - Assumes a payload density of 80 kg/m**3 (typical)
(3) - When the PSS documents do not specify, the volume is assumed to be cubic

Figure 1-1.2.3-5. Reference Lunar P/Ls for Vehicle Design

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- DRMs compatible with NASA reference program description
- DRMs consistent with reference program groundrules
- The Lunar DRMs should cover: **DRM #**
  - Piloted missions   L2/L3
  - Cargo missions   L1/L4
  - Largest payload mass   L1
  - Largest payload volume   L1
  - Largest payload power requirements   L2
  - In orbit propellant transfer   L2/L3
  - Aerobrake reentry   L2/L3
  - Autonomous landing without navigation aids   L1/L4
  - Landing on unprepared surface   L1/L4
  - Autonomous payload unloading   L1/L4
  - Most stressing ETO scenarios   L1/L4
  - Longest duration flight   L2/L3
  - Most stressing operational scenario   L2/L3
  - Most stressing ground operations scenario   L1/L4

Figure 1-1.2.3-6. Lunar DRM Selection Criteria

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| First flight: | 2002 |
| Initial orbit: | Space Station Freedom |

First flight: 2002
Initial orbit: Space Station Freedom
Similar to Option 5 - Flight 0
Cargo only mission
Two HLLV flights for propellant transfer to LEO
One HLLV flight for transportation hardware
Either one STS or one HLLV flight for cargo launch (No crew)
LLO at 300 km circular
Autonomous landing on unprepared surface with no landing aids
Autonomous payload unloading on Lunar surface
LEV expended on lunar surface (No ascent)

Figure 1-1.2.3.1-1. Lunar DRM - L1 Mission Objectives

ITEM	REQUIREMENT
• Physical description (Option 5 - Flight 0)	
• Payload mass	31.6 t (26.3 + 20% margin)
• Payload volume	310.0 m**3 (Flight 0 manifest)
• Payload center of gravity	Center of envelope
• Payload packaging	Payload Unloader mounted on centerline
• Resource Requirements	
• Power	500 watts
• Thermal protection	Maintain benign attitude during transit
• Data communications	200 kbps telemetry
• Support services	
• None	

Figure 1-1.2.3.1-2. Lunar DRM - L1 Payload Objectives

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Two heavy lift launch vehicle (HLLV) flights are available for propellant transfer to LEO, and one HLLV flight is available for the transportation hardware. Either one STS or one HLLV flight is used for launching the cargo into LEO.

The launch vehicle assumptions evolved over the period of performance of the study. Early in the study, when these design reference missions were written, the launch strategy was based on a Shuttle-C type capability to LEO of 71 tons. For most of the rest of the study the launch vehicle capability was considered to be a variable.

Payload Requirements. The payload requirements are derived from Flight 0 of Option 5. The payload mass is 31.6 tons (26.3 tons plus 20%) with a volume of 310.0 cubic meters (Figure 1-1.2.3.1-2). The cargo-carrying capability of the missions that use a single vehicle to fly both piloted and cargo missions was derived from the vehicle design that is capable of supporting piloted missions with a 13-ton cargo capability. The design reference mission assumed the actual payload manifested on Flight 0. With the 20% growth margin the requirement is very similar to the 33 tons specified in the overall Option 5 manifest.

The payload center of gravity is assumed to be at the center of the physical envelope of the payload and mounted on the centerline of the LEV. The payload requires 500 watts of power and 200 Kbps of data communications for telemetry. The thermal protection required of the transfer vehicle is to maintain a benign attitude with respect to the sun (e.g., shaded or rotisserie). The LEV is assumed to be capable of self unloading of the PSS payload.

Mission Profile. Figure 1-1.2.3.1-3 illustrates the mission profile for the expendable cargo lunar delivery mission. The numbered events correspond to the DRM L1 mission timeline (Figure 1-1.2.3.1-4). The LTV/LEV/cargo combination is assembled and verified at the Space Station. The vehicle separates from the Space Station, coasts to the injection point, and performs the translunar injection (TLI) burn. After coasting and correcting the trajectory with several small attitude control system (ACS) firings, the main propulsion system (MPS) performs the lunar orbit injection (LOI) burn. At the proper point in
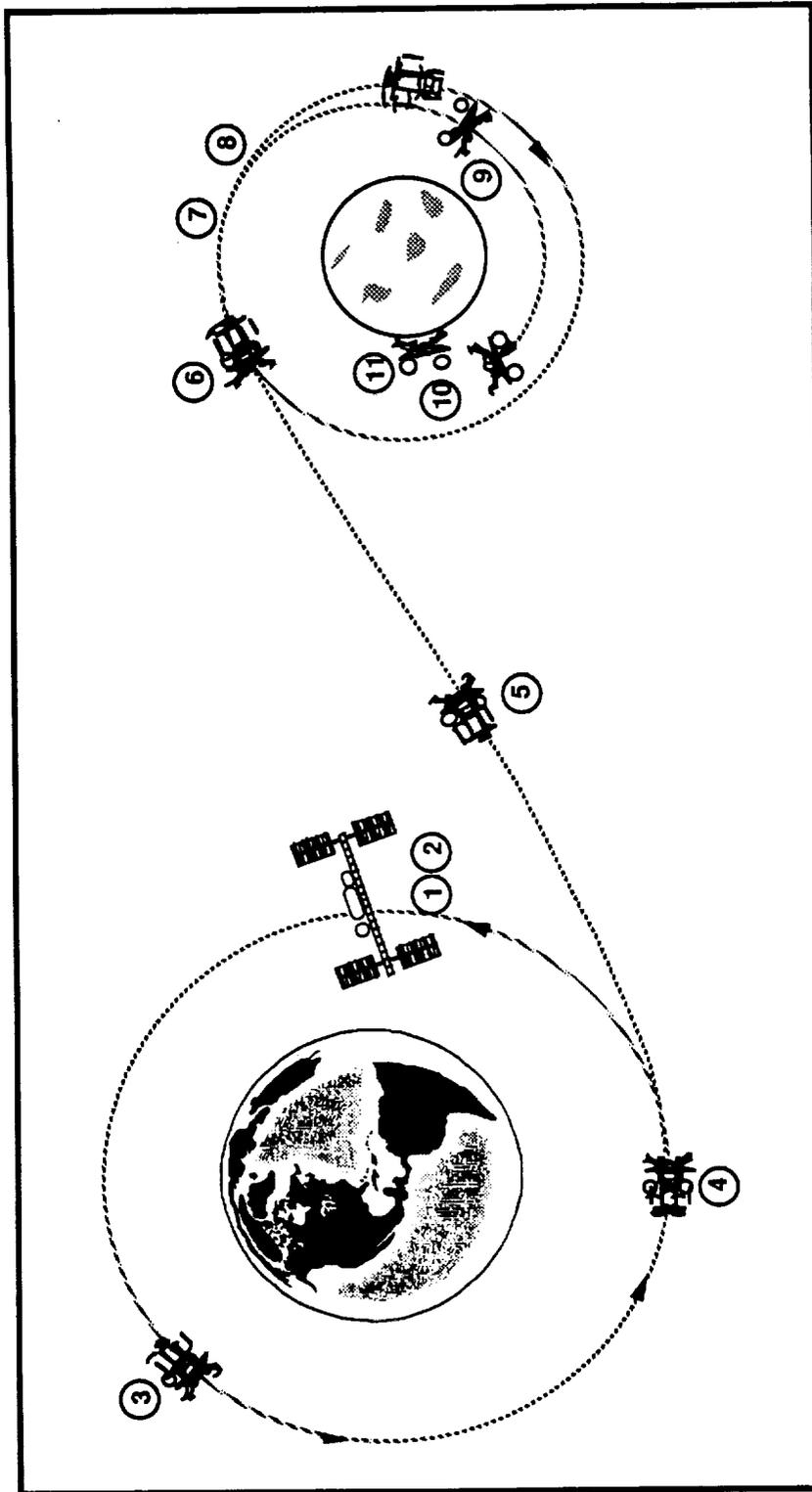
Figure 1-1.2.3.1-3. Lunar DRM - L1 Mission Profile

LLO, the LEV and cargo separate from the LTV and descend to the lunar surface. The LTV is expended in LLO and the LEV is expended on the lunar surface. The LTV would require deorbit to the lunar surface to avoid a space debris problem in orbit around the Moon.

Mission Timeline. The mission timeline for the expended cargo lunar delivery mission, including changes in velocity (ΔV) and event durations (ΔT) is shown in Figure 1-1.2.3.1-4. The expendable mission is concluded after the cargo is unloaded.

1-1.2.3.2 Piloted Lunar (LTV/LEV) DRM - L2

Mission Objectives. The overall objective of the steady-state manned lunar mission, DRM-L2, is to deliver to the lunar surface a crew of four with a two-piece cargo and return the crew and a small payload to the Space Station (Figure 1-1.2.3.2-1). The first flight of this type in the Option 5 manifest is in 2004 and the maximum flight rate is one flight per year. It is assumed that two cargo expendable missions precede the first piloted mission. The piloted mission vehicles are reusable and must be returned to LEO and LLO.

The vehicle is integrated at the Space Station using the same launch strategy as design reference mission L1. The LTV rendezvous and docks with the LEV that is based in LLO, where propellant, payload, and crew are transferred to the LEV.

The LEV must be capable of a 180-day surface stay, with crew support with no assistance from the planet surface systems for up to 2 days. The LTV must be capable of returning the crew of four and a 500-kg payload to the Space Station after a 6-month storage period in LLO. In addition, the transfer vehicle design must be capable of supporting LEV replacement flights involving emplacement of a new LEV in LLO.

Two HLLV flights are available for propellant transfer to LEO, and one STS flight is used for launching the crew and cargo into LEO. The LTV is assumed to be a reused vehicle based at the Space Station. The Option 5 Lunar Initiative introduces use of lunar LOX generated on the surface after approximately 10

Event	Δ V (m/s)	Δ T (hrs)		Event	Δ V (m/s)	Δ T (hrs)
1) Vehicle assembly/verification		121 days	7)	ACS LEV separation	12	0.0
2) ACS separation	3	0.0	8)	MPS LEV deorbit burn	60	0.1
3) ACS coast	3	0.8	9)	MPS Lunar Descent	1,740	1.0
4) MPS TLI burn	3,300	0.3	10)	MPS Landing	200	0.1
5) ACS coast/ corrections	10	84.0	11)	Cargo offload	0	1.0
6) MPS LOI burn	1,100	0.1				

ACS - attitude control system
MPS - main propulsion system

Figure 1-1.2.3.1-4. Lunar DRM - L1 Mission Timeline

First flight:	2004
Flight rate:	One flight per year
Initial orbit:	Space Station Freedom

Manned mission - crew of four
Two HLLV flights for propellant transfer to LEO
One STS launch for crew/cargo
LLO node at 300 km circular
Rendezvous/docking with LEV in LLO
LLO propellant/payload/crew transfer
180 day surface stay (LEV supports crew for 48 hours)
No LEV surface support available for first 30 days
LTV/crew/500 kg cargo returned to SSF
LTV in LLO for 6 months
Configuration capable of supporting replacement missions
Design does not preclude use of LLOX later in program

Figure 1-1.2.3.2-1. Lunar DRM - L2 Mission Objectives

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years of operation. The goal was to develop vehicles that do not preclude the capability of using LOX at a later date.

Payload Requirements. The payload has a mass (in addition to the loaded crew module) of 13.0 tons, a volume of 200 cubic meters, and is split into two pieces (Figure 1-1.2.3.2-2). As stated in DRM L1, the piloted mission, with a 13-ton cargo capability, was used to size the vehicle. The cargo expendable capability was then derived from piloted vehicle sizing. The payload center of gravity is assumed to be at the center of the physical envelope of the payload, and the payload is mounted on the sides of the LEV. Each of the two payloads requires 500 watts of power and 200 Kbps of data communications for telemetry. The thermal protection required from the transfer vehicle during lunar transit is to maintain a benign attitude with respect to the sun (e.g., shaded or rotisserie). The LEV will provide assistance in unloading the payload, if necessary, and support the crew for 2 days after landing on the lunar surface.

Mission Profile. Figure 1-1.2.3.2-3 illustrates the mission profile for the steady-state manned lunar mission DRM-L2. The numbered events correspond to the mission timeline found in Figure 1-1.2.3.2-4. The LTV/crew module/cargo combination is assembled and verified at Space Station Freedom. The vehicle separates from the Space Station, coasts to the injection point, and performs the TLI burn. After coasting and correcting the trajectory with several small ACS firings, the MPS performs the LOI burn. The LTV rendezvous and docks with the LEV in LLO, where propellant, payload, and crew are transferred to the LEV. At the proper point in LLO, the LEV separates from the LTV and descends to the lunar surface. After landing, the crew and cargo are offloaded and the LEV is stored on the lunar surface for 6 months. At the end of the 6-month stay, the LEV ascends with the crew into LLO, where it docks with the LTV and transfers the crew and any return payload. The LTV performs the trans-Earth injection (TEI) burn transferring the vehicle into a return trajectory to Earth. Several trajectory correction maneuvers are made during the coast back to Earth. The LTV then aerobrakes into Earth orbit and circularizes into a 407-km circular orbit. The transfer vehicle rendezvous with the Space Station.

Mission Timeline. Figure 1-1.2.3.2-4 illustrates the mission timeline for the steady-state manned lunar mission, including changes in velocity (ΔV) and

ITEM	REQUIREMENT
• Physical description - cargo	
• Payload mass	13.0 metric tons
• Payload volume	2 x 100 m3 (Submillimeter (IR) interf Elts)
• Payload center of gravity	Center of envelopes
• Payload mounting/packaging	None
• Resource Requirements - cargo	
• Power	2 x 500 watts
• Thermal protection	Maintain benign attitude during transit
• Data communications	200 kbps telemetry
• Physical description - crew modules (LTCM/LECM)	
• Module mass	7570/3580 kg
• Dimensions per STV Orient. Brief. (Dec 1989)	
• Resource Requirements - crew module	
• Power	Power source within module
• Thermal	Thermal rejection cap within module
• Support services	
• Support crew for 2 days after landing	

Figure 1-1.2.3.2-2. Lunar DRM - L2 Payload Requirements
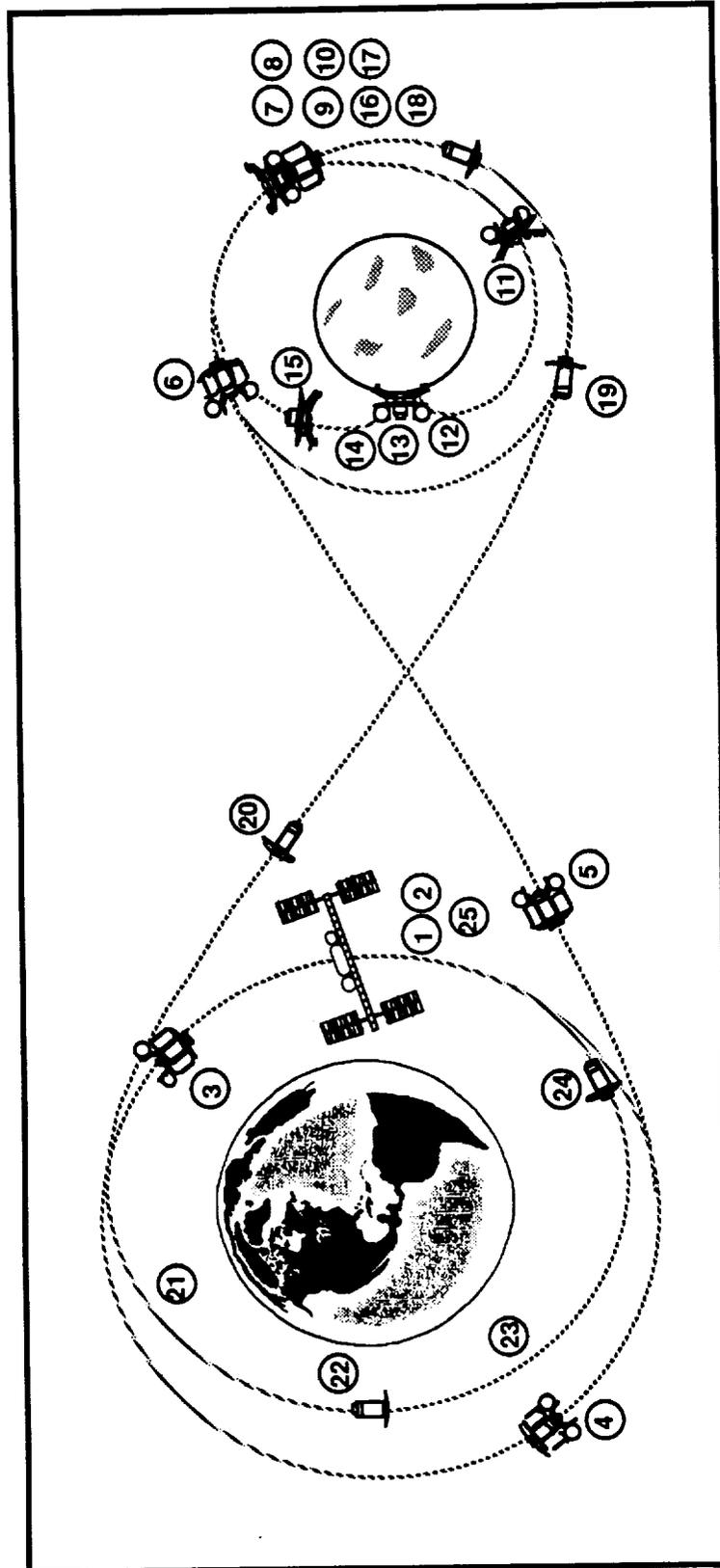
Figure 1-1.2.3.2-3. Lunar DRM - L2 Mission Profile

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Event	ΔV (m/s)	ΔT (hrs)	Event	ΔV (m/s)	ΔT (hrs)
1) Vehicle assembly/verification		121 days	14) Surface storage of LEV	0	6 months
2) ACS separation	3	0.0	15) LEV ascent with crew	1900	2.0
3) ACS coast	3	0.8	16) LTV rendezvous/docking	12	2.0
4) MPS TLI burn	3,300	0.3	17) Crew/payload transfer	0	2.0
5) ACS coast/corrections	10	84.0	18) LTV/LEV separation	12	0.0
6) MPS LOI burn	1,100	0.1	19) MPS TEI burn	1,100	0.1
7) LEV rendezvous/docking	12	2.0	20) ACS coast/corrections	10	84.0
8) Propellant, cargo, and crew transfer		4.0	21) ACS pre-entry correction	8	0.1
9) ACS LEV separation	12	0.0	22) Aeroassist maneuver	0	0.1
10) MPS LEV deorbit burn	60	0.1	23) ACS post-aero correction		0.1
11) MPS Lunar descent	1,740	2.0	24) MPS orbit circularization	310	0.1
12) MPS Landing	200	0.1	25) SSF rendezvous/docking	12	1.0
13) Cargo offload	0	1.0			

ACS - attitude control system
MPS - main propulsion system

Figure 1-1.2.3.2-4. Lunar DRM - L2 Mission Timeline

event durations (ΔT). The ΔV's were supplied by MSFC and correspond to the identical trajectories used during the 90-day study. More detailed mission timelines can be found in section 1-1.2.4.

1-1.2.3.3   Piloted Lunar (Single P/A Module) DRM - L3

Mission Objectives. The mission objectives of the single propulsion/avionics (P/A) module design manned lunar mission, DRM-L3, are to deliver to the lunar surface a crew of four and a two-piece cargo and return the crew and a small amount of cargo to the Space Station. A separate design reference mission for a cargo only expendable mission with the single P/A module was not created because DRM L2 has enough commonality to adequately represent the requirements. The L3 objectives are very similar to L2.

The first flight opportunity for this mission is in 2004 and the maximum flight rate is one flight per year (Figure 1-1.2.3.3-1). The mission is integrated at and started from the Space Station. The vehicle uses one P/A module to perform the entire mission. The transfer vehicle is also used as the lunar surface excursion vehicle. A trade exists concerning use of a LLO versus a direct descent to the surface. Only one space-based vehicle is required to support the scenario. The vehicle is based at Space Station between missions and on the lunar surface during a mission. The design reference mission was constructed early in the study. Based on vehicle performance alone, use of LLO was included in the DRM. Later in the study, when system architecture studies were conducted on the basis of cost, margins/risk, mission capture, and benefits to Mars, cost drove the trade away from use of LLO for staging hardware on the way down to the lunar surface.

Two HLLV flights are available for propellant transfer to LEO, and one STS flight is used for launching the crew and cargo into LEO.

Payload Requirements. The payload has a mass (in addition to the loaded crew module) of 13.0 tons, a volume of 200 cubic meters, and is split into two pieces. The payload center of gravity is assumed to be at the center of the physical envelope of the payload, and the payload is mounted on the sides of the LEV (Figure 1-1.2.3.3-2).

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First flight:	2004

First flight: 2004
Flight rate: One flight per year
Initial orbit: Space Station Freedom
Manned mission - crew of four
Two HLLV flights for propellant transfer to LEO
One STS launch for crew/cargo
LLO node at 300 km circular
LLO propellant transfer
180 day surface stay (Lander supports crew for two days)
No surface support available for first 30 days
Crew and 500 kg cargo returned to SSF
Vehicle/mission uses one P/A module from SSF to LS and back to SSF
Design does not preclude use of LLOX later in program
No LEV waiting in LLO

Figure 1-1.2.3.3-1. Lunar DRM - L3 Mission Objectives

ITEM	REQUIREMENT
• Physical description - cargo	
• Payload mass	13.0 metric tons
• Payload volume	2 x 100 m3 (Submillimeter (IR) Interf Elts)
• Payload center of gravity	Center of envelopes
• Payload mounting/packaging	None
• Resource Requirements - cargo	
• Power	2 x 500 watts
• Thermal protection	Maintain benign attitude during transit
• Data communications	200 kbps telemetry
• Physical description - crew modules (LTCM/LECM)	
• Module mass	7570/3580 kg
• Dimensions per Orientation Briefing (Dec)	
• Resource Requirements - crew module	
• Power	Power source within module
• Thermal	Thermal rejection cap within module
• Support services	
• Support crew for 2 days after landing	

Figure 1-1.2.3.3-2. Lunar DRM - L3 Payload Requirements

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Mission Profile. Figure 1-1.2.3.3-3 illustrates the mission profile for the single P/A module design manned lunar mission, DRM-L3. The numbered events correspond to the mission timeline shown in Figure 1-1.2.3.3-4. The LTV/crew module/cargo combination is assembled and verified at the Space Station. The vehicle separates from the Space Station, coasts to the injection point, and performs the TLI burn. After coasting and correcting the trajectory with several small ACS firings, the MPS performs the LOI burn. In LLO, the aerobrake and propellant tanks for Earth return are separated and the vehicle descends to the lunar surface. After landing, the crew and cargo are offloaded and the vehicle is stored on the surface for 6 months.

At the end of the 6-month stay, the vehicle ascends with the crew into LLO, where it docks with the aerobrake and propellant tanks. Fuel is pumped from the on-orbit tanks into the core tanks integral to the transfer vehicle. The LTV performs the TEI burn injecting the vehicle into a return trajectory to Earth. Several trajectory correction are made during the 3-day coast back to Earth. The LTV aerobrakes into Earth orbit and circularizes at 407 km. The transfer vehicle coasts to rendezvous with the Space Station.

Mission Timeline. The mission timeline for the single P/A module design manned lunar mission, including ΔV's and ΔT's are shown in Figure 1-1.2.3.3-4.

1-1.2.3.4 Lunar Cargo (Single Launch) DRM - L4

Mission Objectives. The mission objective of the single launch cargo lunar mission, DRM-L4, is to deliver to the lunar surface the largest possible cargo, with all transportation hardware, cargo, and propellant launched on one HLLV flight. The overall mission objectives are shown in Figure 1-1.2.3.4-1.

This design reference mission is modeled after the first Option 5 cargo mission in 2002. The HLLV used as the reference early in the study was the Shuttle-C with a payload capacity of 71 tons and a 4.6-meter-diameter fairing. Later in the study a number of options were explored including launch vehicles capable of placing 260 metric tons of payload in LEO. All vehicle elements are expended and the vehicle must make an autonomous landing on an unprepared surface
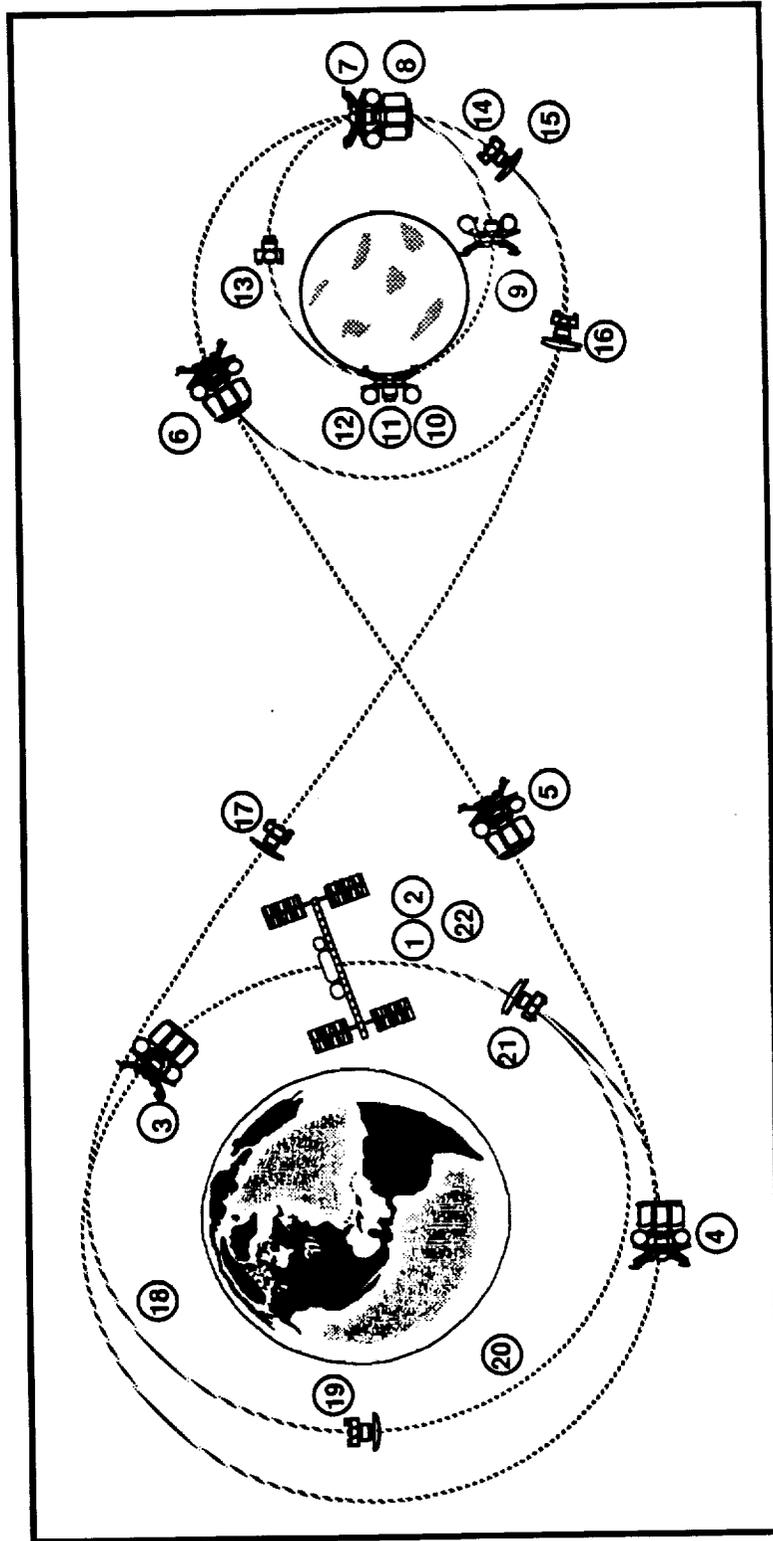
Figure 1-1.2.3.3-3. Lunar DRM - L3 Mission Profile

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Event	ΔV (m/s)	ΔT (hrs)	Event	ΔV (m/s)	ΔT (hrs)
1) Vehicle assembly/verification		121 days	12) Surface storage of LEV	0	6 months
2) ACS separation	3	0.0	13) MPS Ascent with crew	1900	2.0
3) ACS coast	3	0.8	14) ACS rendezvous/docking	12	2.0
4) MPS TLI burn	3,300	0.3	15) Propellant transfer	0	2.0
5) ACS coast/corrections	10	84.0	16) MPS TEI burn	1,100	0.1
6) MPS LOI burn	1,100	0.1	17) ACS coast/corrections	10	84.0
7) ACS brake/tanks separation	12	0.0	18) ACS pre-entry correction	8	0.1
8) MPS deorbit burn	60	0.1	19) Aeroassist maneuver	0	0.1
9) MPS lunar descent	1,740	2.0	20) ACS post-aero correction		0.1
10) MPS landing	200	0.1	21) MPS orbit circularization	310	0.1
11) Cargo offload	0	1.0	22) ACS rendezvous/docking at SSF	12	1.0

ACS - attitude control system
MPS - main propulsion system

Figure 1-1.2.3.3-4. Lunar DRM - L3 Mission Timeline

First flight:	2002
Initial orbit:	LEO
Cargo only mission	
All transportation hardware, cargo, propellant launched on one HLLV	
Payload capability driven by launch vehicle capability	
Lander expended on lunar surface	
Autonomous landing on unprepared surface with no landing aids	

Figure 1-1.2.3.4-1. Lunar DRM - L4 Mission Objectives

without any landing aids in place. The DRM also assumes a single propulsion module.

Payload Requirements. The payload mass will be derived from the launch and transfer vehicle capabilities (Figure 1-1.2.3.4-2). The payload size will be limited by a 4.6- by 25-meter launch vehicle fairing (old Shuttle-C dimensions). Larger shroud sizes were also analyzed later in the study. The payload center of gravity is assumed to be at the center of the physical envelope of the payload. The payload requires 500 watts of power and 200 Kbps of data communications for telemetry. The thermal protection required of the transfer vehicle is to maintain a benign attitude with respect to the sun (e.g., shaded or rotisserie).

Mission Profile. The single launch mission profile for DRM-L4 is shown in Figure 1-1.2.3.4-3. The numerical sequence of events are tabulated in the DRM L4 mission timeline (Figure 1-1.2.3.4-4). The vehicle/cargo combination is launched directly from Earth on one HLLV flight. The vehicle separates from the launch vehicle in LEO, coasts to the injection point, and performs the TLI burn. After coasting and correcting the trajectory with several small ACS firings, the vehicle descends directly to the lunar surface.

Mission Timeline. Figure 1-1.2.3.4-4 illustrates the mission timeline for the single launch cargo lunar mission, including ΔV's and ΔT's. This mission is the least complicated of all lunar design reference missions. A vehicle that uses the same set of engines for TLI and lunar descent was assumed. Other architectures that drop a full stage after the TLI burn could also be considered. The mission timeline is very similar to Surveyor.

1-1.2.3.5 Planetary Delivery Design Reference Mission - P1

Selection Rationale. An analysis was conducted on all of the planetary delivery missions listed in STV mission model to determine the worst case characteristics. The planetary design reference mission should include the highest orbital energy requirements of all the planetary missions in the STV mission model.

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ITEM	REQUIREMENT
• Physical description	
• Payload mass	Derived from launch vehicle capability
• Payload volume	Entire vehicle must fit within launch vehicle fairing
• Payload center of gravity	Center of envelope
• Resource Requirements	
• Power	500 watts
• Thermal protection	Maintain benign attitude during transit
• Data communications	200 kbps telemetry

Figure 1-1.2.3.4-2. Lunar DRM - L4 Payload Requirements
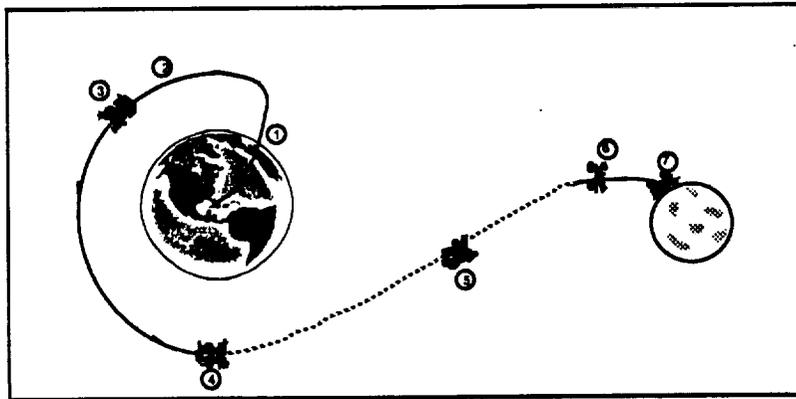
Figure 1-1.2.3.4-3. Lunar DRM - L4 Mission Profile

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Event	ΔV (m/s)	ΔT (hrs)
1) Launch		0.2
2) ACS separation	3	0.0
3) ACS coast	3	0.8
4) MPS TLI burn	3,300	0.3
5) ACS coast/ corrections	10	84.0
6) MPS Lunar Descent	2,310	1.0
7) MPS Lunar Landing	200	0.1

ACS - attitude control system
MPS - main propulsion system

Figure 1-1.2.3.4-4. Lunar DRM - L4 Mission Timeline

For purposes of the analysis, the Cassini mission was selected as the reference point and assigned an energy level of one. As seen in Figure 1-1.2.3.5-1, the solar probe mission in CNDB FY89 and the comet rendezvous asteroid flyby mission have selected trajectories consistent with this range of energy. All of these missions currently baseline approximately the same upper stage capability.

The space nuclear/debris disposal mission requires the highest launch energy. Because of the unique payload rendezvous and grappling requirements for this mission, a DRM will be specifically dedicated to that mission.

The comet nucleus sample return (CNSR) delivery mission, with a payload of 16 metric tons, was selected as the driving planetary DRM. Second only to the space nuclear/debris disposal mission, the energy requirements for this mission were the highest of all planetary missions.

Mission Objectives. The mission objective of DRM-P1 is to deliver an unmanned planetary probe to an Earth escape trajectory targeted for a cometary rendezvous, drop the sample return spacecraft, and return the transfer vehicle to the Space Station (Figure 1-1.2.3.5-2). There is no requirement for a manned capability. This planetary design reference mission and the geosynchronous delivery design reference mission are characteristic of the types of upper stage missions currently flown by the Inertial Upper Stage (IUS) and Centaur stages.

The first flight opportunity occurs in 1999, with the maximum flight rate being one flight per year. The initial operating capability for the Space Transfer Vehicle coincides with this planetary DRM. The initial orbit for the mission is the Space Station orbit, which is circular at an altitude of 407 km and an inclination of 28.5 degrees. This vehicle will be based at Space Station and capable of reusability.

The specified C_3 of 29 km^2/sec^2 is taken from the mission requirements for a CNSR mission to comet Kopff.

- Unmanned
- No servicing
- Worst case energy requirements from CNDB

Mission	Mass, kg	C3, km**2/s**2	Energy Factor	
Cassini	5100	30	1.00	
Solar Probe	1000	140	0.98	
Solar Probe*	1400	140	1.37	
CRAF	5270	28	1.00	
CNSR	6000	26	1.10	Selected for planetary DRM
CNSR*	16000	29	3.09	◄───
MRSR	3500	11	0.49	
MRSR*	4100	11	0.57	
SNDD*	25000	11	3.47	◄─── Separate DRM

* From modified CNDB or augmented set

CRAF - Comet Rendezvous/Asteroid Flyby
CNSR - Comet Nucleus Sample Return
MRSR - Mars Rover Sample Return
SNDD - Space Nuclear/Debris Disposal

Figure 1-1.2.3.5-1. Planetary DRM - P1 Selection Rationale

First flight:	1999
Flight rate:	One flight/year
Initial orbit:	SSF (407 km circular)
Final orbit:	C3 =29 Km**2/sec**2
Delivery Only	
Unmanned	

Figure 1-1.2.3.5-2. Planetary DRM - P1 Mission Objectives

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Payload Requirements. The payload mass of 16,000 kg is taken from the mission requirements for a CNSR mission to comet Kopff. The payload dimensions are estimated assuming STS compatibility and a density of 80 kg/m^3 typical of these types of payloads. The resulting payload volume is a cylinder with a length of 12 meters and a diameter of 4.6 meters. Figure 1-1.2.3.5-3 describes the payload requirements for DRM - P1.

The center of gravity of the payload is assumed to be at the center of the physical envelope of the payload. The average and peak power requirements of 750 and 900 watts are derived from the Magellan/IUS spacecraft requirements. The thermal protection required of the transfer vehicle is to maintain a benign attitude with respect to the sun (e.g., shaded or rotisserie). The data communications requirements of 16 discreet commands and 1000-bps telemetry are derived from TDRS requirements. Integration of the payload with the STV at the Space Station is required.

Mission Profile. The illustration shown in Figure 1-1.2.3.5-4 depicts the planetary DRM mission profile. The STV separates from the Space Station, coasts to the proper point for the first burn, and then accelerates to the escape trajectory required by the payload. The payload is then dropped, and the STV decelerates into an elliptical orbit with an apogee of 500,000 km. At apogee, a small burn is performed to reduce the perigee from 407 km to 83 km. The STV aerobrakes into an 407- by 56-km orbit and circularizes the orbit at 407 km. The transfer vehicle coasts to rendezvous with the Space Station.

Mission Timeline. The mission timeline for the planetary delivery mission, including ΔV's and ΔT's is illustrated in Figure 1-1.2.3.5-5. It should be noted that the ΔV's do not include any gravity losses. For the purpose of developing design reference missions ideal burns were assumed.

1-1.2.3.6 Geosynchronous Delivery Design Reference Mission - G1

Selection Rationale. The geosynchronous delivery and planetary delivery missions constitute the fundamental payload building block of payloads for the initial Space Transfer Vehicle. Five different GEO payloads were listed in the STV mission model. As shown in Figure 1-1.2.3.6-1, the worst case payload

		Source
• Physical Description		
• Mass	16,000 kg	CNSR Mission
• Dimensions	Cylindrical	
	12 x 4.6 meters	Estimated (80 kg/m3)
• Center of gravity	Center of envelope	Estimated
• Resource Requirements		
• Power	750 watts (average)	Magellan/IUS
	900 watts (peak)	Magellan/IUS
• Thermal protection	Maintain benign attitude	
• Data Communications	16 discreet commands	TDRS
	1000 bps telemetry	TDRS
• Support services		
None		

Figure 1-1.2.3.5-3. Planetary DRM - P1 Payload Requirements
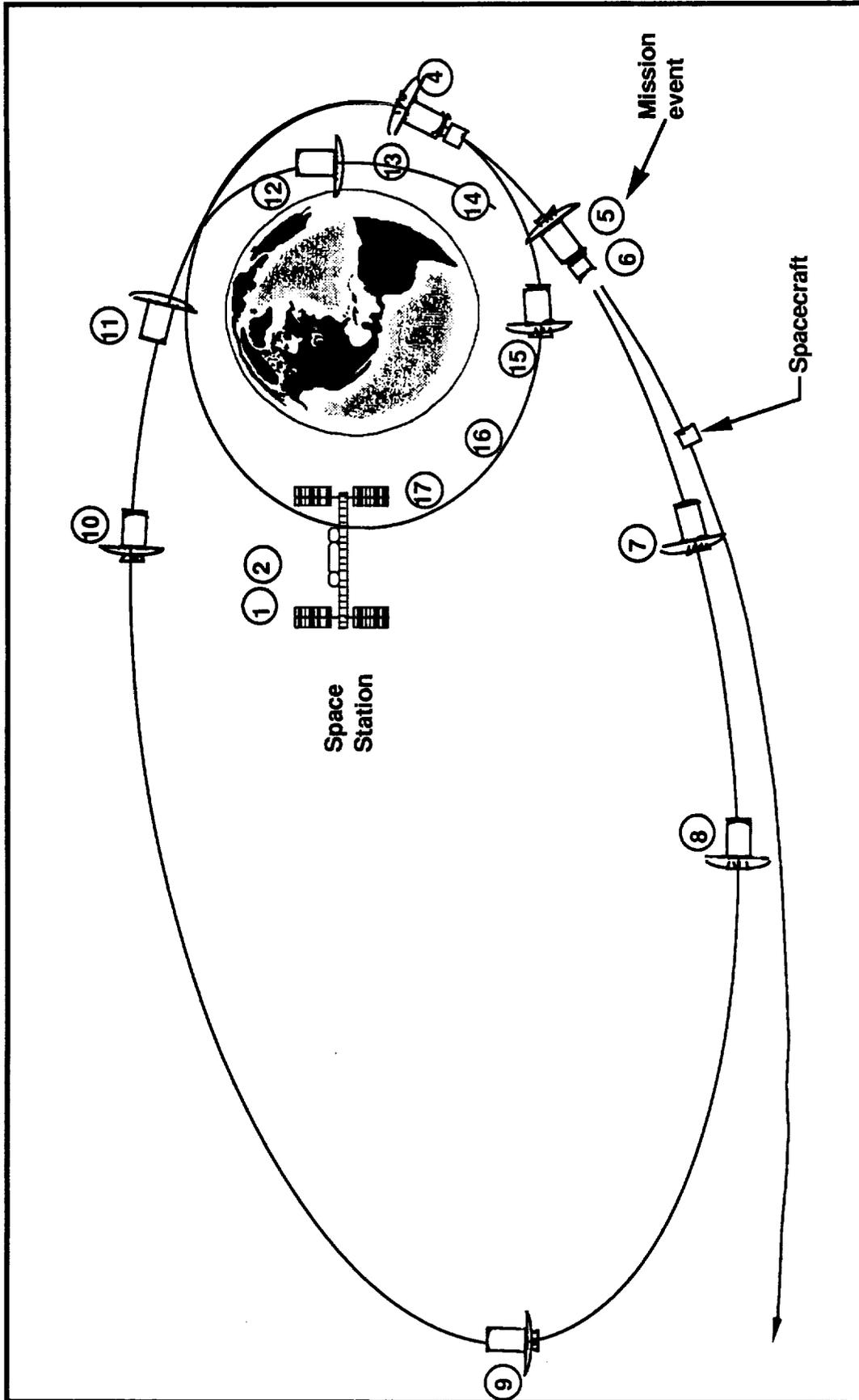
Figure 1-1.2.3.5-4.   Planetary DRM - P1 Mission Profile

Event	ΔV (m/s)	ΔT (hrs)	Event	ΔV (m/s)	ΔT (hrs)
1) Docked at Space Station	0	0.0	10) ACS coast	3	176.1
2) ACS separation	3	0.0	11) MPS pre aero correct	15	0.1
3) ACS coast	3	0.8	12) Aeromaneuver	0	0.1
4) MPS perigee burn	4,440	0.3	13) MPS post aero correct	77	0.1
5) ACS payload positioning	5	1.0	14) ACS coast	3	0.8
6) Drop payload	0	0.0	15) MPS circularization burn	103	0.1
7) MPS deorbit burn	1,337	0.1	16) ACS coast	3	0.8
8) ACS coast	3	176.3	17) ACS rendezvous and docking	12	1.0
9) ACS deorbit burn (Perigee trim)	3	0.1 ·			

ACS - Attitude control system
MPS - Main propulsion system

ΔV's do not include gravity or steering losses

Figure 1-1.2.3.5-5. Planetary DRM - P1 Mission Timeline

- Typical geosynchronous delivery mission
- Unmanned

Mission	Mass, kg	
Deep Space Relay Satellite	2950	
EOTS -1 (NASDA)	1000	
TD of Large GEO Satellite (NASDA)	6200	
GEO Relay	3850	Selected for geosynchronous DRM
GEO Platform (Mission to Planet Earth)	10000 ◄—	

Figure 1-1.2.3.6-1. Geosynchronous DRM - G1 Selection Rationale

from the standpoint of injection energy was the 22,000-lb (10 metric tons) geosynchronous platform for the mission to planet Earth.

Mission Objectives. The mission objectives of DRM-G1 are to deliver an unmanned spacecraft to GEO and return the transfer vehicle to the Space Station (Figure 1-1.2.3.6-2). The first flight opportunity occurs in 2005, with the maximum flight rate being three flights per 2 years taken from the STV mission model. The initial orbit for the mission is the 407-km altitude, 28.5-degree inclination Space Station orbit. The final orbit for the payload is a circular GEO, with an altitude of 35,760 km and an inclination of 0 degrees. There is no requirement to support a crew module and the vehicle is returned to the space base at the end of the mission.

Payload Requirements. The payload mass of 10,000 kg and dimensions of 6.1 by 4.6 meters cylindrical (Figure 1-1.2.3.6-3) are taken directly from the civil needs database entry for the GEO platform. The center of gravity of the payload is assumed to be at the center of the physical envelope of the payload. The average and peak power requirements of 400 and 600 watts are derived from the requirements of a typical communications satellite. The thermal protection required of the transfer vehicle is to maintain a benign attitude with respect to the sun (e.g., shaded or rotisserie).

The data communications requirements of 16 discreet commands and 1000-bps telemetry are derived from TDRS spacecraft and are characteristic of geosynchronous payloads. Integration of the payload with the STV at the Space Station is required.

Mission Profile. The mission profile for the geosynchronous delivery mission is shown in Figure 1-1.2.3.6-4. The numbered events correspond to the mission timeline in Figure 1-1.2.3.6-5. The STV separates from the Space Station, coasts to the proper point for the first burn, and performs a main propulsion burn to put the vehicle/spacecraft combination into geosynchronous transfer orbit. After coasting to the apogee of the transfer orbit, the vehicle does an apogee burn to circularize the orbit at geosynchronous altitude. The payload is positioned and dropped, and the vehicle coasts in orbit until the proper time to perform the deorbit burn. This burn puts the vehicle into an elliptical orbit with a

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First flight:	2005
Flight rate:	3 flights/2 years
Initial orbit:	SSF (407 km circular)
Final orbit altitude:	35,760 km
	Circular
	Inclination = 0°
Delivery Only	
Unmanned	

Figure 1-1.2.3.6-2. Geosynchronous DRM - G1 Mission Objectives

		Source
• Physical Description		
• Mass	10,000 kg	CNDB GEO Platform (Mission to Planet Earth)
• Dimensions	Cylindrical	
	6.1 x 4.6 meters	CNDB
• Center of gravity	Center of envelope	Estimated
• Resource Requirements		
• Power	400 watts (average)	Typical Comm Sat.
	600 watts (peak)	Typical Comm Sat.
• Thermal protection	Maintain benign thermal attitude	
• Data Communications	16 discreet commands	TDRS
	1000b/s telemetry	TDRS
• Support services		

Figure 1-1.2.3.6-3. Geosynchronous DRM - G1 Payload Requirements

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BOEING

Space Station

Mission event

Spacecraft
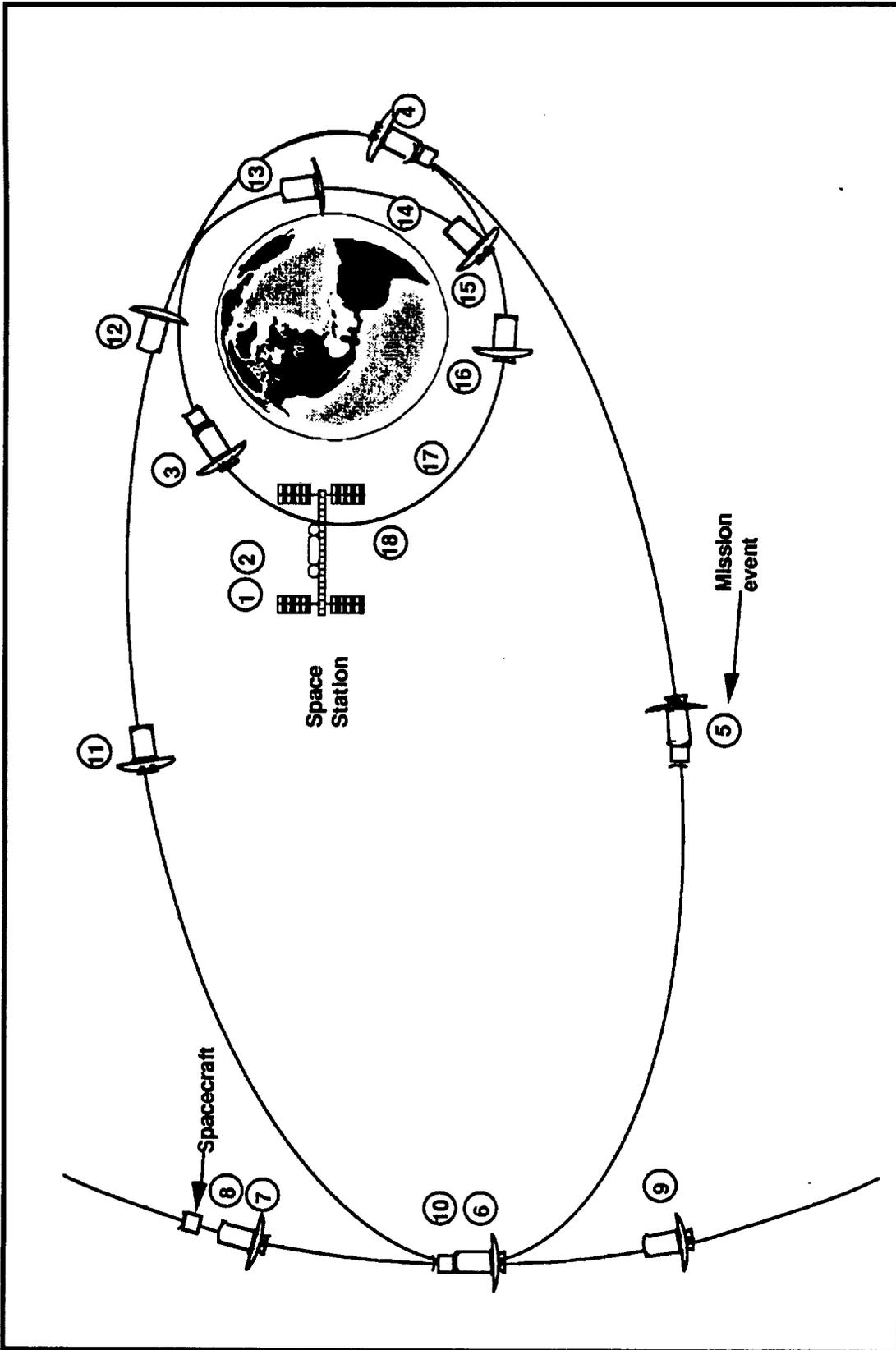
Figure 1-1.2.3.6-4. Geosynchronous DRM - G1 Mission Profile

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Event	Δ V (m/s)	Δ T (hrs)	Event	Δ V (m/s)	Δ T (hrs)
1) Docked at Space Station	0	0.0	10) MPS deorbit burn	1,844	0.1
2) ACS separation	3	0.0	11) ACS coast	3	5.2
3) ACS coast	3	0.8	12) MPS pre aero correct	15	0.1
4) MPS perigee burn	2,419	0.2	13) Aeromaneuver	0	0.1
5) ACS coast	3	5.3	14) MPS post aero correct	77	0.1
6) MPS apogee burn	1,774	0.1	15) ACS coast	3	0.7
7) ACS payload positioning	5	1.0	16) MPS circularization burn	103	0.1
8) Drop payload	0	0.0	17) ACS coast	3	0.8
9) ACS coast	6	24.0	18) ACS rendezvous and docking	12	1.0

ACS - Attitude control system
MPS - Main propulsion system

ΔV's do not include gravity or steering losses

Figure 1-1.2.3.6-5. Geosynchronous DRM - G1 Mission Timeline

83-km perigee. The STV aerobrakes into an 407- by 56-km orbit and circularizes the orbit at 407 km. The transfer vehicle coasts to rendezvous with the Space Station.

Mission Timeline. Figure 1-1.2.3.6-5 shows the mission timeline for the GEO delivery mission, including ΔV's and ΔT's. A small percentage of the inclination change is accomplished during the perigee burn. The STV with payload coasts for just under 6 hours in the geosynchronous transfer orbit. After reaching apogee, the orbit is circularized. One orbit (24 hours) is allowed until deorbit from the geosynchronous altitude and aeroassisted return.

1-1.2.3.7   Molniya Delivery Design Reference Mission - D1

Selection Rationale. As indicated in the mission model analysis, many of the DoD missions and mission requirements are already covered in the design reference mission categories identified for civil needs. The only DoD mission with distinct characteristics and requirements was the molniya mission. The molniya mission uses a three-burn transfer to achieve orbit.

A DRM was constructed for a molniya mission based on sample payload characteristics recently used in the Air Force Upper Stage Responsiveness Study (USRS). The purpose of the molniya delivery design reference mission was to determine if the energy requirements would drive the STV sizing.

Mission Objectives. The mission objectives of DRM-D1 (Figure 1-1.2.3.7-1) are to deliver an unmanned spacecraft to a 12-hour elliptical orbit with an inclination of 63.4 degrees and a 500-km perigee altitude and return the transfer vehicle to the Space Station. The first flight opportunity occurs in 2000, with the maximum flight rate of one flight per 2 years.

The mission is initiated from the Space Station orbit at a circular orbit altitude of 407 km and an inclination of 28.5 degrees. The STV would not be piloted for this mission.

Payload Requirements. The sample payload mass of 6800 kg (Figure 1-1.2.3.7-2) was typical of these types of payloads and taken directly from the

First flight:	2000 (Estimated)
Flight rate:	One every two years (Estimated)
Initial orbit:	SSF (407 km circular)
Final orbit:	12-Hour Elliptical
	Inclination = 63.4°
	500 km perigee altitude
Delivery Only	
Unmanned	

Figure 1-1.2.3.7-1. Molniya DRM - D1 Mission Objectives

		Source
• Physical Description		
• Mass	6800 kg	Estimated
• Dimensions	Cylinder	
	4 x 4.6 meters	Estimated
• Center of gravity	Center of envelope	Estimated
• Resource Requirements		
• Power	400 watts (average)	Typical Comm. Sat.
	600 watts (peak)	Typical Comm. Sat.
• Thermal protection	Maintain benign thermal attitude	
• Data Communications	16 discreet commands	TDRS
	1000 bps telemetry	TDRS
• Support services		
None		

Figure 1-1.2.3.7-2. Molniya DRM - D1 Payload Requirements

reference payload description used in the Air Force USRS. The USRS payload was approximately a 4-meter-long and 4.6-meter-diameter cylinder. As with all the design reference mission payloads, the center of gravity of the payload is assumed to be at the center of the physical envelope.

The average and peak power requirements of 400 and 600 watts are derived from the requirements of a typical communications satellite. The thermal protection required of the transfer vehicle is to maintain a benign attitude with respect to the sun (e.g., shaded or rotisserie). The data communications requirements of 16 discreet commands and 1000-bps telemetry are derived from TDRS requirements and are typical of large complex satellite systems. Integration of the payload with the STV at the Space Station is required.

**Mission Profile.** Figure 1-1.2.3.7-3 depicts the mission profile for the molniya delivery mission, DRM-D1. The numbered events correspond to the mission timeline in Figure 1-1.2.3.7-4. The STV separates from the Space Station, coasts to the injection point, and performs a main propulsion burn to put the vehicle/spacecraft combination into the first of two transfer orbits. Two more main propulsion burns put the vehicle into the mission orbit. The payload is positioned and dropped and the vehicle performs the first deorbit burn. This burn puts the vehicle into the first of two transfer legs. After one more propulsive burn, the STV aerobrakes into an 407- by 56-km orbit and circularizes the orbit at 407 km. The transfer vehicle coasts to rendezvous with the Space Station.

Mission Timeline. Figure 1-1.2.3.7-4 describes the mission timeline for the molniya delivery design reference mission, including ΔV's and ΔT's. Although the ΔV's presented in this chart are based on a 500-km circular initial orbit, the ΔV's for a 407-km Space Station Freedom orbit should not be greatly different.

The STV separates from the Space Station, coasts to the injection point, and performs a main propulsion burn to put the vehicle/spacecraft combination into the first of two transfer orbits. The first burn pumps up the apogee altitude of the 28.5-degree inclination orbit. The second burn, performed during ascent to orbit apogee, changes the orbit inclination and adds energy to the orbit. The third and final burn inserts the STV and payload into the 12-hour molniya orbit.
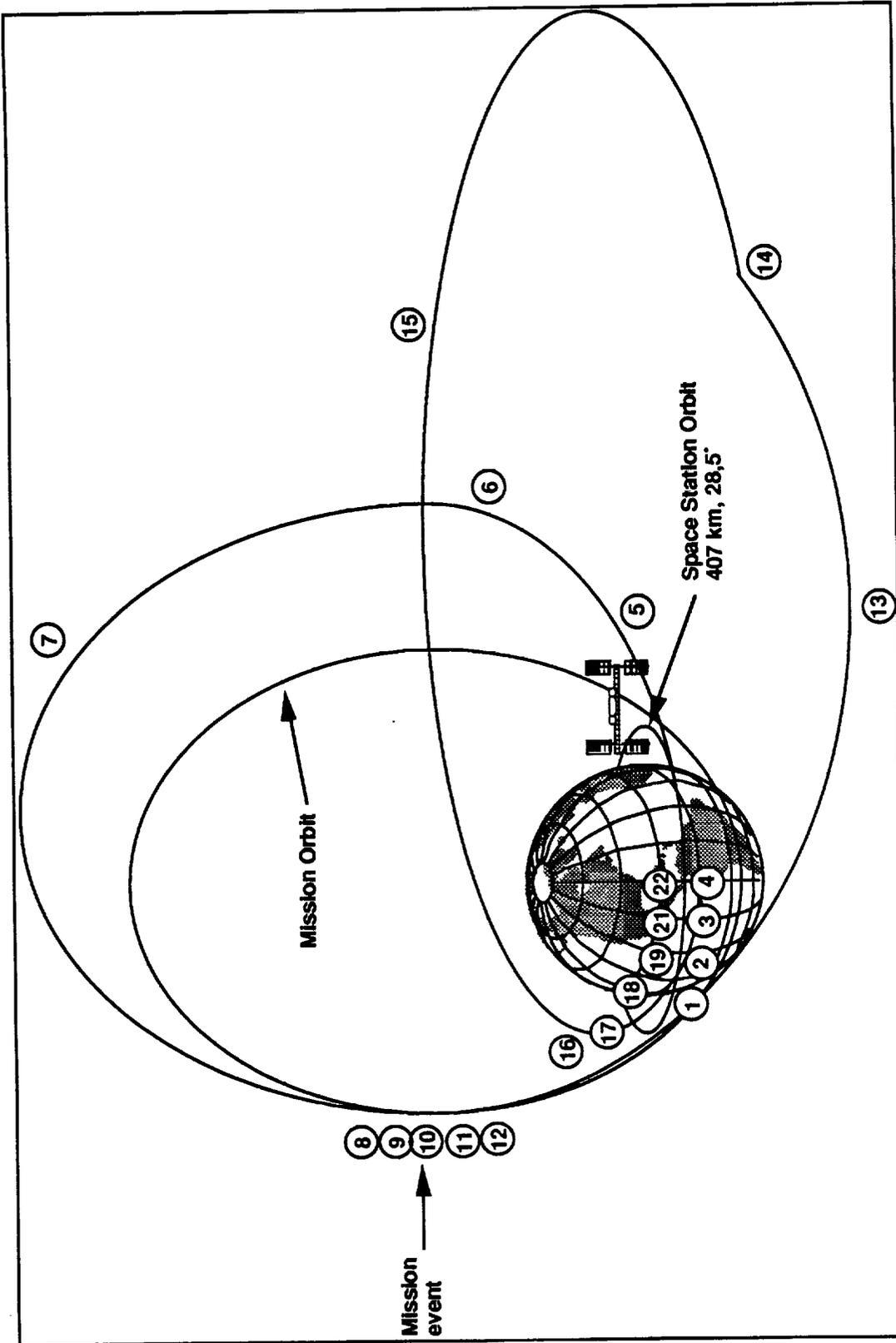
Figure 1-1.2.3.7-3. Molniya DRM - D1 Mission Profile

Event	Δ V (m/s)	Δ T (hrs)	Event	Δ V (m/s)	Δ T (hrs)
1) Docked at Space Station	0	0.0	12) MPS deorbit burn	1,031	0.1
2) ACS separation	3	0.0	13) ACS coast	3	7.9
3) ACS coast	3	0.8	14) MPS deorbit burn	785	0.1
4) MPS burn (apogee raise)	2,402	0.2	15) ACS coast	3	24.6
5) ACS coast	3	2.3	16) MPS pre aero correct	15	0.1
6) MPS burn	1,415	0.1	17) Aeromaneuver	0	0.1
7) ACS coast	3	12.0	18) MPS post aero correct	77	0.1
8) MPS burn	659	0.1	19) ACS coast	3	0.7
9) ACS coast	6	23.6	20) MPS circularization burn	103	0.1
10) ACS payload positioning	5	1.0	21) ACS coast	3	0.5
11) Drop payload	0	0.0	22) ACS rendezvous and docking	12	1.0

ACS - Attitude control system
MPS - Main propulsion system

• ΔV's do not include gravity or steering losses
• ΔV's are based on a 500 km Space Station orbit

Figure 1-1.2.3.7-4. Molniya DRM - D1 Mission Timeline

After payload separation, the vehicle will perform two major propulsive maneuvers to achieve a highly elliptical 28.5-degree inclination orbit with the ascending node matching Space Station Freedom and the perigee altitude of approximately 56 km. The Space Transfer Vehicle would then perform an aeroassisted maneuver through the upper atmosphere and a circularization burn to match orbit with the Space Station.

1-1.2.3.8 Manned GEO Servicing Design Reference Mission - G2

Selection Rationale. The augmented set of the STV mission model introduced several servicing missions into the analysis that can be broken down between piloted and unpiloted servicing missions. The geosynchronous servicing mission includes a piloted capability to assist in the servicing procedure. The life support, man-machine interfaces, man rating, grappling, and operational aspects of this servicing mission introduce many requirements.

As a starting point we assumed crew module capabilities similar to the lunar excursion crew module (LECM) from the lunar transportation system. The mission model defines the crew module mass to be 3,700 kg, almost identical to the mass currently estimated for the LECM. In addition to the mass of the crew module itself, 4,000 kg of servicing equipment was included as part of the payload. It was also assumed that the crew module would supply the resource requirements of the mission payload. Other factors concerning the DRM G2 selection rationale appear in Figure 1-1.2.3.8-1.

Mission Objectives. The mission objectives of DRM-G2 (Figure 1-1.2.3.8-2) included delivery of a manned crew module and servicing equipment to GEO to perform a servicing mission and return of the transfer vehicle, crew module, and crew to the Space Station. The first flight occurs in 2006, with the maximum flight rate being one flight per year as indicated in the augmented set of the FY89 CNDB.

The target spacecraft in need of servicing is in a circular GEO, with an altitude of 35,760 km and an inclination of 0 degrees.

- Servicing equipment and crew module characteristics taken from STV mission model database
- Crew cab capabilities same as LTCM for commonality
- Resources (power, telemetry and command, etc.) supplied by crew module and servicing equipment
- The manned servicing characteristics of this DRM are intended to cover manned polar servicing mission

Figure 1-1.2.3.8-1. Geosynchronous DRM - G2 Selection Rationale

First flight:	2006
Flight rate:	One flight/year
Initial orbit:	SSF (407 km circular)
Servicing orbit:	35,760 km altitude
	Circular
	Inclination = 0°
Servicing	
Manned	

Figure 1-1.2.3.8-2. Geosynchronous DRM - G2 Mission Objectives

		Source
• Physical Description		
• Mass	4000 kg (Servicing Equip.)	STV Mission Model
	3700 kg (Crew Module)	STV Mission Model
• Dimensions	Crew Mod. same as LECM	Estimated
• Center of gravity	Center of envelope	Estimated
• Resource Requirements		
• Power	0 watts	(Supplied by crew module)
• Thermal protection	Maintain benign thermal attitude	
• Data Communications	0 discreet commands	(Supplied by crew module)
	0 kb/sec telemetry	(Supplied by crew module)
• Support services		
• Manned cab provides servicing capability		
• Servicing equipment support fixture		
• Platform or target satellite housekeeping resources		

Figure 1-1.2.3.8-3. Geosynchronous DRM - G2 Payload Requirements

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Payload Requirements. The payload mass of 3,700 kg for the crew module and 4,000 kg for servicing equipment are taken from the CNDB entry for geosynchronous platform servicing. For purposes of commonality and vehicle evolution, the servicing crew module was assumed to have the same design as the LECM. The dimensions and mass of the crew module are assumed to be the same as the LECM. The LECM design was generated during the early days of the 90-day study as the smaller of two modules to be used in conjunction with a lunar architecture using an LTV/LEV combination for lunar exploration. The payload requirements for the geosynchronous payload servicing mission appear in Figure 1-1.2.3.8-3.

The center of gravity of the payload is assumed to be at the center of the physical envelope of the payload. The power and data communications requirements are assumed to be supplied by the crew module. The thermal protection required of the transfer vehicle is to maintain a benign attitude with respect to the sun (e.g., shaded or rotisserie).

Mission Profile. Figure 1-1.2.3.8-4 illustrates the mission profile for the manned GEO servicing mission, DRM-G2. The numbered events correspond to the mission timeline shown in Figure 1-1.2.3.8-5.

The STV separates from the Space Station and coasts to the injection point for the first major ΔV maneuver. The main engines burn to put the vehicle/crew module/servicing equipment payload combination into geosynchronous transfer orbit. After coasting to the apogee of the transfer orbit, the vehicle does an apogee burn to circularize the orbit at geosynchronous altitude and rendezvous and grapple the payload in need of servicing or repair. The servicing mission is performed, and the vehicle coasts in orbit until the proper time to perform the deorbit burn. This burn puts the vehicle into an elliptical orbit with a 83-km perigee. The STV aerobrakes into an 407- by 56-km orbit and circularizes the orbit at 407 km. The transfer vehicle coasts to rendezvous with the Space Station.

Mission Timeline. The mission timeline for the manned GEO servicing mission, including ΔV's and ΔT's. The mission involves multiple passes through the Van Allen radiation belts. As with the lunar crew module, radiation shielding
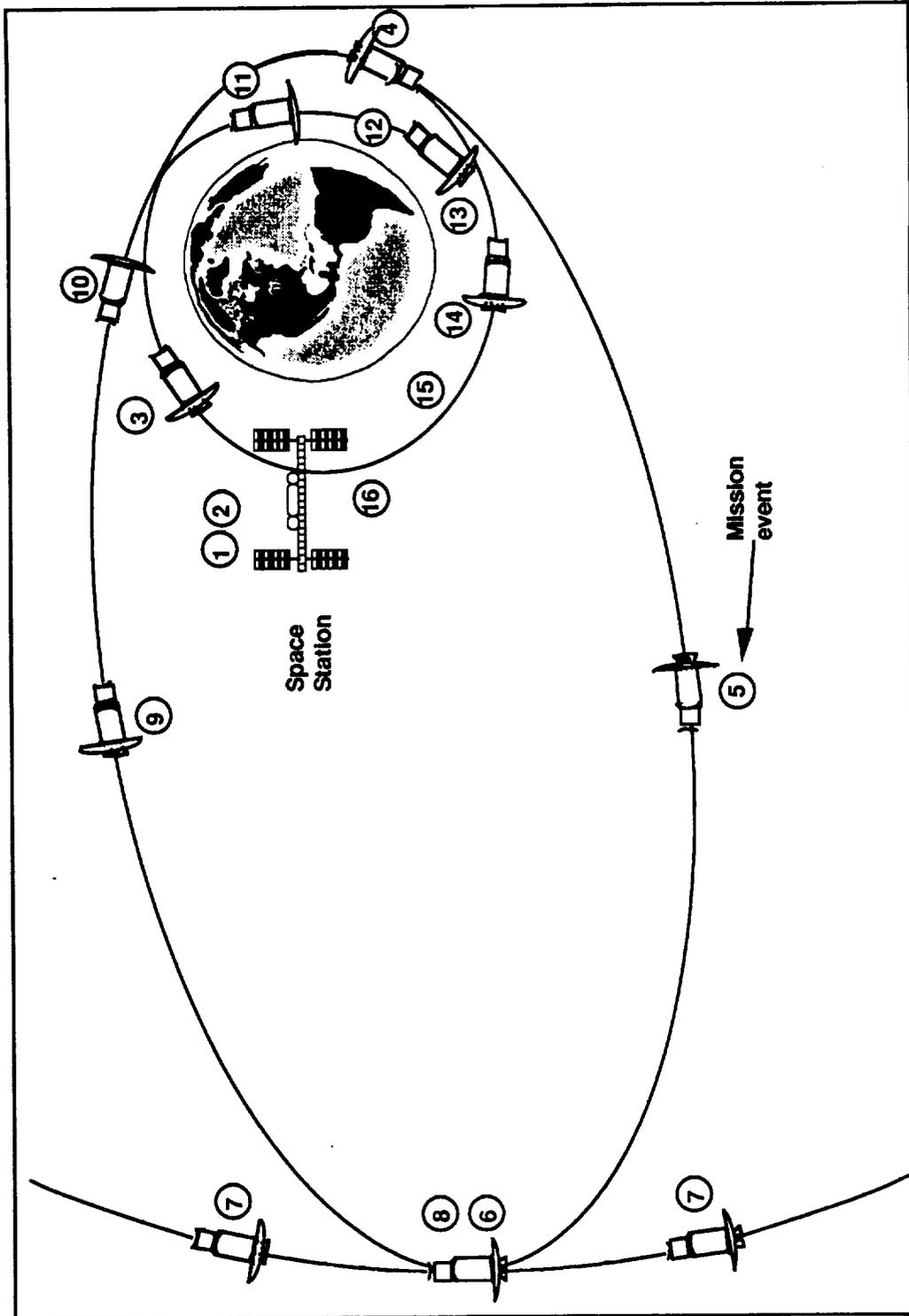
Figure 1-1.2.3.8-4. Geosynchronous DRM - G2 Mission Profile

Event	Δ V (m/s)	Δ T (hrs)		Event	Δ V (m/s)	Δ T (hrs)
1) Docked at Space Station	0	0.0	9)	ACS coast	3	5.2
2) ACS separation	3	0.0	10)	MPS pre aero correct	15	0.1
3) ACS coast	3	0.8	11)	Aeromaneuver	0	0.1
4) MPS perigee burn	2,419	0.2	12)	MPS post aero correct	77	0.1
5) ACS coast	3	5.3	13)	ACS coast	3	0.7
6) MPS apogee burn	1,774	0.1	14)	MPS circularization burn	103	0.1
7) GEO operations	TBD	TBD	15)	ACS coast	3	0.8
8) MPS deorbit burn	1,844	0.1	16)	ACS rendezvous and docking	12	1.0

ACS - attitude control system
MPS - main propulsion system

ΔV's do not include gravity or steering losses

Figure 1-1.2.3.8-5. Geosynchronous DRM - G2 Mission Timeline

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adequate to withstand the total dosage would be required as a part of the module design.

1-1.2.3.9 Unmanned Polar Platform Servicing DRM - S1

Selection Rationale. The following DRM was designed to include a polar orbit mission in the analysis. While the mission model included both piloted and unpiloted missions to both polar low Earth and geosynchronous orbits, DRM G2 and DRM S1 ensure that the energy requirements for both orbits and operational requirements for both autonomous and piloted missions are included in the STV design.

Polar orbits are very hard to obtain when launching from KSC and require a looping initial trajectory with high apogee where a large plane change maneuver is performed to move from the Space Station Freedom inclination to polar or sun-synchronous inclination (-98 degrees).

In addition, this DRM will explore the autonomous requirements of an unmanned servicing mission. The most likely candidate for this mission would be the Earth observational satellites included in the mission to planet earth. The servicing payload weighs a total of 4 metric tons with a 1 metric ton servicing kit and 3 metric tons of replacement cargo.

Mission Objectives. The mission objectives of DRM-S1 are to deliver an unmanned servicing kit and cargo to a polar orbit (824-km altitude, 98.7-degree inclination) to perform a servicing mission and return the transfer vehicle, servicing kit, and cargo to the Space Station (Figure 1-1.2.3.9-1).

The first flight opportunity occurs in 2001, with the maximum flight rate of one flight per 2 years. The initial orbit is circular at an altitude of 407 km and an inclination of 28.5 degrees.

Payload Requirements. The payload mass of 1,000 kg for the servicing kit and 3,500 kg for servicing cargo are taken from the STV mission model entry for polar platform servicing. The cargo dimensions are estimated from the mass of

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BOEING

First flight:	2001
Flight rate:	One flights every two years
Initial orbit:	SSF (407 km circular)
Final orbit:	824 km Altitude
	Circular
	Inclination = 98.7°
Servicing	
Unmanned	

Figure 1-1.2.3.9-1. Polar Servicing DRM - S1 Mission Objectives

		Source
• Physical Description		
• Mass	1000 kg Servicing kit	STV Mission Model
	3500 kg Cargo	STV Mission Model
• Cargo dimensions	3 x 3 x 5 meters	Estimated
• Cargo center of gravity	Center of envelope	Estimated
• Resource Requirements		
• Power	0 watts (Supplied by servicing kit)	
• Thermal protection	Maintain benign thermal attitude	
• Data Communications	200 kbps telemetry	
• Support services		
• Automated/telerobotic servicer		

Figure 1-1.2.3.9-2. Polar Servicing DRM - S1 Payload Requirements

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the cargo. The servicing cargo includes an automated and/or telerobotic servicer. The S1 payload requirements are listed in Figure 1-1.2.3.9-2.

The center of gravity of the payload is assumed to be at the center of the physical envelope of the payload. The power requirements are assumed to be supplied by the servicing kit. The thermal protection required of the transfer vehicle is to maintain a benign attitude with respect to the sun (e.g., shaded or rotisserie).

Mission Profile. Figure 1-1.2.3.9-3 graphically portrays the mission profile for the unmanned polar platform servicing mission, DRM-S1. The numbered events correspond to the S1 mission timeline.

The STV separates from the Space Station, coasts to the proper point for the first burn, and performs a main propulsion burn to put the vehicle/spacecraft combination into a 407- by 130,000-km orbit. After coasting to the apogee of this orbit, the vehicle does a main propulsion burn to change the inclination and perigee of the orbit to 98.7 degrees and 824 km. At the perigee of this second transfer orbit, the vehicle performs a circularization burn. The servicing mission is performed, and the vehicle coasts in orbit until the proper time to perform the first return burn. This burn puts the vehicle into a 407- by 130,000-km orbit. A second burn changes the inclination back to 28.5 degrees and decreases the orbit perigee to 83 km. The STV aerobrakes into an 407- by 56-km orbit and circularizes the orbit at 407 km. The transfer vehicle coasts to rendezvous with the Space Station.

Mission Timeline. Figure 1-1.2.3.9-4 shows a mission timeline for the unmanned polar platform servicing mission, including ΔV's and ΔT's.

1-1.2.3.10 LEO Space Tug Design Reference Mission - T1

Selection Rationale. During 1990, the Orbital Maneuver Vehicle program was canceled. The space tug mission would use the STV in a manner much like the Orbital Maneuvering Vehicle for ferrying payloads from one point to another in LEO.
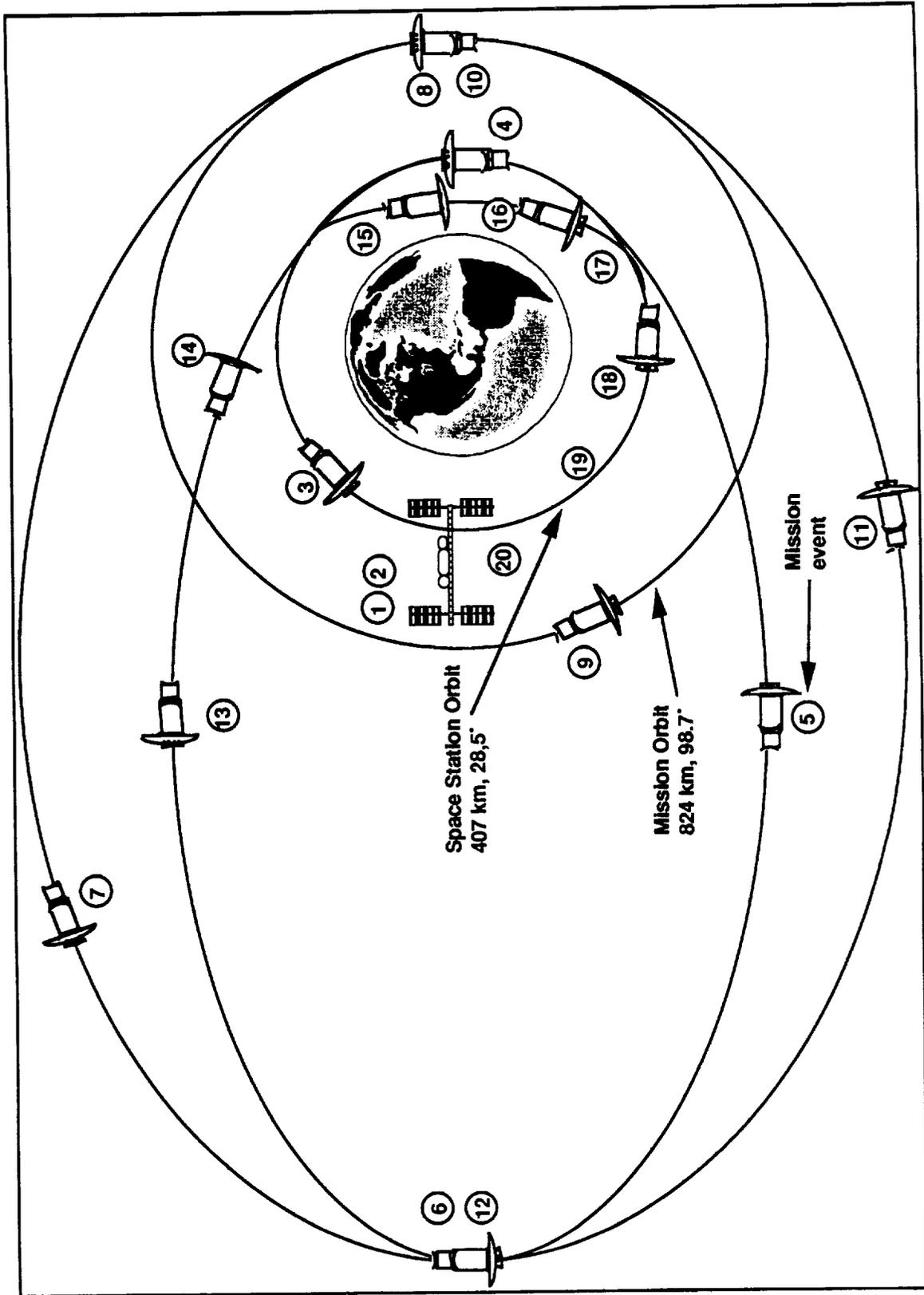
Figure 1-1.2.3.9-3. Polar Servicing DRM - S1 Mission Profile

Space Station Orbit
407 km, 28,5°

Mission Orbit
824 km, 98.7°

Mission
event

Event	ΔV (m/s)	ΔT (hrs)	Event	ΔV (m/s)	ΔT (hrs)
1) Docked at Space Station	0	0.0	11) ACS coast	3	26.6
2) ACS separation	3	0.0	12) MPS deorbit/plane ch. burn	592	0.1
3) ACS coast	3	0.8	13) ACS coast	3	26.4
4) MPS perigee burn	2,915	0.2	14) MPS pre aero correct	15	0.1
5) ACS coast	3	26.5	15) Aeromaneuver	0	0.1
6) MPS plane change burn	614	0.1	16) MPS post aero correct	77	0.1
7) ACS coast	3	26.6	17) ACS coast	3	0.7
8) MPS circularization burn	2,815	0.1	18) MPS circularization burn	103	0.1
9) ACS orbital operations	TBD	TBD	19) ACS coast	3	0.8
10) MPS perigee burn	2,815	0.1	20) ACS rendezvous and docking	12	1.0

ACS - attitude control system
MPS - main propulsion system

ΔV's do not include gravity or steering losses

Figure 1-1.2.3.9-4. Polar Servicing DRM - S1 Mission Timeline

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The primary mission for this DRM would be to retrieve lunar mission propellant and cargo elements from expendable launch vehicles in LEO. Because the propulsive assets of a space-based STV are already on orbit and operational, the vehicle could be used to assist in gathering the components of a multilaunch mission scenario in LEO.

This design reference mission is also intended to capture the great observatory reboost missions such as Hubble telescope, SIRTF (which is one of the missions in the study), model, AXAF, and the Gamma Ray Observatory.

Mission Objectives. The mission objectives of DRM-T1 (Figure 1-1.2.3.10-1) are to proceed outside the command and control zone of the Space Station, pick up payloads such as propellant tanks from the launch vehicle, and return the transfer vehicle and payloads to the Space Station.

The first flight opportunity occurs in 1999, with multiple flights per year. The initial orbit for the mission is the Space Station orbit, which is circular at an altitude of 407 km and an inclination of 28.5 degrees.

Payload Requirements. The payload mass of 71 tons was based on the projected size of the propellant launches to prepare for one lunar mission. The cargo dimensions were estimated from the projected payload capacity of Shuttle-C (Figure 1-1.2.3.10-2).

The center of gravity of the payload is assumed to be at the center of the physical envelope of the payload. The thermal protection required of the transfer vehicle is to maintain a benign attitude with respect to the sun (e.g., shaded or rotisserie). The vehicle would require a means of grappling the payloads for return to the Space Station.

Mission Profile. The mission profile for the unmanned space tug mission, DRM-T1, is shown in Figure 1-1.2.3.10-3. The numbered events correspond to the mission timeline shown in Figure 1-1.2.3.10-4.

The STV separates from the Space Station and coasts to the position of the waiting launch vehicle and payload. After rendezvous with the launch vehicle,

BOEING

First flight:	1999
Flight rate:	Multiple flights/year
Initial orbit:	SSF (407 km circular)
Final orbit:	Outside SSF command and control zone
	Circular
	Inclination = 28.5°
Payload pickup	
Unmanned	

Figure 1-1.2.3.10-1. Space Tug DRM - T1 Mission Objectives

		Source
• Physical Description		
• Mass	71 t	Shuttle C
• Dimensions	Cylindrical	Lunar propellant tanks
	4.6 x 25 meters	Shuttle C
• Center of gravity	Center of envelope	Actual c.g.
• Resource Requirements		
• Power	0 watts	Estimated
• Thermal protection	Maintain benign attitude	
• Data Communications	200 kbps telemetry	Estimated
• Support services		
• Grapple payloads and return to Space Station		

Figure 1-1.2.3.10-2. Space Tug DRM - T1 Payload Requirements
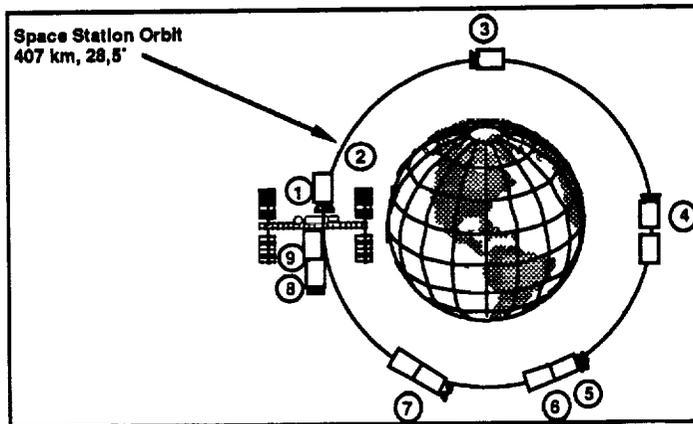
Figure 1-1.2.3.10-3. Space Tug DRM - T1 Mission Profile

Event	Δ V (m/s)	Δ T (hrs)	Event	Δ V (m/s)	Δ T (hrs)
1) Docked at Space Station	0	0.0	6) ACS transfer burn	3	0.0
2) ACS separation/transfer	3	0.0	7) ACS coast	3	0.8
3) ACS coast	3	0.8	8) ACS SSF rendezvous burn	3	0.1
4) ACS P/L rendezvous burn	3	0.1	9) SSF operations	10	4.0
5) Docking/retrieval operations	10	8.0			

ACS - Attitude control system
MPS - Main propulsion system

Figure 1-1.2.3.10-4. Space Tug DRM - T1 Mission Timeline

the transfer vehicle docks with and removes the payload from the launch vehicle. The transfer vehicle/payload combination then transfers back to the Space Station where further operations are performed such as docking the payload to the Space Station.

Mission Timeline. The mission timeline in Figure 1-1.2.3.10-4 steps through the sequence of events for the unmanned space tug mission and includes ΔV's and ΔT's. Entry into and departing from the Space Station zone of influence is regulated by flight operations requirements dictated by the Space Station program. The actual approach and departure from Space Station Freedom would include a number of additional steps not included in this description.

1-1.2.3.11 Nuclear/Debris Disposal Design Reference Mission - N1

Selection Rationale. The nuclear/debris disposal design reference mission has been designed to retrieve a large nuclear electric propulsion vehicle that has failed in LEO and move the vehicle to a safe orbit outside of the Earth/Moon system. It is assumed that the vehicle has failed at 1,300 km while on a spiraling low-thrust trajectory.

The DRM includes elements of the unmanned servicing missions because the vehicle is required to rendezvous with the failed vehicle, grapple and secure the payload to the STV, then restart engines and provide a large ΔV to achieve escape velocity.

Since there is only one disposal mission envisioned in the current CNDB, the design reference mission expends the vehicle on this mission. This mission could be used as the final use of a reusable vehicle. Due to the large size of the nuclear debris and the interplanetary orbit required for disposal, the total energy requirements of this mission could be very high.

Mission Objectives. The mission objectives of DRM-N1 (Figure 1-1.2.3.11-1) included retrieval of an undesirable object, such as a spent or failed nuclear reactor, and disposal of this object in a 1.1-AU solar orbit. The mission is unmanned and the transfer vehicle is expended.

First flight:	2010
Flight rate:	One flight
Initial orbit:	SSF (407 km circular)
Intermediate orbit altitude:	1300 km circular
Final orbit:	Solar Orbit
	1.1 AU Circular
Retrieval and Disposal .	
Unmanned	
STV expended	

Figure 1-1.2.3.11-1. Nuclear Disposal DRM - N1 Mission Objectives

		Source
• Physical Description		
• Mass	25,000 kg	STV Mission Model
• Dimensions	Cylindrical	
	7 x 7.6 meter	Estimated - Shuttle C/80 kg/m3
• Center of gravity	Center of envelope	Estimated
• Resource Requirements		
• Power	500 watts	Estimated
• Thermal protection	Maintain benign attitude	Estimated
• Data Communications	200 kbps telemetry	Estimated
• Support services		
• Rendezvous/grapple payload		
• Secure payload to vehicle		

Figure 1-1.2.3.11-2. Nuclear Disposal DRM - N1 Payload Requirements

The first and only flight occurs in 2010. The initial orbit for the mission is the Space Station orbit, which is circular at an altitude of 407 km and an inclination of 28.5 degrees. The object to be disposed of is assumed to be in a 1,300 km circular orbit with an inclination of 28.5 degrees.

Payload Requirements. The payload mass of 25,000 kg is taken from the STV mission model entry for nuclear/debris disposal. The cargo dimensions are estimated from the mass of the cargo.

The center of gravity of the payload is assumed to be at the center of the physical envelope of the payload. A mechanism to grapple the payload and secure it to the transfer vehicle was required. Other payload requirements appear in Figure 1-1.2.3.11-2.

Mission Profile. Figure 1-1.2.3.11-3 depicts the mission profile for the unmanned nuclear/debris disposal mission, DRM-N1. The numbered events correspond to the mission timeline shown in Figure 1-1.2.3.11-4. The STV separates from the Space Station, coasts to the proper point for the injection burn, and performs a main propulsion firing to put the vehicle/spacecraft combination into a 407- by 1,300-km orbit. After coasting to the apogee of this orbit, the vehicle does a main propulsion burn to circularize the orbit at an altitude of 1,300 km. The transfer vehicle rendezvous with the payload, grapples it, and accelerates the payload to escape velocity. After coasting in a solar orbit , the transfer vehicle performs a circularization burn to put itself and the payload into a 1.1-AU circular solar orbit.

Mission Timeline. The mission timeline for the unmanned nuclear/debris disposal mission, including ΔV's and ΔT's, is illustrated in Figure 1-1.2.3.11-4.

1-1.2.3.12 Manned Capsule Recovery DRM - C1

Selection Rationale. The manned capsule capture and recovery mission is very similar to the manned geosynchronous missions but was included in the DRMs to determine if the orbital energy requirements of the mission surpass the GEO requirements.

Space Station Orbit
407 km, 28,5'

Mission Orbit
1,296 km, 28.5'

Mission
event

1.1 AU
Solar Orbit
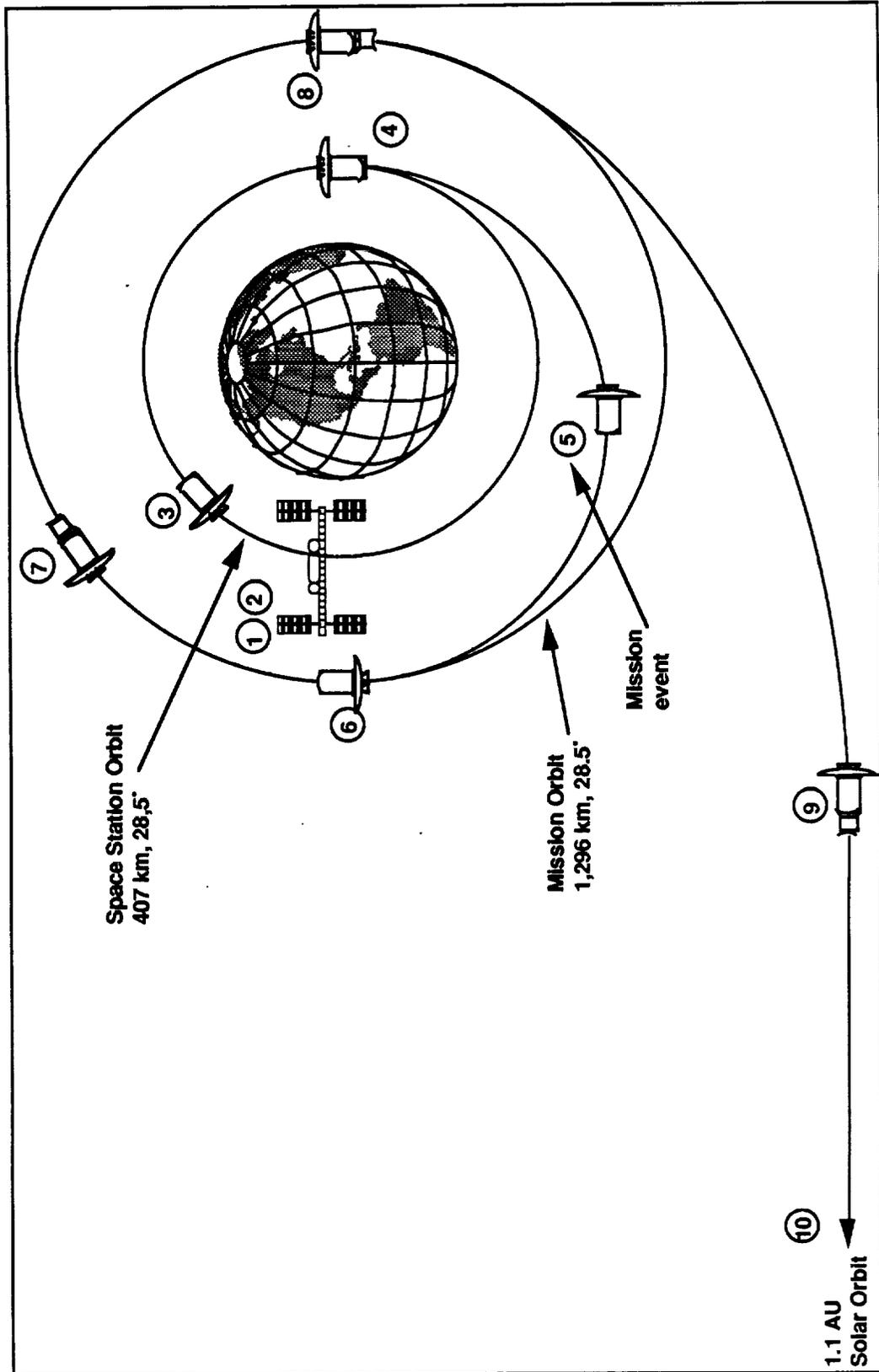
Figure 1-1.2.3.11-3. Nuclear Disposal DRM - N1 Mission Profile

Event	Δ V (m/s)	Δ T (hrs)
1) Docked at Space Station	0	0.0
2) ACS separation	3	0.0
3) ACS coast	3	0.8
4) MPS perigee burn	232	0.1
5) ACS coast	3	0.8
6) MPS circularization burn	225	0.1
7) ACS rendezvous, grapple P/L	10	1 day
8) MPS escape burn	3,010	0.2
9) ACS coast	5	196 days
10) MPS circularization burn (1.1AU)	684	0.1

ACS - attitude control system
MPS - main propulsion system

ΔV's do not include gravity or steering losses

Figure 1-1.2.3.11-4. Nuclear Disposal DRM - N1 Mission Timeline

First flight:	2002
Flight rate:	1 flight/year
Initial orbit:	SSF (407 km circular)
Final orbit:	407 km x 62968 km
	Inclination = 28.5°
Retrieval	
Manned	

* • Inert capsule with planetary sample (Mars regolith or comet material)
  • Manned recovery vehicle

Figure 1-1.2.3.12-1. Sample Return DRM - C1 Mission Objectives

The highly elliptical recovery orbit combined with a manned presence could also introduce potential driving requirements. In addition, the sample material could impose some additional contamination constraints on the system.

Mission Objectives. The mission objectives of DRM-C1 (Figure 1-1.2.3.12-1) were to capture and return a sample capsule returned to a highly elliptical Earth orbit by an unmanned planetary probe, such as Mars Rover Sample Return or Comet Nucleus Sample Return.

The first flight opportunity occurs in 2002, with the maximum flight rate of one flight per year.

The mission starts from the Space Station orbit, which is a circular orbit at an altitude of 407 km and an inclination of 28.5 degrees. The specified orbit for the capsule is taken directly from the STV mission model. The model also calls for a manned capability to retrieve the sample capsule.

Payload Requirements. The payload mass of 3,600 kg for the piloted module and 500 kg for the sample return capsule are taken from the STV mission model. The dimensions of the crew module are assumed to be the same as for the LECM. Overall payload requirements for the sample return design reference mission are shown in Figure 1-1.2.3.12-2.

The center of gravity of the payload is assumed to be at the center of the physical envelope of the payload. The power and data communications requirements are assumed to be supplied by the crew module. A manned crew cab is required, as is a fixture to capture the sample return capsule.

Mission Profile. Figure 1-1.2.3.12-3 is the mission profile for the sample capsule capture/recovery mission, DRM-C1. The numbered events correspond to the mission timeline shown in Figure 1-1.2.3.12-4. The STV separates from the Space Station and coasts to the proper point for the first burn, which puts the transfer vehicle into an elliptical orbit matching that of the sample capsule. The vehicle rendezvous with and attaches to the capsule. At apogee, a small burn is performed to reduce the perigee from 407 to 83 km. The STV

	Source
• Physical Description	

- • Physical Description
  - • Mass — 3,600 kg Manned Cab — STV Mission Model
  - 500 kg sample return — STV Mission Model
  - • Dimensions — Same as LECM — Estimated
  - • Center of gravity — Center of envelope — Estimated
- • Resource Requirements
  - • Power — 0 watts — (Supplied by crew module)
  - • Thermal protection — Maintain benign thermal attitude
  - • Data Communications — 0 discreet commands — (Supplied by crew module)
  - 0 kb/sec telemetry — (Supplied by crew module)
- • Support services
  - • Integration with STV at SSF
  - • Manned cab
  - • Capsule capture fixture
  - • Potential contamination constraints

Figure 1-1.2.3.12-2. Sample Return DRM - C1 Payload Requirements

Space Station Orbit
407 km, 28,5'
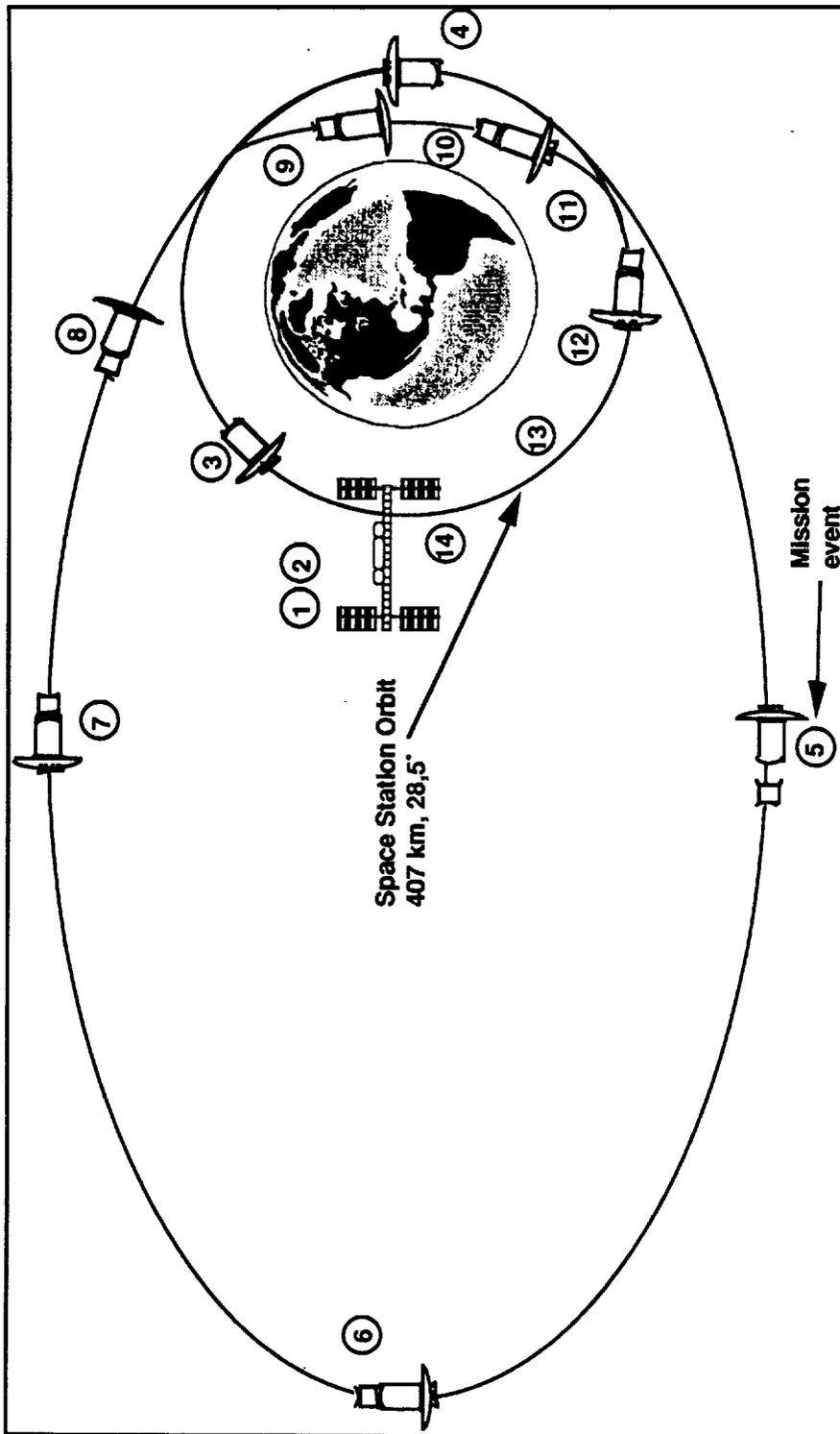
Mission
event

Figure 1-1.2.3.12-3. Sample Return DRM - C1 Mission Profile

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	Event	Δ V (m/s)	Δ T (hrs)		Event	Δ V (m/s)	Δ T (hrs)
1)	Docked at Space Station	0	0.0	8)	MPS pre aero correct	15	0.1
2)	ACS separation	3	0.0	9)	Aeromaneuver	0	0.1
3)	ACS coast	3	0.8	10)	MPS post aero correct	77	0.1
4)	MPS apogee raise burn	2,680	0.2	11)	ACS coast	3	0.7
5)	ACS orbital ops10		10.2	12)	MPS circularization burn	103	0.1
6)	MPS perigee lower burn	22	0.1	13)	ACS coast	3	0.8
7)	ACS coast	3	10.2	14)	ACS rendezvous and docking	12	1.0

ACS - attitude control system
MPS - main propulsion system

ΔV's do not include gravity or steering losses

Figure 1-1.2.3.12-4. Sample Return DRM - C1 Mission Timeline

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aerobrakes into an 407- by 56-km orbit and circularizes the orbit at 407 km. The transfer vehicle coasts to rendezvous with the Space Station.

Mission Timeline. The timeline shown in Figure 1-1.2.3.12-4 describes the sample capsule capture/recovery mission, including ΔV's and ΔT's.

1-1.2.3.13 Steady-State Piloted Mars DRM - M1

Selection Rationale. The manned Mars mission is the key evolutionary objective for the Space Transfer Vehicle. As discussed in the section 1-1.2.2, the Mars program listed in the CNDB FY89 was traded for the Mars program contained in the 90-day study report. One Mars design reference mission was desired that characterized the driving requirements for the piloted Mars missions. Figure 1-1.2.3.13-1 highlights the major requirement areas of interest.

The DRM was created based on typical Mars transfer trajectories being worked on in the Code Z STV studies. ΔV's requirements were taken from the Human Exploration Study Requirements document dated December 1, 1989.

Mission Objectives. The mission objective of DRM-M1 (Figure 1-1.2.3.13-2) was to deliver a crew of four and 25 tons of cargo to the Martian surface for a stay of 30 to 90 days and return the crew and a 1 metric ton of payload to LEO. The first flight leaves Earth orbit in 2015, with follow-on flights at 2-to 4-year intervals.

Figure 1-1.2.3.13-2 shows some typical trajectory parameters for a mission starting in 2015, using a Venus flyby on the outbound leg.

Payload Requirements. The payload mass of 24 metric tons was extracted from the Human Exploration Study Requirements document. The maximum dimensions of the payload are taken from the assumptions in the Boeing Code Z study.

The center of gravity of the payload is assumed to be at the center of the physical envelope of the payload. The Mars transfer vehicle (MTV) must support the crew for up to 3.5 years in transit. The Mars excursion vehicle (MEV) must

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- The Mars DRM should cover as many "tentpole" requirements as possible:
  - Initial launch date
  - Mission duration
  - Manned mission
  - Crew size
  - Payload capability
    - Mass
    - Volume
    - Resource Requirements
  - Surface stay time

Figure 1-1.2.3.13-1. Piloted Mars DRM - M1 Selection Rationale

First flight:	2015
Flight rate:	3 flights in 9 years
Initial orbit:	Space Station Freedom

Manned mission - crew of four

30 day surface stay

Crew and 1 metric ton returned to LEO

Typical mission: (2015, Venus swingby on outbound leg)

Earth Departure C3=16.63 km^2/sec^2

Mars Departure C3=30.69 km^2/sec^2

Trip Time=565 days

Figure 1-1.2.3.13-2. Piloted Mars DRM - M1 Mission Objectives

support the crew for up to 90 days on the Martian surface. The payload requirements are listed in Figure 1-1.2.3.13-3.

Mission Profile. The Mars mission profile for design reference mission DRM-M1 is shown in Figure 1-1.2.3.13-4. The numbered events correspond to the M1 mission timeline.

The MTV separates from the Space Station and performs any necessary pre-injection preparations and a MPS trans-Mars injection (TMI) burn puts the vehicle into an Earth-escape trajectory. After a 320-day coast with several trajectory correction maneuvers and a pre-entry correction, the MTV uses an aeroassisted maneuver to inset the entire vehicle into Mars orbit. The crew transfers to the MEV, which separates from the MTV and deorbits using an aerobraking reentry to descend to the Martian surface.

The MEV supports the crew for a 30- to 90-day stay on the surface, after which it ascends with the crew and return payload to orbit. The MEV and MTV rendezvous and dock, the crew transfers to the MTV, and the two vehicle separate. The MTV performs the TEI burn, putting the vehicle on a return trajectory. After a coast with several trajectory correction maneuvers and a pre-entry correction, the MTV uses an aerobraking maneuver to inset itself into Earth orbit, where the MTV rendezvous with the Space Station.

Mission Timeline. The mission timeline for the manned Mars mission, including ΔV's and ΔT's appears in Figure 1-1.2.3.13-5.

ITEM	REQUIREMENT
• Physical description	
• Payload mass	24.0 metric tons
• Payload volume	6 m x 13 m (maximum surface payload size)
• Payload center of gravity	Center of envelope
• Resource Requirements	
• Power	500 We
• Thermal protection	Maintain benign attitude during transit
• Data communications	TBD kbps telemetry
• Support services	
• MTV supports crew for up to 3.5 years in transit	
• MEV supports crew for up to 90 days on surface	

Figure 1-1.2.3.13-3. Piloted Mars DRM - M1 Payload Requirements
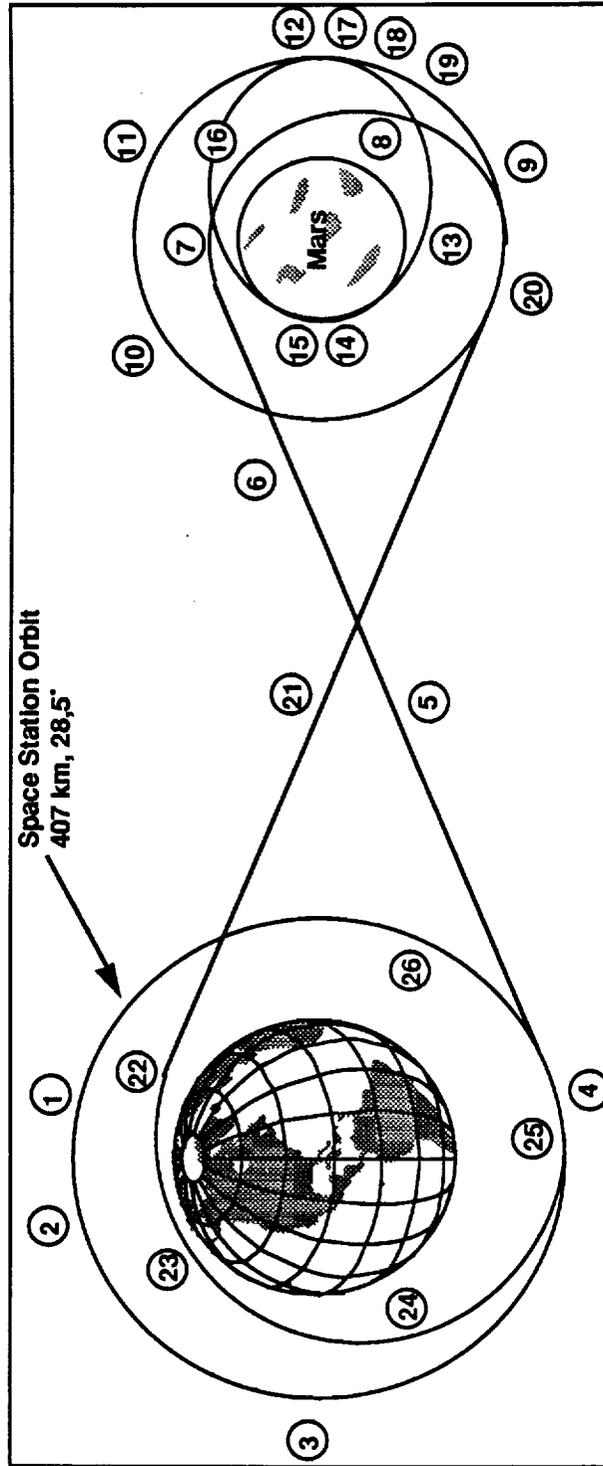
Figure 1-1.2.3.13-4.   Piloted Mars DRM - M1 Mission Profile

Event	Δ V (m/s)	Δ T (hrs)	Event	Δ V (m/s)	Δ T (hrs)
1) Vehicle assembly/verification		TBD	14) MPS Landing	200	0.1
2) ACS separation	3	0.0	15) Surface stay	0	30days
3) Pre-injection preparation	100	20 days	16) MEV ascent with crew	5,800	2.0
4) MPS TMI burn	4,600	0.3	17) MTV rendezvous/docking	25	2.0
5) ACS coast/corrections	50	320 days	18) Crew/payload transfer	0	2.0
6) ACS pre-entry correction	10	0.1	19) MTV/MEV separation	25	0.0
7) Aeroassist maneuver	0	0.1	20) MPS TEI burn	3,000	0.1
8) ACS post-aero correction	10	0.1	21) ACS coast/corrections	50	950 days*
9) MPS orbit circularization	50	0.1	22) ACS pre-entry correction	20	0.1
10) Crew transfer to MEV	0	1.0	23) Aeroassist maneuver	0	0.1
11) ACS MEV separation	10	1 day	24) ACS post-aero correction	20	0.1
12) MPS LEV deorbit burn	100	0.1	25) MPS orbit circularization	200	1.0
13) MPS Mars descent	1,100	2.0	26) SSF rendezvous/docking	12	1.0

ACS - attitude control system
MPS - main propulsion system

* Mars flyby abort

Figure 1-1.2.3.13-5. Piloted Mars DRM - M1 Mission Timeline

1-1.2.4   Detailed Mission Timelines

Timelines describing the sequence of events to perform a complete mission were developed at various times during the conduct of the study.

Design Reference Mission Timelines. The first set of mission timelines were developed to describe the 13 different DRMs and are include in section 1-1.2.3. These timelines were intended to provide quantitative information to determine the total propulsive energy requirements for the mission and to assist in the overall understanding of the mission architecture and requirements. The design reference mission timelines were not particularly detailed and were oriented toward the major propulsive events. The DRM timelines were developed during February 1990 for presentation at Interim Review Briefing #2 delivered in March 1990.

System Architecture Trade Study Timelines. Another set of mission timelines were developed to support the system architecture trade study during April and May 1990. Ten mission timelines were developed to characterize the sequence of events for the orbital options (shown in Figure 1-1.2.4-1) identified in the trade study.

Acronym	Mission Description
GB1	Ground Based - Single Launch
GB2	Ground Launched - LLO Node
SB1	Space Based - Direct To Surface
SB2	Space Based - LLO Node
SG1	Space & Ground Based - Direct
SG2	Space & Ground Based - LLO Node
GO1	Ground Based/LEO Assembly - Direct
GO2	Ground Based/LEO Assembly - LLO Node
EC1	Ground Based Single Launch - Expendable Cargo
EC2	Ground Based/LEO Assembly - Expendable Cargo

Figure 1-1.2.4-1.   Mission Timelines

The 10 timelines were developed for the nominal mission profiles. In addition, an abort mission timeline was developed for the space-based LLO node (SB2) mission. The SB2 mission was analogous to NASA's 90-day study scenario and was considered the reference mission at that time. The abort timeline assumed that a contingency occurred just prior to the deorbit burn to descend to the lunar surface. The crew was required to remain in LLO until the correct transfer orbit node alignment for return to Space Station Freedom.

Many assumptions were made to provide a basis for developing the mission timelines. Although these assumptions evolved over the performance period of the study, they provided a basic framework to develop the mission timelines that allowed for a comparison of various mission design and performance parameters such as crew support duration, total electrical energy requirements, and boiloff durations in support of the system architecture trade study. Most of the timelines assume steady-state piloted missions for the Option 5 lunar program with crew rotation through a man-tended base 6 months per year. The EC timelines are one-way expendable cargo missions.

The timelines assume a 6-hour time period to approach or depart Space Station Freedom. For LEO operations involving space-based hardware, the durations and operations were taken directly from the previous On-Orbit Assembly/Servicing Task Definition Study conducted by MDSSC. The timelines also assumed three orbits between injection into LEO (for ground-based systems) or separation from the LEO node (for space-based systems) and TLI similar to the Apollo operational flow.

The space-based scenarios and the on-orbit assembly scenario (GO) that require assembly of multiple flight elements in space assumed three launches to orbit all required hardware and propellant for a single lunar mission. This was the same assumption used during the 90-day skunkworks study. The GO option assumed that the crew is launched on the third mission and remains in the crew module during the on-orbit assembly and docking process.

The NASA trajectory profile was used with the added assumption that an all propulsive system would require 3,300 m/s to insert back into LEO at the end of

the mission. Twenty-four hours were dedicated to on-orbit maneuvers for the vehicle to achieve the same orbit as the LLO-based hardware. Forty-eight hours were allowed for the LEO maneuvers required to get back to Space Station. Earlier analysis of abort scenario timelines indicates that a worst case of 8 days may be required to achieve the proper landing geometry from LLO. In addition, a worst case of 8 days is assumed to wait in LLO for proper Earth return geometry.

The timelines assumed two orbits in LLO between vehicle separation and TEI. A 6-hour period was assumed to transfer propellant from the LOI and TEI droptanks into the core tanks of the return vehicle and also for LEO refueling operations. All rendezvous and docking maneuvers, considered separately from maneuvers required to match orbits with a target vehicle, were allotted 2 hours.

On the lunar surface the space transportation system was assumed to support the lunar personnel for 48 hours from landing and the lander vehicle will not receive any surface support for the first 30 days on the lunar surface as agreed to with the PSS team at JSC. The lunar mission was assumed to last 6 months as specified for the majority of the piloted missions in the Option 5 lunar program manifest.

Figure 1-1.2.4-2 provides a comparison of the total mission duration for the 10 missions analyzed. The 10 nominal mission timelines and one abort timeline can be found in Interim Review Briefing #3 data package presented in June 1990.

Timelines for Final Recommended Options. The system architecture trade study led to a downselection to three mission architecture options as described in section 2-1.0. During August and September 1990, expanded mission timelines were developed for these three missions and are described in the following paragraphs.
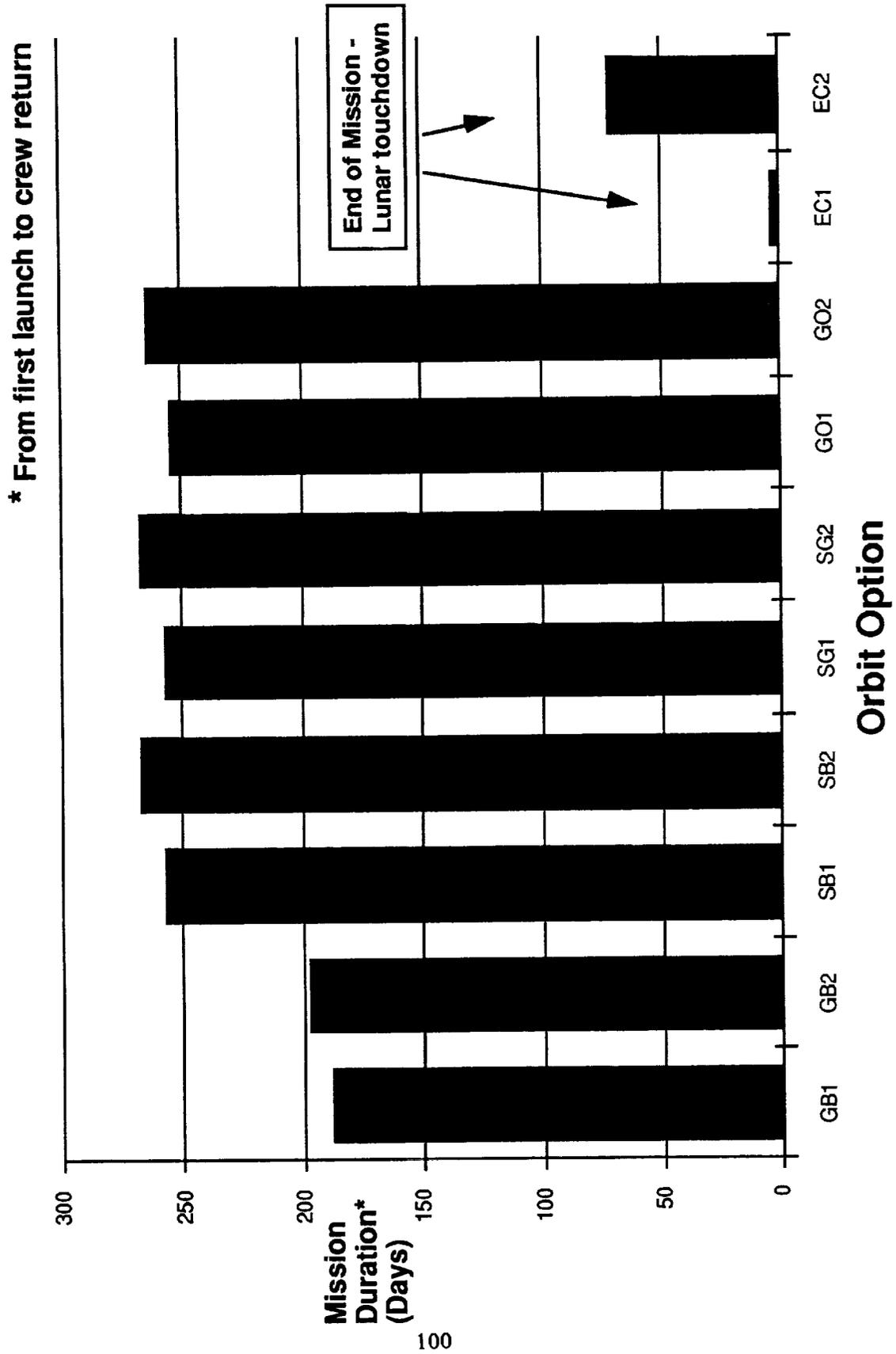
Figure 1-1.2.4-2.   Total Mission Duration

1-1.2.4.1 Space-Based Timeline

A number of product development team (PDT) meetings were held to establish each of the final three mission scenarios. Representatives each of the four major disciplines on this study (systems engineering, concept definition, interfaces, and programmatics) jointly agreed to the step-by-step details of all mission operations associated with the space-based and ground-based multiple launch and ground-based single launch options.

The timelines were expanded to include the sequenced mass of the entire vehicle and the ΔV requirements of each of the major propulsive events in addition to the duration of each event and mission elapsed time (MET). As with the timelines developed for the system architecture trade study, there were many assumptions made to develop the timelines. Most of the assumptions for this analysis were the same as the assumptions for the previous timelines.

The mission timeline for the space-based (SB) option is shown in Figure 1-1.2.4.1-1. The SB option assumed use of Space Station Freedom as the LEO transportation node but the timeline to any generic LEO node. The mission requires five launches to take the vehicle (this timeline assumes that the reusable space-based vehicle is not yet on orbit), crew module, aerobrake, various droptanks, cargo, and crew to orbit. The Advanced Launch System operational scenario was used to develop the last three mission timelines with 21 days between launches. The crew has a 30-day stay time on lunar surface.

The total mass of the vehicle as a function of time is plotted in Figure 1-1.2.4.1-2. The total mass of the vehicle after buildup on orbit is just over 250 metric tons. The plot show the slight boiloff that occurs during the storage periods on orbit.

1-1.2.4.2 Ground-Based Multiple Launch Timeline

The ground-based multiple launch option does not require support from a LEO transportation node. The vehicle is ground based and the crew returns directly to the Earth at the end of the mission. The mission timeline (Figure 1-1.2.4.2-1) indicates five launches to orbit and all required hardware and is assemble in

Event	Event Duration (Hours)	Mission E.T. (Hours)	Mission E.T. (Days)	Sequenced Mass (kg)	ΔV (m/s)
Launch #1 - Core, crew mod., and aerobrake	0.0	0.0	0.0	16791	
Storage at SSF (21 days between launches)	504.0	504.0	21.0	16791	
Launch #2 - LD droptanks, and PSS cargo	0.0	504.0	21.0	88264	
Storage at SSF	504.0	1008.0	42.0	87921	
Launch #3 - First TLI tankset	0.0	1008.0	42.0	158955	
Storage at SSF	504.0	1512.0	63.0	156500	
Launch #4 - Second TLI tankset	0.0	1512.0	63.0	227534	
Final vehicle integration	72.0	1584.0	66.0	227534	
Add protective water	0.0	1584.0	66.0	229334	
Fill core vehicle propellant tanks	4.0	1588.0	66.2	251119	
Flight readiness verification	96.0	1684.0	70.2	251119	
Vehicle closeout	192.0	1876.0	78.2	251119	
Launch #5 - Crew, crew consum. (STS)	0.0	1876.0	78.2	246788	
Crew ingress	1.0	1877.0	78.2	247879	
Final vehicle checkout	4.0	1881.0	78.4	247879	
Separate from LEO node	0.0	1881.0	78.4	247879	
Departure prox ops	4.3	1885.3	78.6	247879	
Wait for TLI node (up to one revolution)	1.5	1886.8	78.6	247879	
Trans Lunar Injection (TLI) burn	0.2	1887.0	78.6	120761	3300
Stage TLI droptanks	0.0	1887.0	78.6	110585	
TCM to "free return" trajectory	0.1	1887.1	78.6	110585	
Lunar transit	72.0	1959.1	81.6	110585	
TCM (target to landing site)	0.0	1959.1	81.6	110585	95
Lunar approach	12.0	1971.1	82.1	110585	
First lunar descent burn	0.2	1971.3	82.1	87085	1075
Low Lunar orbit coast (up to one revolution)	2.0	1973.3	82.2	87085	
Lunar landing burn	0.2	1973.5	82.2	54467	1920
Crew remains in transit module	48.0	2021.5	84.2	54467	
Crew transfer to habitat module	0.0	2021.5	84.2	53667	
Unload cargo	0.0	2021.5	84.2	43797	
Lander on surface with no surface support	672.0	2693.5	112.2	43497	
Lander activation, crew ingress with return P/L	4.0	2697.5	112.4	44797	
Transfer LD droptanks residuals to core tanks	1.0	2698.5	112.4	44797	
Drop descent tanks	0.0	2698.5	112.4	39967	
Lunar ascent burn	0.2	2698.7	112.4	26585	1822
Low Lunar orbit coast	2.0	2700.7	112.5	26585	
Second Lunar ascent burn	0.2	2700.9	112.5	19894	1075
Earth transit	72.0	2772.9	115.5	19894	16
TCM	0.0	2772.9	115.5	19894	
Earth approach	12.0	2784.9	116.0	19894	
Dump protective water	0.1	2785.0	116.0	18094	
Final TCM	0.1	2785.1	116.0	18094	
Aeroassist maneuver	0.1	2785.2	116.0	18094	
Coast	0.8	2786.0	116.1	18094	
Earth orbit circularization burn	0.1	2786.1	116.1	16821	310
LEO node rendezvous orbital maneuvers	48.0	2834.1	118.1	16821	
LEO node arrival prox ops	4.3	2838.4	118.3	16821	
Vehicle closeout	1.0	2839.4	118.3	16821	
Crew egress	1.0	2840.4	118.3	15730	
Remove Lunar payload	1	2841.4	118.4	15230	

E.T. - Elapsed Time
TCM - Trajectory Correction Maneuver

Figure 1-1.2.4.1-1. Space-Based Mission Timeline
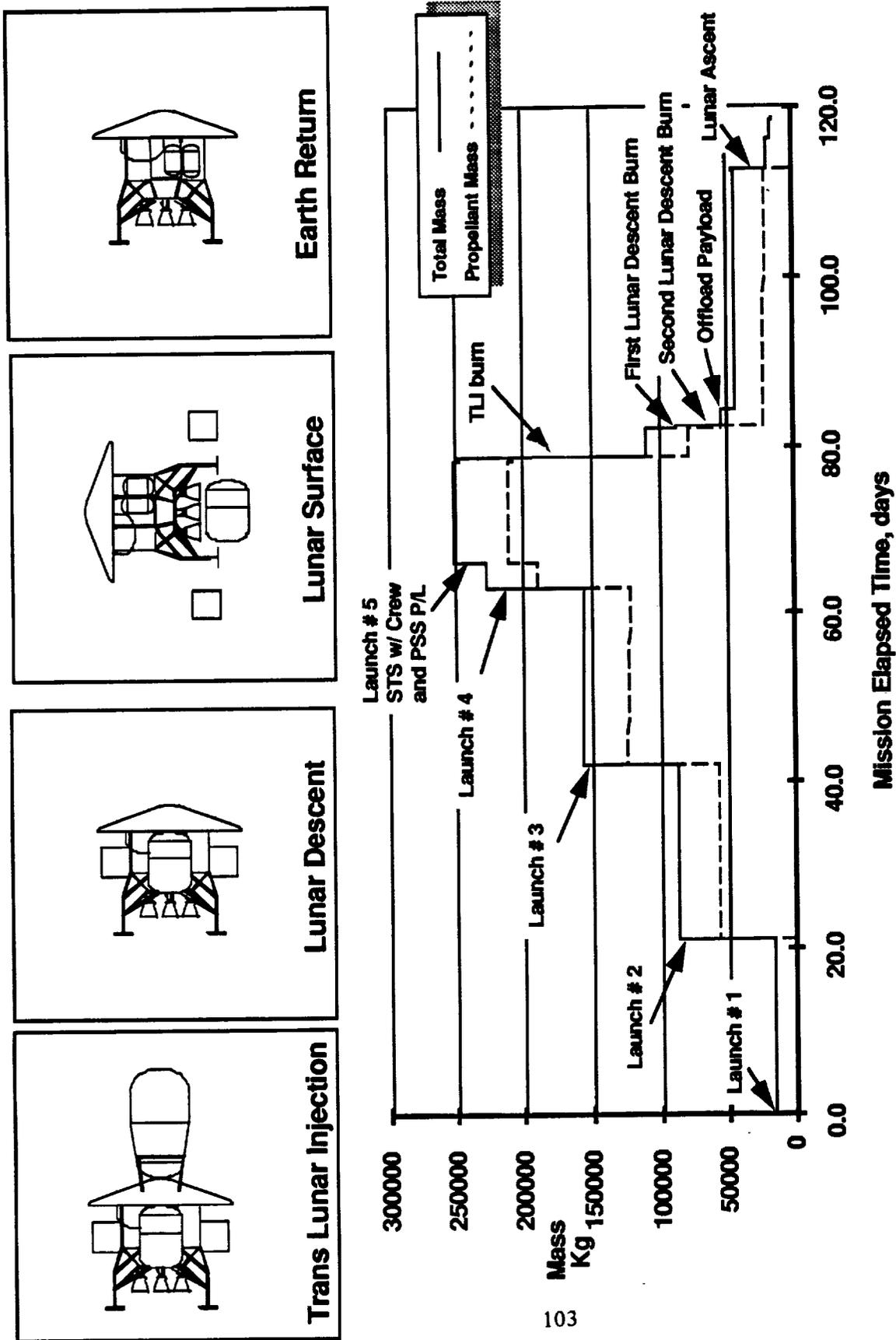
Figure 1-1.2.4.1-2. Space-Based Mission Mass Profile

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Event	Event Duration (Hours)	Mission E.T. (Hours)	Mission E.T. (Days)	Sequenced Mass (kg)	ΔV (m/s)
Launch #1 - Wet core and crew module	0.0	0.0	0.0	36922	
Storage in LEO (21 days between launches)	504.0	504.0	21.0	36833	
Launch #2 - LD droptanks	0.0	504.0	21.0	95466	
Storage in LEO	504.0	1008.0	42.0	95236	
Launch #3 - First TLI Tankset	0.0	1008.0	42.0	163643	
Storage in LEO	504.0	1512.0	63.0	161432	
Launch #4 - Second TLI Tankset	0.0	1512.0	63.0	229839	
Storage in LEO	72.0	1584.0	66.0	225620	
Launch #5 - Crew and PSS p/l (STS)	0.0	1584.0	66.0	225620	
Crew ingress	1.0	1585.0	66.0	226835	
Integrate PSS payload	6.0	1591.0	66.3	237465	
Flight readiness verification	96.0	1687.0	70.3	237465	
Final vehicle checkout	4.0	1691.0	70.5	237465	
Wait for TLI node (up to one revolution)	1.5	1692.5	70.5	237465	
Trans Lunar injection (TLI) burn	0.2	1692.7	70.5	115003	3300
Stage TLI droptanks	0.0	1692.7	70.5	106675	
TCM to "free return" trajectory	0.1	1692.8	70.5	106675	
Lunar transit	72.0	1764.8	73.5	106675	
TCM (target to landing site)	0.0	1764.8	73.5	106675	95
Lunar approach	12.0	1776.8	74.0	106675	
First lunar descent burn	0.2	1777.0	74.0	83084	1075
Low Lunar orbit coast (up to one revolution)	2.0	1779.0	74.1	83084	
Lunar landing burn	0.2	1779.2	74.1	53345	1920
Crew remains in transit module	48.0	1827.2	76.1	53345	
Crew transfer to habitat module	0.0	1827.2	76.1	52545	
Unload cargo	0.0	1827.2	76.1	41915	
Lander on surface with no surface support	672.0	2499.2	104.1	41615	
Lander activation, crew ingress with return P/L	4.0	2503.2	104.3	42915	
Transfer LD droptanks residuals to core tanks	1.0	2504.2	104.3	42915	
Drop descent tanks	0.0	2504.2	104.3	38523	
Stage landing gear	0.0	2504.2	104.3	37716	
Lunar ascent burn	0.2	2504.4	104.3	25216	1822
Low Lunar orbit coast	2.0	2506.4	104.4	25216	
Second Lunar ascent burn	0.2	2506.6	104.4	19801	1075
Earth transit	72.0	2578.6	107.4	19801	16
TCM	0.0	2578.6	107.4	19801	
Earth approach	12.0	2590.6	107.9	19801	
Dump protective water	0.1	2590.7	107.9	18001	
Final TCM	0.1	2590.8	107.9	18001	
Stage core vehicle	0.1	2590.9	108.0	11247	
Coast	0.8	2591.7	108.0	11247	
Ballistic entry and touchdown on Earth	0.1	2591.8	108.0	11247	

E.T. - Elapsed Time
TCM - Trajectory Correction Maneuver

Figure 1-1.2.4.2-1. Ground-Based Multiple Launch Mission Timeline

Earth Return

Lunar Surface

Lunar Descent

Trans Lunar Injection
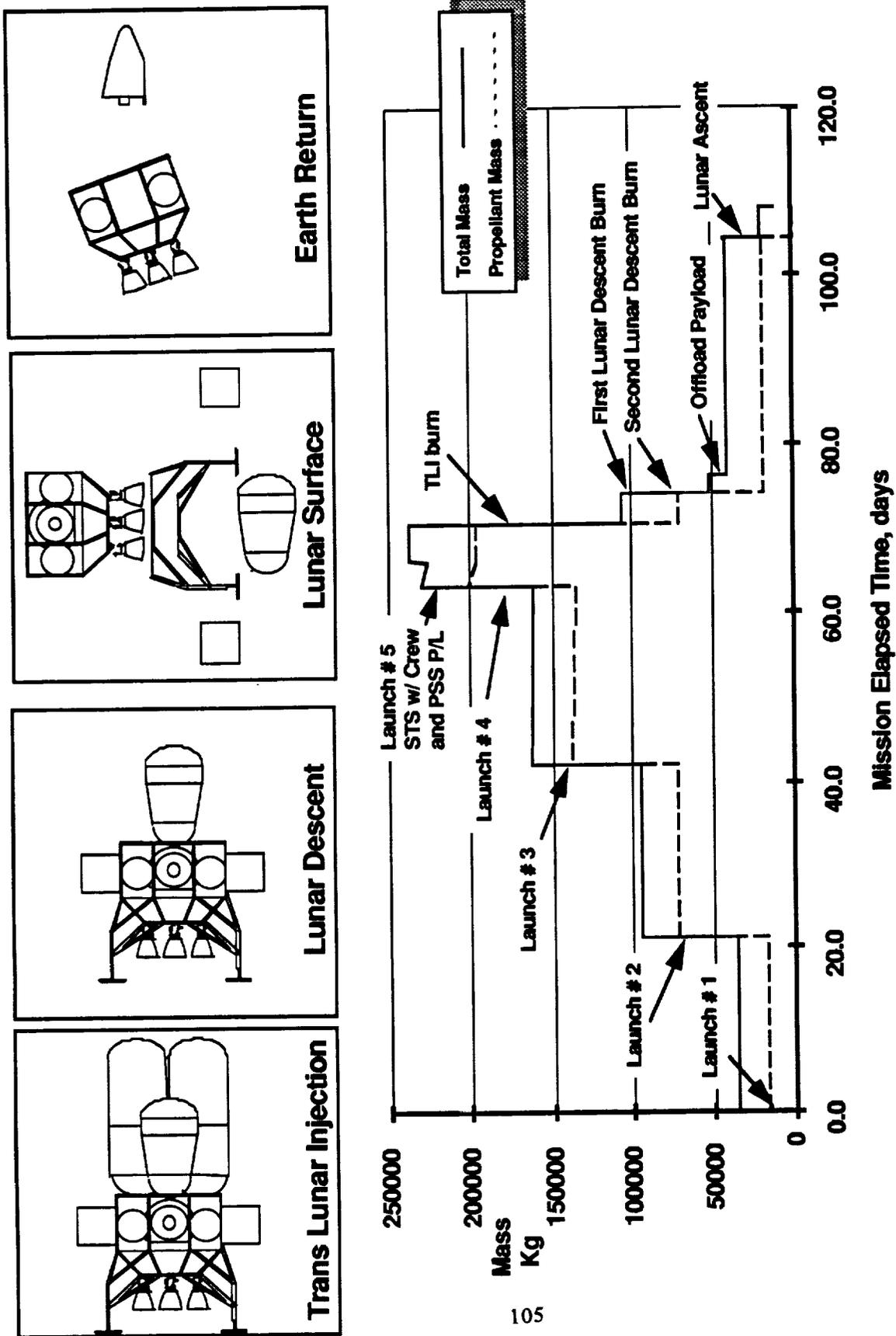
Figure 1-1.2.4.2-2.   Ground-Based Multiple Launch Mission Mass Profile

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LEO. The same planet surface system interface assumptions were used on all three final mission timelines. The sequenced mass of the total vehicle over the entire mission duration is shown in Figure 1-1.2.4.2-2.

1-1.2.4.3 Ground-Based Single Launch Timeline

The ground-based single launch option assumes the entire vehicle is launched at one time. The vehicle requires no assembly in LEO, which results in a greatly simplified mission timeline (Figure 1-1.2.4.3-1). The vehicle is ground based and the crew returns directly to the Earth at the end of the mission. The sequenced mass of the entire vehicle is shown in Figure 1-1.2.4.3-2.

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Event	Event Duration (Hours)	Mission E.T. (Hours)	Mission E.T. (Days)	Sequenced Mass (kg)	ΔV (m/s)
Launch	0.2	0.2	0.0	235064	
Coast in LEO (3 revolutions)	4.5	4.7	0.2	234686	
Trans Lunar Injection (TLI) burn	0.2	4.9	0.2	116788	3300
Stage TLI droptanks	0.0	4.9	0.2	107369	
TCM to "free return" trajectory	0.1	5.0	0.2	105246	
Lunar transit	72.0	77.0	3.2	105246	
TCM (target to landing site)	0.0	77.0	3.2	103123	95
Lunar approach	12.0	89.0	3.7	103123	
First lunar descent burn	0.2	89.2	3.7	81887	1075
Low Lunar orbit coast (up to one revolution)	2.0	91.2	3.8	81887	
Lunar landing burn	0.2	91.4	3.8	54166	1920
Crew remains in transit module	48.0	139.4	5.8	54166	
Crew transfer to habitat module	0.0	139.4	5.8	53366	
Unload cargo	0.0	139.4	5.8	42296	
Lander on surface with no surface support	672.0	811.4	33.8	41996	
Lander activation, crew ingress with return P/L	4.0	815.4	34.0	43296	
Transfer LD droptanks residuals to core tanks	1.0	816.4	34.0	43296	
Drop descent tanks	0.0	816.4	34.0	38058	
Stage landing gear	0.0	816.4	34.0	37130	
Lunar ascent burn	0.2	816.6	34.0	25274	1822
Low Lunar orbit coast	2.0	818.6	34.1	25274	
Second Lunar ascent burn	0.2	818.8	34.1	20199	1075
Earth transit	72.0	890.8	37.1	20133	16
TCM	0.0	890.8	37.1	20133	
Earth approach	12.0	902.8	37.6	20133	
Dump protective water	0.1	902.9	37.6	18333	
Final TCM	0.1	903.0	37.6	18333	
Stage core vehicle	0.1	903.1	37.6	12192	
Coast	0.8	903.9	37.7	12192	
Ballistic entry and touchdown on Earth	0.1	904.0	37.7	12192	

E.T. - Elapsed Time
TCM - Trajectory Correction Maneuver

Figure 1-1.2.4.3-1. Ground-Based Single Launch Mission Timeline

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Trans Lunar Injection

Lunar Descent

Lunar Surface

Earth Return
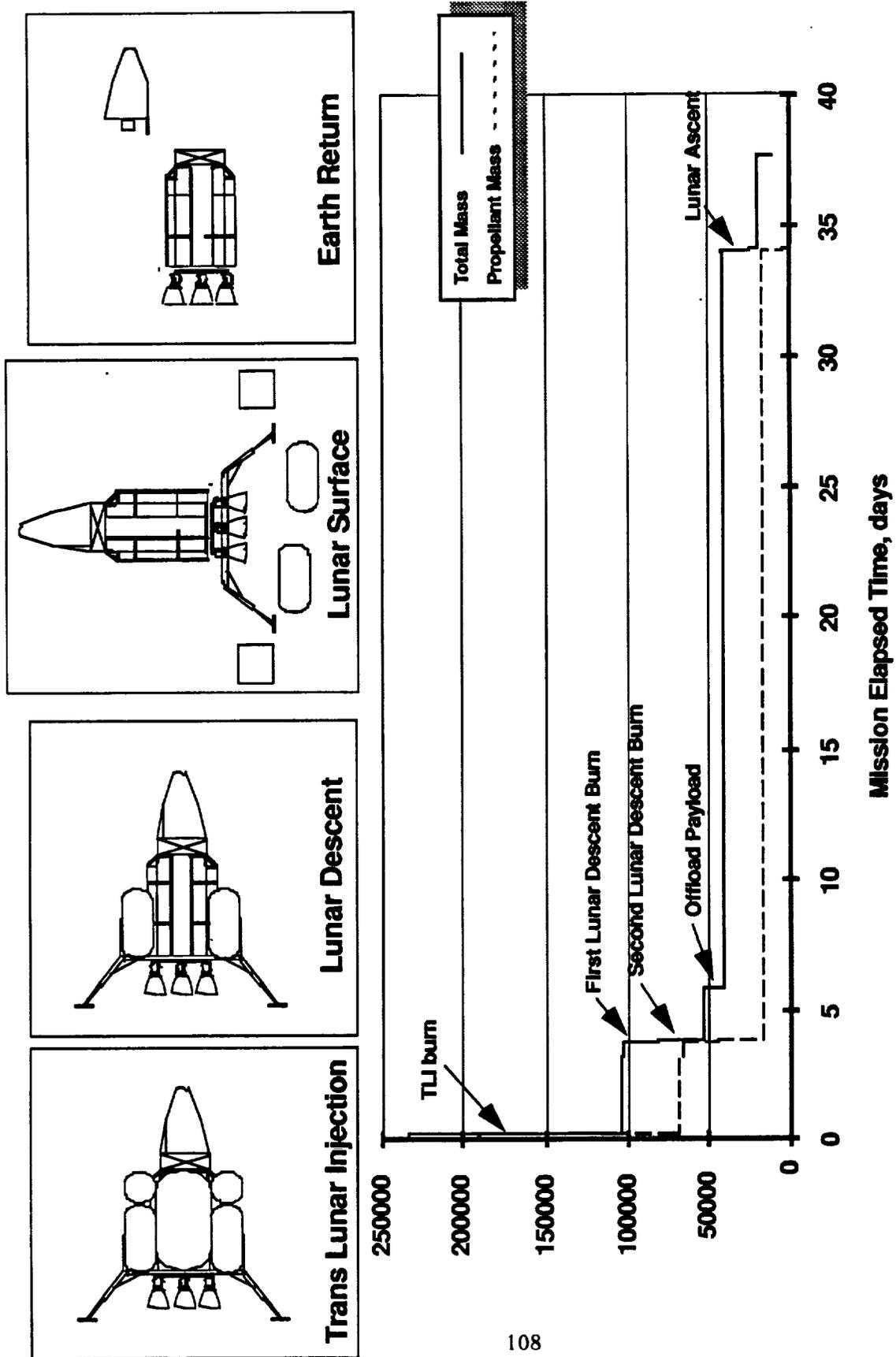
Figure 1-1.2.4.3-2. Ground-Based Single Launch Mission Mass Profile

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1-1.2.5 Mission Drivers and Characteristics

1-1.2.5.1 Lunar Mission Drivers and Characteristics

In an effort to include all the pertinent requirements and drivers of the Option 5 lunar scenario, the manifest was analyzed flight by flight to draw out unique operational and design characteristics and requirements (Figure 1-1.2.5.1-1). These characteristics were not only used as the basis for the derivation of the four lunar design reference missions, but they also served as direct input in the vehicle design process.

Flight 0 stresses the STV system in a number of requirement categories in addition to being the first flight to land at the lunar base. Although precursor missions could install navigational landing aids for the first lunar lander to track during descent to the surface, the manifest used as the basis for this study does not include homing devices for Flight 0. The unmanned cargo vehicle must be capable of landing autonomously on the lunar surface. This would include the capability to land on an unprepared surface with potentially limited knowledge of the detailed characteristics of the landing site. The lunar orbiter (LO) spacecraft, currently scheduled for launch in 1997, would provide landing site reconnaissance, but the lunar lander would still be required to autonomously avoid boulders and craters smaller than the resolution of the orbiter. During transfer from LEO to the lunar surface some scenarios include staging of hardware in LLO. The operations required to affect the separation would also be autonomous for the cargo delivery missions.

The payload unloading scheme was a vehicle design driver. Many schemes exist for the design of the planet surface system (PSS) to be used during the operational phases of the lunar program. All of these systems must first be transported to the lunar surface as payload. The first flight requires autonomous unloading of the PSS payloads.

During the course of this study the reference PSS scheme included a large payload unloader to be mounted on top of the lunar lander. After touchdown on the lunar surface, the payload unloader would autonomously deploy by extending three legs to the surface, raising up on the legs thus separating the

main body of the payload unloader from the lander, and finally rolling away from the vehicle.

Many other concepts were investigated during the study to determine the alternatives available for mounting payloads in a single location or on two sides of the lunar landing vehicle. The various concepts are illustrated and described in the section 1-3.0.

Included on Flight 0 with the payload unloader were a set of attachments and tools for the unloader, excavation pyrotechnics, communications equipment, and an unpressurized manned/robotic rover with a total volume of over 300 m^3, the largest volume required of any mission. A full description of the cargo manifested on Flight 0 and Flight 2 can be found in Figure 1-1.2.3-4.

Flight 1, also an expendable cargo flight, included a large nuclear power source as part of the payload compliment. This payload would require special viewing of deep space or other means of thermal control to assist in rejecting the large amounts of thermal energy generated during transit to the lunar surface. This mission also had the highest total payload mass of all 25 missions listed in the Option 5 manifest. As previously stated, early in the study the STV designs were sized for a 13 metric ton capability for steady-state piloted missions using an LTV and LEV. Later in the study when a downselection was made to one and a half stage vehicles, the cargo capability of the LTV was optimized for the sum total of cargo to be taken to the lunar surface (~420 metric tons). In the case of the space-based one and a half stage design, the split between cargo on piloted missions and cargo expendable missions works out to 10 and 50 metric tons rather than the 13 and 33 metric ton split for the separate LTV/LEV approach.

The lunar payload delivery requirements were probably the single most driving factor for overall STV sizing. A performance comparison was made of the ground-based STV concept to Apollo to better understand the differences (Figure 1-1.2.5.1-2). To decouple TLI differences, the vehicle mass prior to lunar orbit insertion is shown as a function of lunar-delivered cargo.

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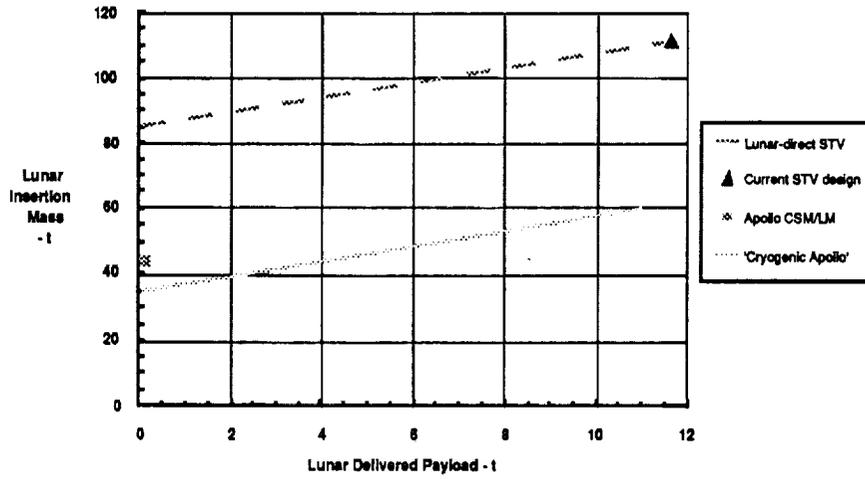
Figure 1-1.2.5.1-2.   Lunar Insertion Mass Versus Delivered Payload

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The current STV piloted concept has no lunar-orbit rendezvous and delivers 11.6 tons of cargo to the lunar surface, with a lunar insertion mass of about 111 tons. Even by reducing the STV delivered cargo to 0.1 tons, the lunar insertion mass is still 42 tons greater than that of Apollo.

The next step was to analyze an Apollo-type vehicle with lunar-orbit rendezvous and inert mass, crew size, and mission duration similar to Apollo but with cryogenic propellants and advanced engines (Isp= 481 seconds). This "cryogenic Apollo" concept delivers an Apollo-type cargo with a lunar insertion mass less than Apollo. From a performance-only point of view, it can be seen that crew size, mission duration, and lunar-direct mission type significantly impact the size of the vehicle.

Flight 2 is the first piloted mission to the lunar surface. The early flights do not have the advantage of a servicer available on the lunar surface. A groundrule of no transportation system support from PSS elements for up 30 days on the lunar surface was applied to the design. Since the early missions are all only 30 days in duration the vehicle will be capable of self support for each of these missions.

This flight requires the first ascent and return to Earth of both crew and return cargo. The requirement to transport and protect a team of four crewmembers was also a considerable design driver. A summary of these and other crew module requirements appears in Figure 1-1.2.5.1-3. The resource requirements, including power for operation of the ECLSS, telemetry and command, and communications for the piloted missions, drive the overall vehicle design. In an abort scenario the vehicle must support the crew for a total of 26 to 32 days. The large overall mass and volume of the crew module significantly influence the layout of the vehicle. In addition, the crew modules required certain viewing capabilities to provide line-of-sight viewing for pilots during critical vehicle operations such as lunar landing (Figure 1-1.2.5.1-4) or in space rendezvous and docking.

In the case of the scenarios that use a lunar excursion vehicle, the LEV would be left in orbit after the mission and reactivated and reused on Flight 3. Upon

	Mass	Volume	Crew	Provisions	Comments
Command Module	empty= 5091 kg Flt wt= 5649 kg	80 cu ft per person	3	12 days	• No radiation protection • Fecal bags • Ocean recovery
Space Based	empty= 4492 kg Flt wt= 7383 kg	180 cu ft per person	4	32 days	• Radiation protection • Commode • Galley • No TPS, RCS
Ground Based	empty= 8333 kg Flt wt= 11302 kg	180 cu ft per person	4	12 days	• Radiation protection • Commode • Galley • Parafoil ground recovery
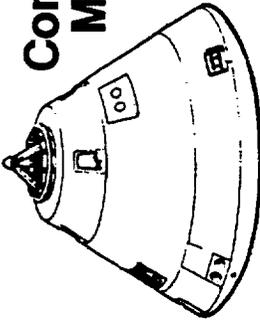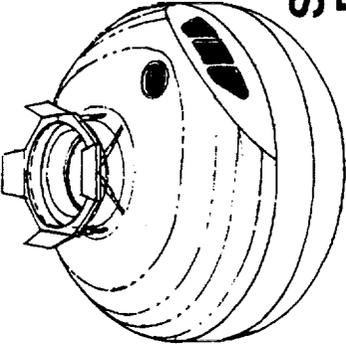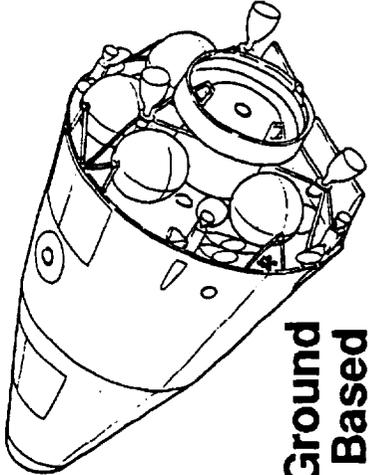
Figure 1-1.2.5.1-3. Crew Module Requirements

return to Earth, the crew module would introduce a first use of the aerobrake system for the space-based vehicles or first use of the ballistic crew module for the ground-based scenarios.

Flight 3 introduces rendezvous, docking, and transfer of crew, cargo, and potentially propellant from the LTV to the LEV for the scenarios using low Lunar orbit as a transportation node.

Flight 4 places the laboratory module containing a health maintenance facility on the lunar surface that could potentially alleviate the need to provide some health care facilities in the transportation system. Flight 5 emplaces the LEV servicer and extends the lunar surface stay for the lander to 90 days.

Flight 8 takes a logistics module requiring up to 3,000 watts of power during transit to the lunar surface. The baseline approach is to provide 500 watts of power as a standard service with the lunar lander and provide the additional power with kits. The payload carrying capability would be reduced while carrying the additional power kits.

Flight 11 introduced a number of logistical requirements and constraints to design the vehicles to be capable of supporting a 600-day lunar stay with a resupply flight 1 year into mission.

Flight 20, intended to explore the far side of the lunar surface, would also have introduced additional capabilities for the vehicle but was deleted from the reference manifest by the customer.

A summary of the heaviest and largest PSS payloads is shown in Figure 1-1.2.5.1-5. The laboratory, logistics, and habitat modules are the largest and heaviest single payloads on the lunar manifest. The 16-meter constructible is the largest and heaviest payload when considered in total, but the assembly is broken down into smaller subpackages for delivery to the lunar surface.
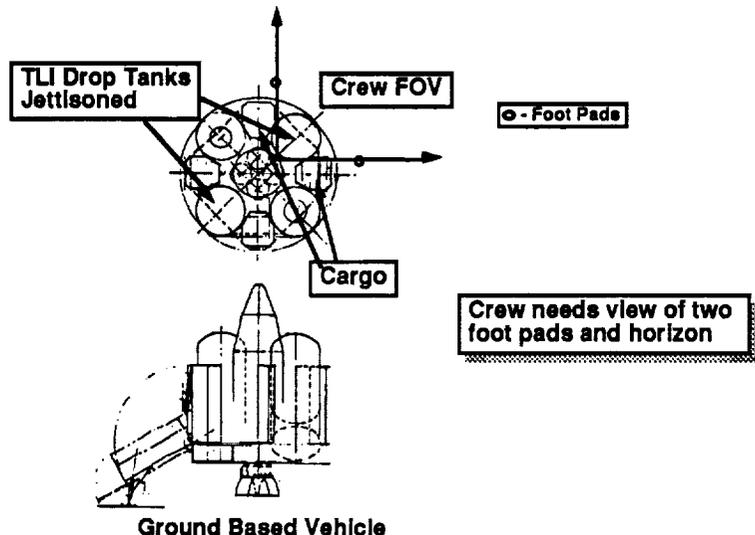
Figure 1-1.2.5.1-4. Ground-Based Vehicle Viewing Angles

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	5.5	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

Figure 1-1.2.5.1-5. Driving Planet Surface System Payloads

1-1.2.5.2 Other Mission Drivers and Characteristics

The STV mission model was analyzed to determine the mission characteristic categories required to adequately cover the payloads that drive vehicle design and interface requirements. The selection process is covered in section 1-1.2.2. The selection rationale for all the design reference missions is contained in the detailed DRM descriptions in section 1-1.2.3. Figure 1-1.2.5.2-1 summarizes the payload characteristics of the non-lunar design reference missions.

The DoD missions introduce special design requirements for handling and operating classified payloads. For classified missions requiring assembly in LEO at a node shared with civil operations, the facilities would require TEMPEST upgrades. The molniya mission to be flown by an initial STV configuration includes a transfer requiring three distinct burns with two restarts of the cryogenic engines.

The space tug mission introduces requirements for payload grappling capability, man-machine interface supporting teleoperation, and small impulse bit operations during rendezvous and docking operations.

The unmanned servicing missions require support of an autonomous payload servicing kit along with an autonomous rendezvous and docking capability. This could be a new requirement introduced by the polar servicing mission since some of the lunar mission scenarios that use a single P/A module do not require an autonomous rendezvous and docking capability.

The manned capsule recovery mission and manned geosynchronous servicing missions require a man-rated vehicle with two failure tolerance against catastrophic failures and single failure tolerance for mission critical failures.

A summary of the top-level functional requirements for the Space Transfer Vehicle appears in Figure 1-1.2.5.2-2. The functional requirements do not apply to all three lunar scenarios remaining at the end of the study. The figure indicates how each requirement is allocated to the various mission scenarios.

Mission	Transit Mass, mt	Stowed Volume, m3	Transit Power, Watts	Transit Thermal Rejection, Watts
Planetary (CNSR)	16.0	200	900	0
Geosynchronous Delivery (MTPE)	10.0	32	600	0
Molniya Delivery	6.0	21	600	0
Manned GEO Servicing (MTPE)	7.7	TBD	Supplied by crew module	0
Polar Platform Servicing	4.5	TBD	Supplied by servicing kit	0
LEO Space Tug (TLI Tank)	71.0	132	0	0
Nuclear/Debris Disposal	25.0	320	500	0
Manned Capsule Recovery	4.1	TBD	0	0
Manned Mars	24.0	117	500	0

Figure 1-1.2.5.2-1. Driving Non-Lunar Payload Requirements

	Functional Requirement	Space Based	Ground Based Multiple Launch	Ground Based Single Launch
#	Description			
1	Provide large Δ V's	X	X	X
2	Assembly/checkout in LEO	X	X	
3	Reusable vehicle systems	X		
4	Long life on orbit	X		
5	Autonomous Lunar landing	X	X	X
6	Unload Lunar payload	X	X	X
7	Support satellite servicing	X	X	X
8	Retrieve payload from orbit	X	X	X
9	Aeroassisted return to LEO	X		
10	Rendezvous/docking with LEO node	X		
11	Ballistic reentry to Earth		X	X

Figure 1-1.2.5.2-2. Top-Level Functional Requirements

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1-1.2.6   Mission Capture Results

Early Mission Capture Results. In the spring of 1990, the study emphasis was on optimizing the 90-day study mission architecture that used a separate LTV and LEV. Mission capture analyses were conducted to evaluate the capability of each lunar STV concept developed at that time and presented in the Interim Review Briefing #2 to meet other evolutionary missions.

The core elements were analyzed to determine which pieces are necessary for each evolutionary mission and the modification capability of those elements. Standard interface provision requirements for the evolutionary DRM payload elements will be assessed for STV core element scar, EVA provisions, and power and fluid transfer provisions. In addition, each STV concept was evaluated for IMLEO propellant required for each evolutionary mission performance.

Figure 1-1.2.6-1 highlights the results of the analysis. The first data block summarizes the required STV core elements necessary to perform each mission. The second data block summarizes evolutionary DRM payload elements required by each mission. The third data block summarizes IMLEO propellant required for mission performance. This sample evaluation uses vehicle concept 1 from Interim Review Briefing #2 core element sizing.

Architecture Trade Study Mission Capture Results. Mission capture was used as one of the four main evaluation criteria in the system architecture trade study. All candidate architectures were evaluated for their ability to capture the requirements of the non-lunar design reference missions. The concepts were scored both by stage efficiency and Earth-to-orbit launched mass. Details of the mission capture assessment can be bound in section 2-1.1.3.

Final Mission Capture Results. A final mission capture analysis was conducted to determine how well the final ground-based and space-based STV concepts capture other CNDB missions. The missions are based on the 1989 CNDB and include those shown in Figure 1-1.2.6-2. For this analysis, it was assumed that only the ground-based vehicles have a return leg, and of the

Vehicle Concept Variable Evaluation Configuration #1	Planetary P1	GEO Delivery G1	Molniya Delivery D1	GEO Servicing G2	Polar Platform Servicing S1	LEO Space Tug T1	Nuclear/Debris Disposal N1	Capsule Capture Recovery C1
· Lunar STV Core Elements								
· Droptank set (TLI)	yes	yes	yes	yes	yes		yes	yes
· Droptank set (Insert)					yes			yes
· Aerobrake	yes	yes	yes	yes	yes			yes
· P/A Module	yes	yes	yes	yes	yes	yes	yes	yes
· Crew Cab				yes				
· Evolutionary DRM Payload Elements								
· One-Piece Payload, Delivered (KG)	16000	10000	5000					500
· Payload Capture Fixture, Small Aerobrake Returned P/L (KG)								
· Payload Capture Fixture, Large No Aerobrake Required, P/L (KG)						71000	25000	
· Servicing Platform				yes	yes			
· Servicing Tankage/Fluid Transfer				yes	yes			
· RMS				yes	yes			
· Automated/Telerobotic Servicer				yes	yes			
· MMU								yes
· Droptanks (small)								
· Power supply requirement (Watts)	900	600	600				500	
· Performance								
· Total Propellent Required (KG)	60665	91152	49067	69245	121450	431	53840	13604
· Performance Growth Capability	yes	yes	yes	yes	yes	yes	yes	yes
· Mission Capture Achieved ?	yes	yes	yes	yes	yes	yes	yes	yes

Figure 1-1.2.6-1. Mission Capture Analysis Matrix

Objective:

- Determine how well the reference space-based and ground-based STV concepts, designed primarily for lunar missions, capture other piloted and unpiloted missions in the Civil Needs Data Base.

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 1-1.2.6-2. Mission Capture Analysis

space-based vehicles only the piloted and polar platform servicing missions return to the LEO node.

The cargo masses shown for the piloted missions do not include the crew module or other returned non-payload hardware. The lunar-delivered cargo masses are those given in the CNDB FY1989. The STV designs using a direct to the lunar surface trajectory have a different cargo split between piloted and cargo-only flights, based on an optimized performance split.

The final ground-based lunar vehicle concept includes a crew module with an avionics/power pallet, a core stage with tankset and propulsion module, two strap-on delivery stages, two strap-on TLI tanksets, and a lander platform. The delivery stages and TLI tanksets have common tank sizes and support structure. A summary of the vehicle element configurations and mass properties appears in Figure 1-1.2.6-3.

These flight elements are modular in design to capture other CNDB missions prior to the lunar missions. For unpiloted delivery missions prior to advanced engine development, a single delivery stage with an avionics/power pallet and RL10 engine can be used as a delivery stage. For non-lunar piloted missions, the ascent stage with crew module and avionics/power pallet can be used. For greater capability, an ascent stage with avionics/power pallet and two delivery stages can be integrated onto a lander platform. Capabilities of these configuration options are shown in following charts.

The descent strap-on for ground-based configurations can be used with little modification to capture the non-lunar missions (Figure 1-1.2.6-4). This element provides a 25 metric ton P/L capability to GEO with a 45 metric ton propellant mass. The modifications to this strap-on from the lunar mission configuration include the addition of avionics and an interface for the payload.

Cargo delivery capabilities of various configurations of the ground-based STV concept are given by the curves in Figure 1-1.2.6-5. Also shown are the CNDB mission payloads and ΔV's.

Flight Element Mass Summary

Flight Element	Inert Mass (kg)	Propellant Mass (kg)
Crew Module / Avionics	11474	50
- with Protection	13274	50
Core Tankset	2240	17640
Core P/A Module	1791	0
Delivery Stage (1)	3465	45440
- with Avionics	4908	45440
Lander	3832	0
TLI Tankset (1)	2950	45440

Figure 1-1.2.6-3. *Ground-Based Concept Description*

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- 25 mt P/L capability to GEO
- 45 mt propellant mass

Single engine, single tankset vehicle: add avionics, P/L interface

Descent "strap-on" (2)
(tank, plumbing, engine)
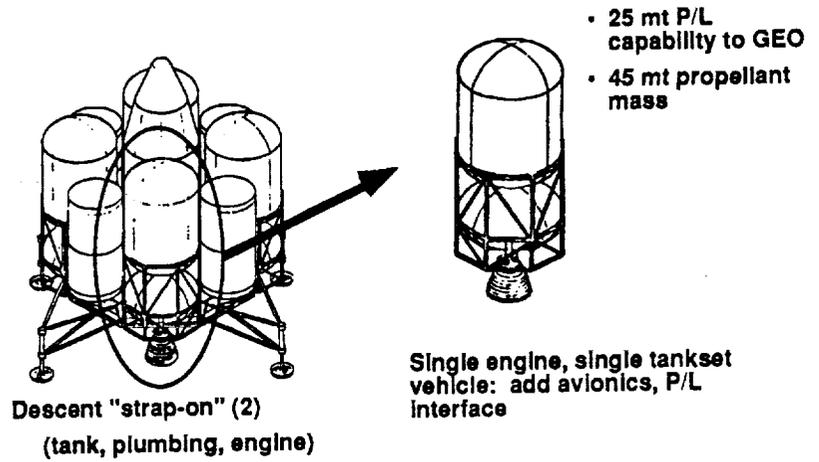
CNDB Missions are captured with little modification to descent strap-on

Figure 1-1.2.6-4. Use of Lunar Elements for CNDB Missic
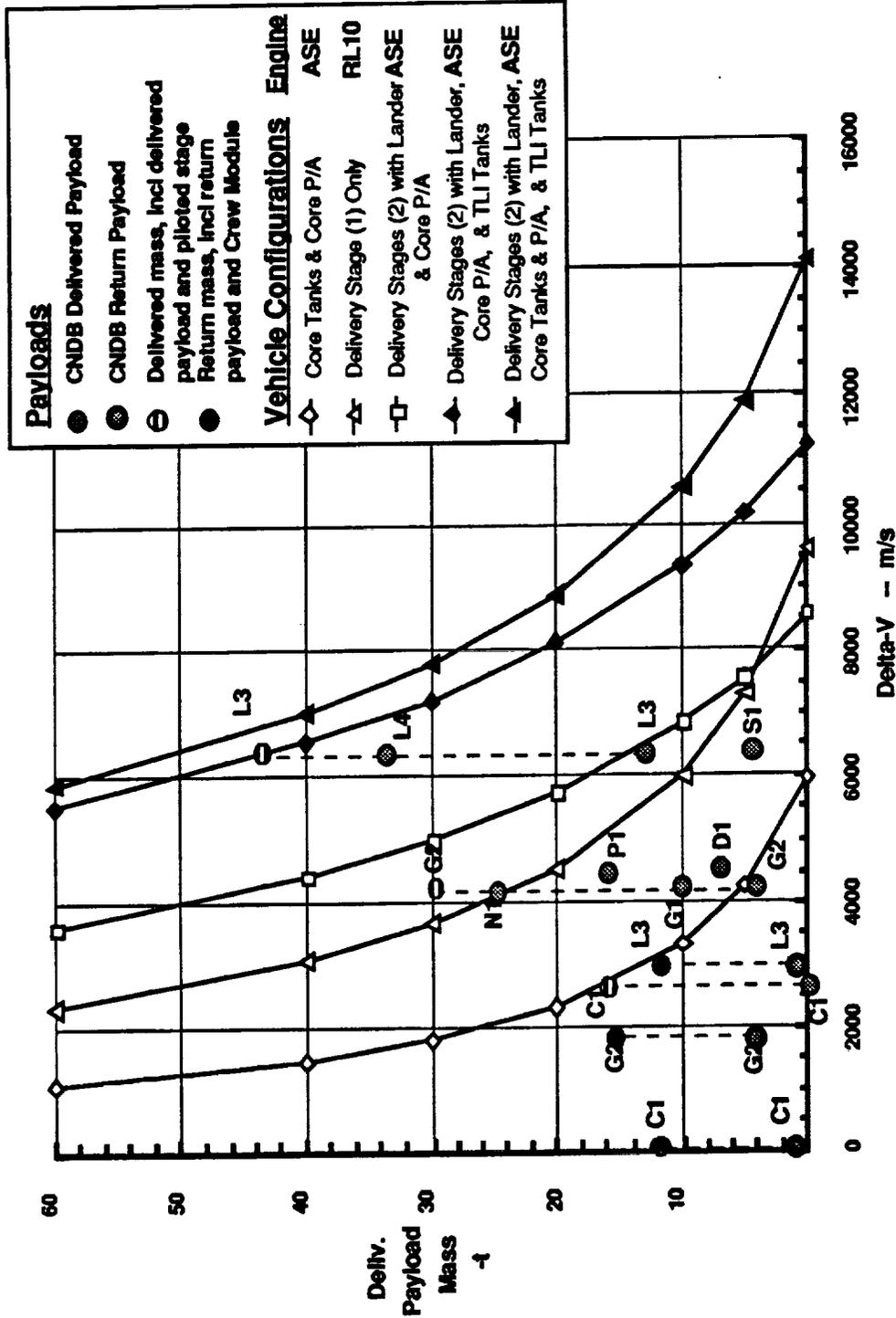
Figure 1-1.2.6-5. Ground-Based Vehicle Capabilities

CNDB Missions are Captured by Components of a Lunar Mission-designed Vehicle.

Unpiloted delivery missions are shown as single points on the chart and are captured by a single delivery stage with RL10 except for the lunar cargo delivery mission, which requires the lunar vehicle with advanced engines. Piloted missions are shown with dashed lines connecting delivered mass to a delivered payload and return mass to return payload quantities. The sample return mission (C1) is captured completely by the ascent stage. 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 a lander platform is required. To deliver the lunar return stage and lunar cargo, the full lunar vehicle is required.

For the lunar missions, the CNDB payload quantities are shown, but the vehicle concept is sized for an optimized payload split of 43.4 tons for the cargo-only flight and 11.6 tons for the piloted flight.

The current space-based lunar vehicle concept includes an aerobrake, crew module, core stage with tanks, propulsion, landing gear, two strap-on descent tanksets, and two strap-on TLI tanksets. These flight elements, shown in Figure 1-1.2.6-6, can be used in different combinations to capture other CNDB missions. For unpiloted delivery missions and aerobrake tests prior to advanced engine development, a core stage with two RL10 engines and without landing gear can be used as a delivery stage. For non-lunar piloted missions, the core stage with crew module and aerobrake can be used. For greater capability, descent and TLI tanksets can be added. Capabilities of these configuration options are shown in Figure 1-1.2.6-7.

Cargo delivery capabilities of various configurations of the space-based STV concept are given by the curves in Figure 1-1.2.6-7. Also shown are the CNDB mission payloads and ΔV's.

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Flight Element Mass Summary

Flight Element	Inert Mass (kg)	Propellant Mass (kg)
Crew Module	5583	0
- with Protection	7383	0
Aerobrake	4110	0
Core Stage	8520	21450
Descent Tankset (1)	2415	28386
TLI Tankset (1)	4307	66727

Figure 1-1.2.6-6. Space-Based Concept Description

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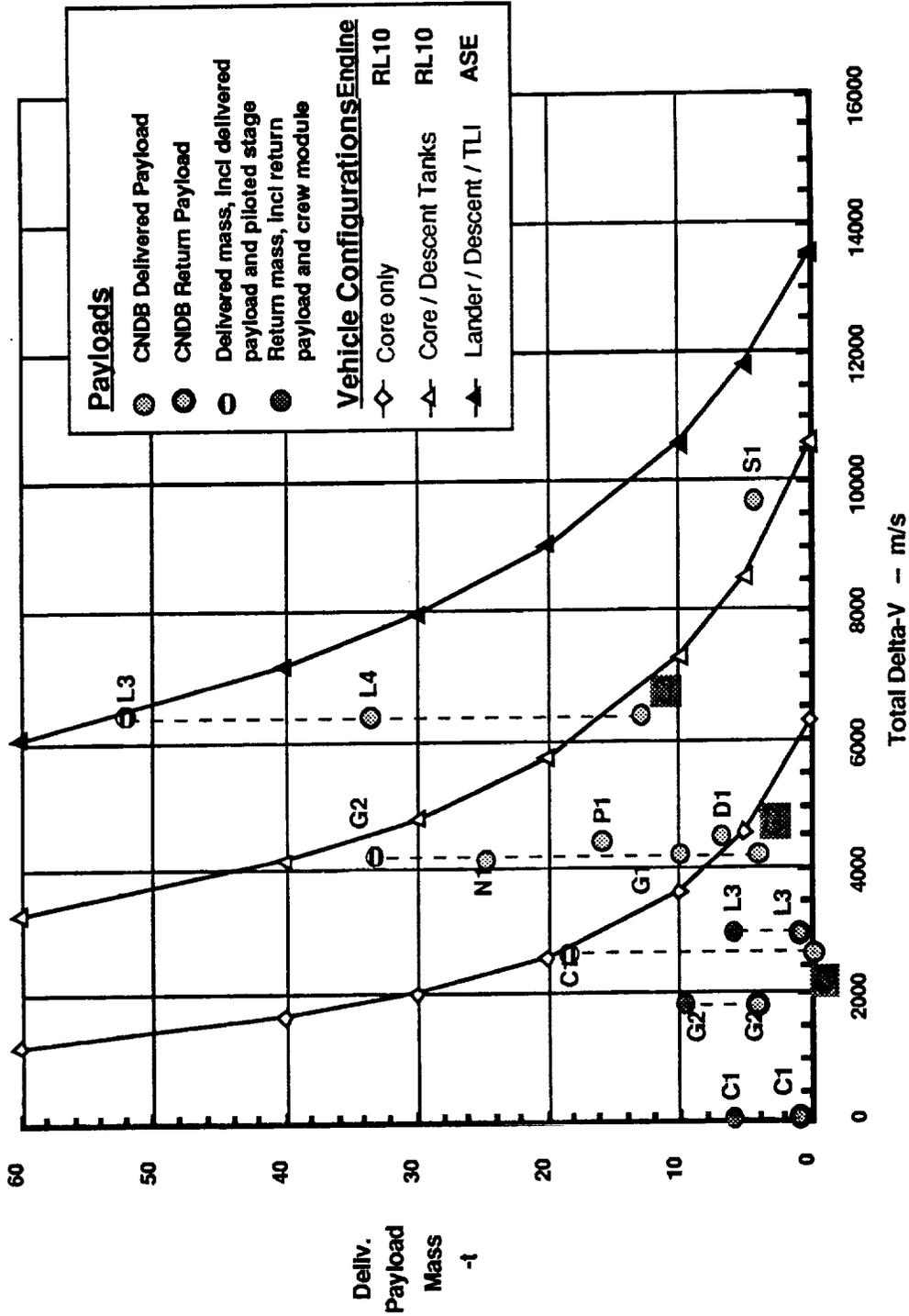
CNDB Missions are Captured by Components of a Lunar Mission-designed Vehicle.

Figure 1-1.2.6-7. Space-Based Vehicle Element Capabilities

1-1.3   TRAJECTORY ANALYSIS

The performance and trajectory analyses performed during the study can be broken down into four categories: (1) creation of an Earth-to-Moon trajectory database, (2) lunar mission survey (LMS) program development, (3) Boeing Lunar Trajectory (BOLT) multiphase trajectory program development, and (4) detailed analysis of specific performance and trajectory analysis issues. Copies of the database and program codes, and descriptions of these and the analysis results, were provided to MSFC as completed in 1989 and 1990 in working group and program review meetings. A summary of the four areas of work is given in the following paragraphs.

1-1.3.1   Earth-to-Moon Trajectory Database

Parametric data were generated summarizing 364 Earth-to-Moon trajectories with initial ascending (south to north) motion at TLI. The parameters varied are listed as follows.

Parameter	Range	Interval	(# of values)
Lunar true anomaly, deg	0-360	30	(13)
Transit time, hours	48-120	12	(7)
Translunar inclination, deg	0-60	20	(4)

Conditions assumed in generating the data are as follows:

Translunar perigee radius	6,478 km
Translunar orbit motion near Earth	posigrade
Approach hyperbola periapsis radius	2,000 km
Approach hyperbola motion	retrograde
Approach hyperbola targeting	minimum inclination

Figure 1-1.3.1-1 graphically illustrates the data. The trajectories were generated by integration including Moon and Earth perturbations. Results, which

- Base plane = Earth-Moon plane

- Ref axis = Earth-Moon line at time of closest approach to Moon

- Set of integrated trajectories from same SSF orbit altitude and to same altitude of closest approach to Moon (retrograde)

- Form of data:
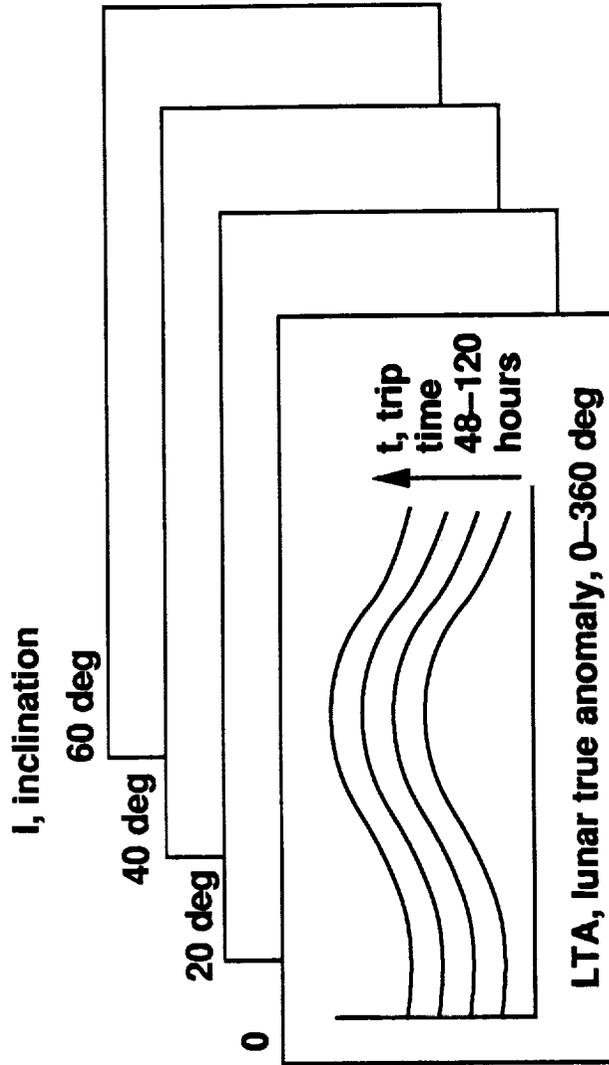
Figure 1-1.3.1-1. LMS - Parametric Database

completely define Earth-to-Moon (by symmetry, Moon-to-Earth) trajectories, are stored in four ASCII files of 91 trajectories each for automated lookup by programs such as the LMS program.

A plot of lunar approach energy, C3M, against lunar longitude of approach, SLONA, is included in Figure 1-1.3.1-2 to illustrate the contents of the database. The closed curves indicate the entire cycle of Moon positions each month. Transit time and inclination are shown parametrically.

1-1.3.2 Lunar Mission Survey Program

The LMS program is an analytical tool for the preliminary mission planning stage of lunar missions originating in, and returning to, low Earth orbit. It provides definition of timing and ΔV requirements for the impulses out of LEO, into and out of LLO, and the timing and orientation of the return approach to LEO.

Accessing a dataset consisting of integrated Earth-to-Moon coast trajectories parametric with respect to the Earth-Moon plane, LMS iteratively solves for the recurring geometry required between the regressing Space Station orbit plane and the lunar ephemeris. For each of a series of Space Station to Moon opportunities starting at a specified time, a series of return opportunities is found and data on the opportunities are provided.

Figure 1-1.3.2-1 illustrates results of the LMS program in terms of lunar orbit injection (LOI) ΔV for consecutive mission opportunities. The effect of orientation of the Space Station orbit plane is addressed. For each of six equally spaced orbit plane node locations, the three consecutive opportunities existing in January 1990 are shown connected by straight lines. Orbit plane orientation is seen to greatly affect time of month when the 3-day mission may be launched and causes LOI ΔV to 300-km orbit to vary from 870 to 1030 m/s. LMS has Fortran 77 source code and was initially hosted on microcomputers.
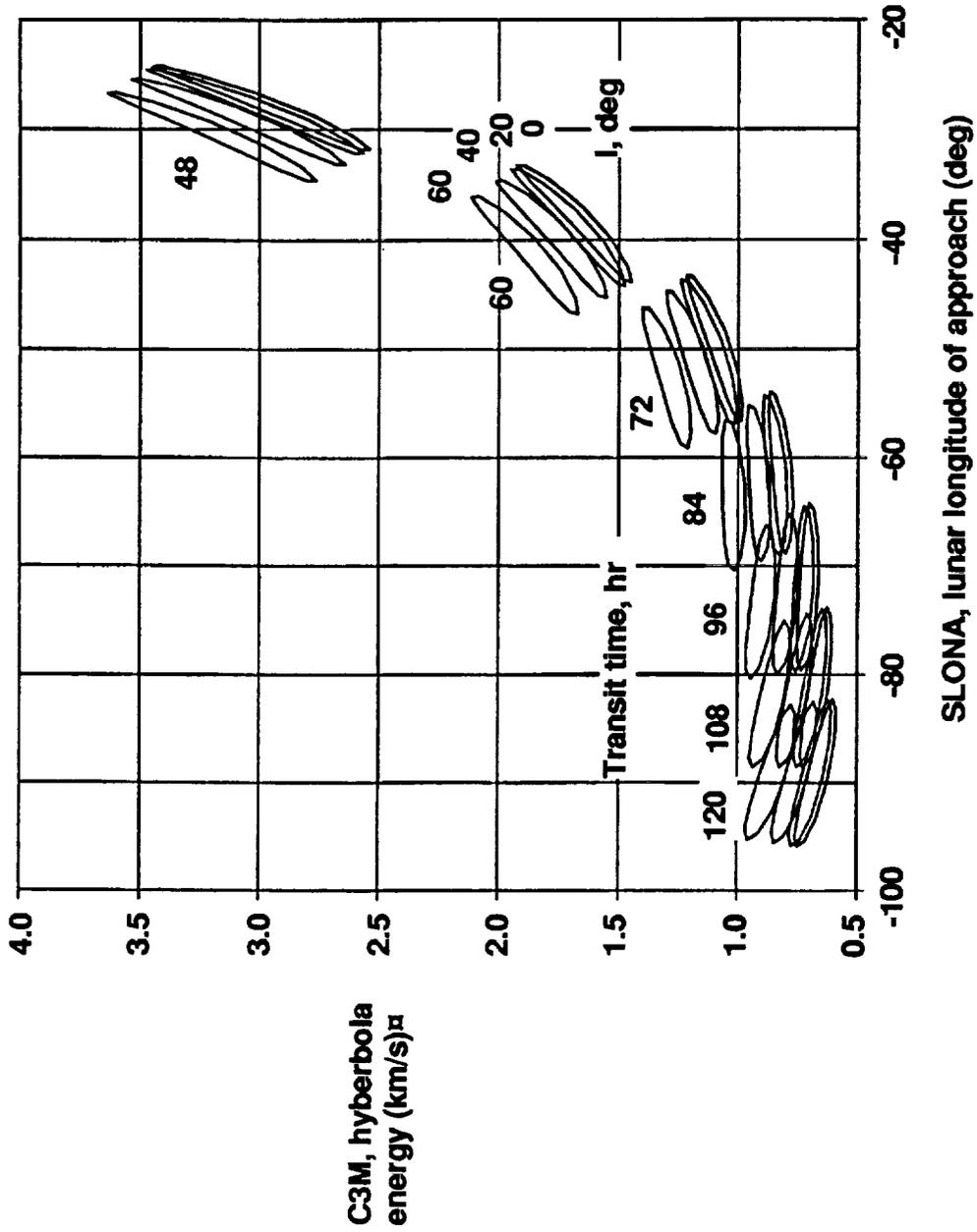
Figure 1-1.3.1-2.   LMS - Sample Parametric Data

LMS 315-1
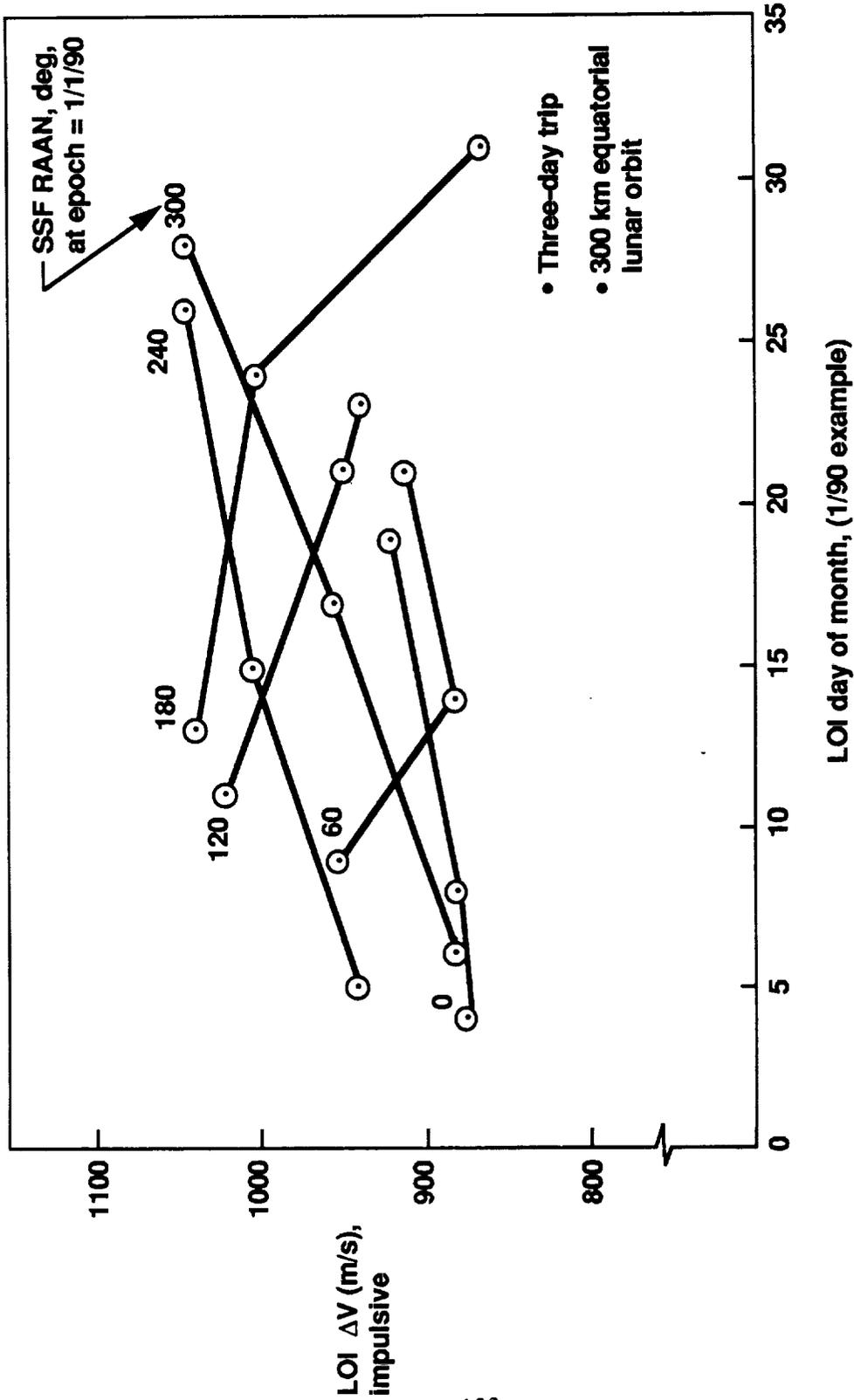
Figure 1-1.3.2-1.   SSF Orbit RAAN Effect on Mission Design

1-1.3.3   Boeing Lunar Trajectory Program

BOLT is a three-degree-of-freedom point mass trajectory simulation program used to rapidly analyze lunar missions. All phases of a mission, including launch, Earth orbit, thrusting, translunar and trans-Earth coast, lunar orbit, descent, stay time, ascent, and aerobrake can be included in the same trajectory. Analysis can be by explicit forwarding, search, optimization, or a combination of the three. An Encke scheme integrates about an updated two-body orbit for fast computation.

BOLT can be used effectively in conjunction with the LMS program. The LMS program, run in advance, supplies approximate times and ΔV values that can be input to the BOLT program for further analysis.

The mission analyzed can be as simple as an orbit about the Earth or Moon. It may also have many phases of coasts and burns mixed as desired and including trips between the Earth and Moon as well as orbits about the Earth or Moon. Flight to and from the Earth or Moon surface, and stay times on the surface, may be included. Flight through the atmosphere has drag and can have controllable lift. Multiple trajectories may be analyzed in the same case, separately initialized or branched from an earlier condition.

Vehicle modeling is by multiple stages, each with initial dry and propellant loads. Jettison or transfer of dry and/or propellant weight may be simulated at any time. Staging off the top and/or bottom may also occur at any time. Stage thrust is defined as a tabular function of time, and any stage may burn in any phase with arbitrary cutoff and restart capability.

Explicit trajectories may be run in which all control parameters are defined by input. Searches may also be performed, with control parameters automatically driven to satisfy en route and/or end conditions. Optimization is available to drive an explicit or search result to minimum or maximum. It is possible to combine explicit, search, and optimization analyses in the same case.

A series of explicit or search runs may be performed changing selected control variables parametrically. The modeling precision is controllable over a wide

range of the computational speed versus accuracy yardstick. This is made possible by the Encke computational method.

Inputs are in an easy to understand namelist style. Very flexible methods are provided for defining the problem. Result outputs are controlled by choice of output names and output interval, controllable in individual phases of the mission analyzed. Different parts of a mission may be output to different files. Subsequent plotting of results is easily supported by the block header and block data output versus time. Informative screen displays report on the program's progress while executing and can include selected excepts from the computed results.

The code is portable, having been developed in standard Fortran 77 programming language. The program was initially hosted on microcomputers.

Integration Method. The Encke integrating technique used in BOLT is illustrated in Figure 1-1.3.3-1. Integration is about an updated two-body orbit, which makes possible wide variation in choice of step size. Step size may vary from multiple orbit revolutions with a spherical central body for computational speed, down to several seconds for accuracy in a thrusting and/or dragging flight.

The two-body orbit used for integration is centered about the Earth for flight in the vicinity of the Earth, and this switches to the Moon's center when near the Moon. This takes advantage of the dominance of the gravitationally nearer body, treating the other as a perturbation. The sun's perturbation may be included if desired.

Sample Results. BOLT was used to simulate a lunar mission from translunar injection ignition, through burn into lunar orbit, stay in lunar orbit, return to Earth, aerobrake, and match with Space Station orbit. Figure 1-1.3.3-2 shows the Earth-to-Moon and Moon-to-Earth legs by means of a view from the celestial north pole. Figure 1-1.3.3-3 shows the near-Earth portions of the flight, with long straight lines directed to the Moon for the outbound leg and from the Moon for the return.

Delta V added impulsively

Two-body forwarding first half of interval

Two-body forwarding second half of interval

Final conditions

Initial conditions
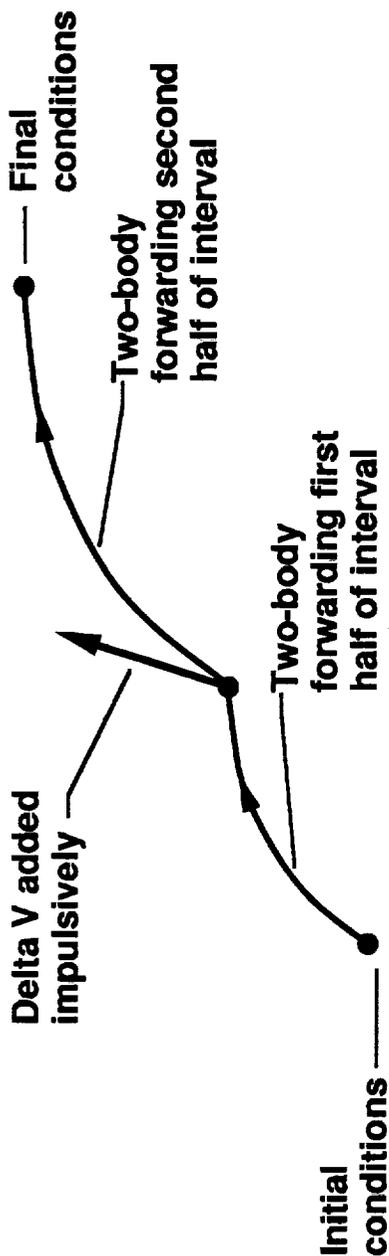
- Program integrates about reference two-body orbit
  - Goodyear method of forwarding
  - Origin at Earth or Moon

- Mid-step delta V represents effect over entire interval
  - Thrust
  - Drag and lift
  - Oblateness
  - Perturbing bodies

- Step size selectable

Figure 1-1.3.3-1. BOLT- Integration Step in Lunar Trajectory Program

Figure 1-1.3.3-2. BOLT - Sample Output Plotted
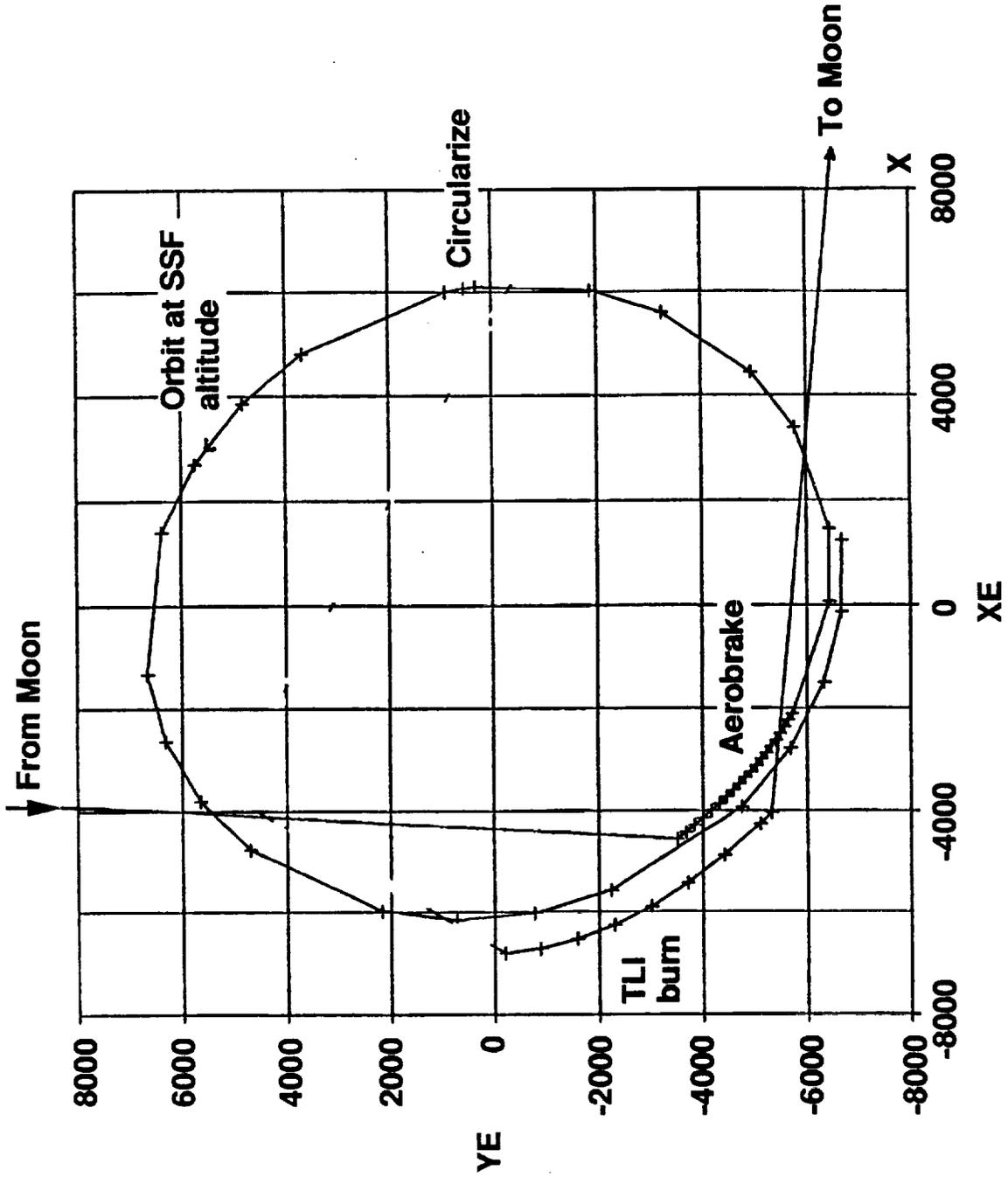
Figure 1-1.3.3-3. BOLT - Sample Output Plotted (Near Earth)

1-1.3.4   Performance and Trajectory Analysis

Analyses were performed as needed during the performance period of the STV study. The following sections address a few of the problems analyzed to support the generation of requirements, vehicle design, and interface trades.

1-1.3.4.1 Effect of Earth Orbit Departure Delays

The everyday flexibility of surface launches to the Moon is not available from Space Station orbit. Instead, mission opportunities average about one every 9 days. The question arises, having selected a nominal departure time, how long may departure be delayed, even to the next orbit revolution, or later? This problem was analyzed from the standpoint of TLI burns having the same geometry relative to the flight path (tangential) but with differing ignition times. That is, there is no change in burn pointing and ΔV in response to the delay. Rather, a correction is performed 1 to 2 days later, combined with the normally scheduled midcourse correction burn.

Translunar Injection Burn. Figure 1-1.3.4.1-1 has been included to illustrate the basic geometry of the lunar approach path. While the Earth-to-Moon trajectory may be elliptical relative to Earth at first, as it approaches the Moon the path becomes a Moon-centered hyperbola. The figure shows the path projected through the point of closest approach and the hyperbola's approach asymptote.

Forwarding the translunar trajectory to define lunar approach conditions involves definition of the lunar approach hyperbola relative to the aim point or target plane. (The target plane is a plane containing the Moon's center and is perpendicular to the selenocentric hyperbola's incoming asymptote. Orientation of the target plane is nearly fixed for all lunar approaches from a 3-day translunar trajectory.) The pierce point, where the asymptote intersects the target plane, defines the inclination and periapsis altitude of the flyby path. Slight, early changes to the translunar path can be thought of as moving the asymptote but not changing its direction. Figure 1-1.3.4.1-2 shows the effect on the asymptote as TLI ignition time varies 0.02 revolutions (about 2 minutes) about a favorable time in each of six different Space Station orbit passes. In this
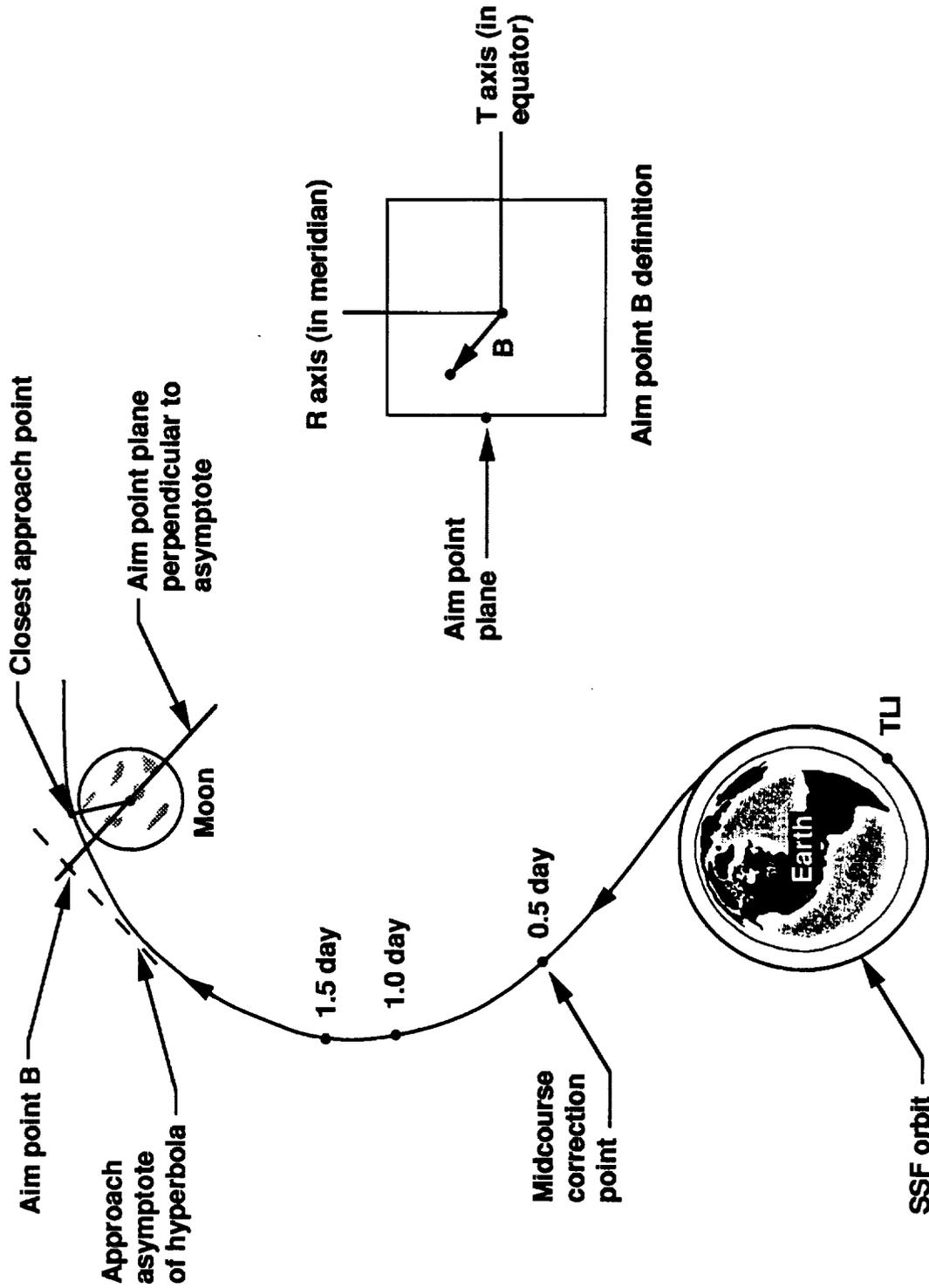
Figure 1-1.3.4.1-1.   Lunar Approach Targeting

Figure 1-1.3.4.1-2. Correction Required for TLI in Off-Nominal SSF Revolution
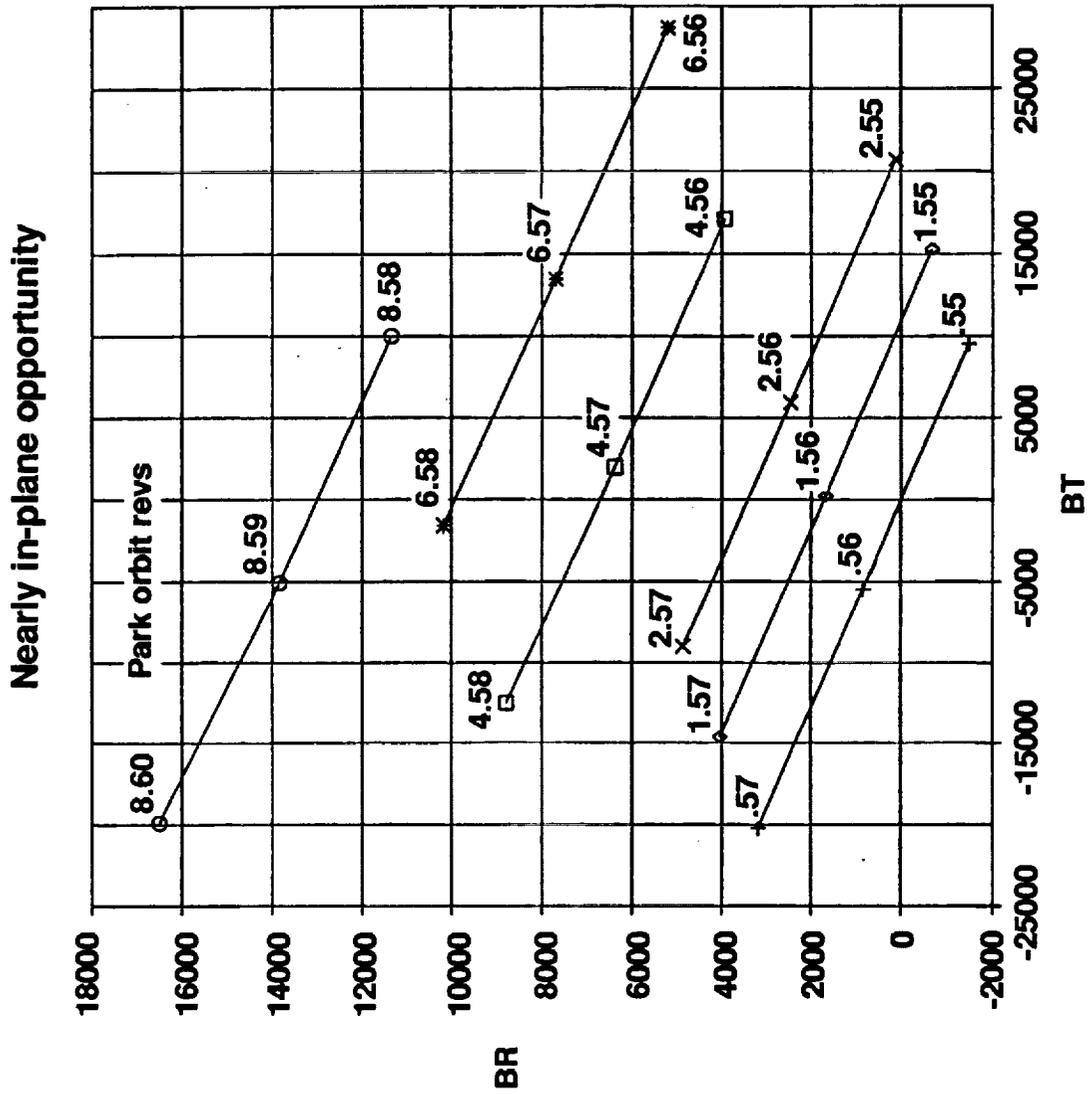
figure, the pierce point locations are referenced to rectangular axes with origin at the Moon's center, so a vertical impact trajectory has a pierce point at 0,0. The T-axis (BT in the figure) lies in the equator (to the east of the vertical impact point) and the R-axis (BR in the figure) is defined by the north pole. The desired pierce point for injecting into the 300-km posigrade orbit is at T,R coordinates at -5400, 800.

A TLI burn timed to ignite 0.56 revolutions after the ascending node of the nominal Space Station orbital pass gives the desired pierce point. TLI burns that ignite 0.01 revolutions (about 1 minute) earlier and later than this move the pierce point about 15,000 km to the 0.55 and 0.57 locations. A line is shown connecting the 0.55, 0.56, and 0.57 points in the figure to emphasize the continuous opportunities.

Delaying to the next orbit revolution moves the line of pierce points about 1,700 km in the R-axis to the position shown by 1.55, 1.56, and 1.57, and a similar pattern is also seen for orbit revolutions 2, 4, 6, and 8. The close spacing of these lines is due to the nearly in-plane translunar trajectory; wider spacing would occur for many opportunities.

In summary, there can be several successive orbital revolutions having a window of about 2 minutes from which TLI burns can place the lunar approach asymptote within 15,000 km of the desired aim point. (This is without considering correction of TLI pointing, and increased ΔV, in response to the delays.) Midcourse correction of the pierce point is discussed in the following section.

Midcourse Correction. The lunar approach hyperbola's asymptote may be moved to a new position in the T-R target plane by a midcourse correction performed after TLI. The effectiveness of the midcourse burn is shown in Figure 1-1.3.4.1-3, where midcourse changes in a 3-day trajectory nominally going to T,R coordinates (-5400, 800) are depicted. Midcourse correction times ranging from 0.5 to 2.0 days after TLI were analyzed. Direction of the correction impulse was normal to the flight path (four points connected by closed curve) and tangential, both posigrade and retrograde. A constant midcourse correction ΔV of 10.8 m/s was assumed.
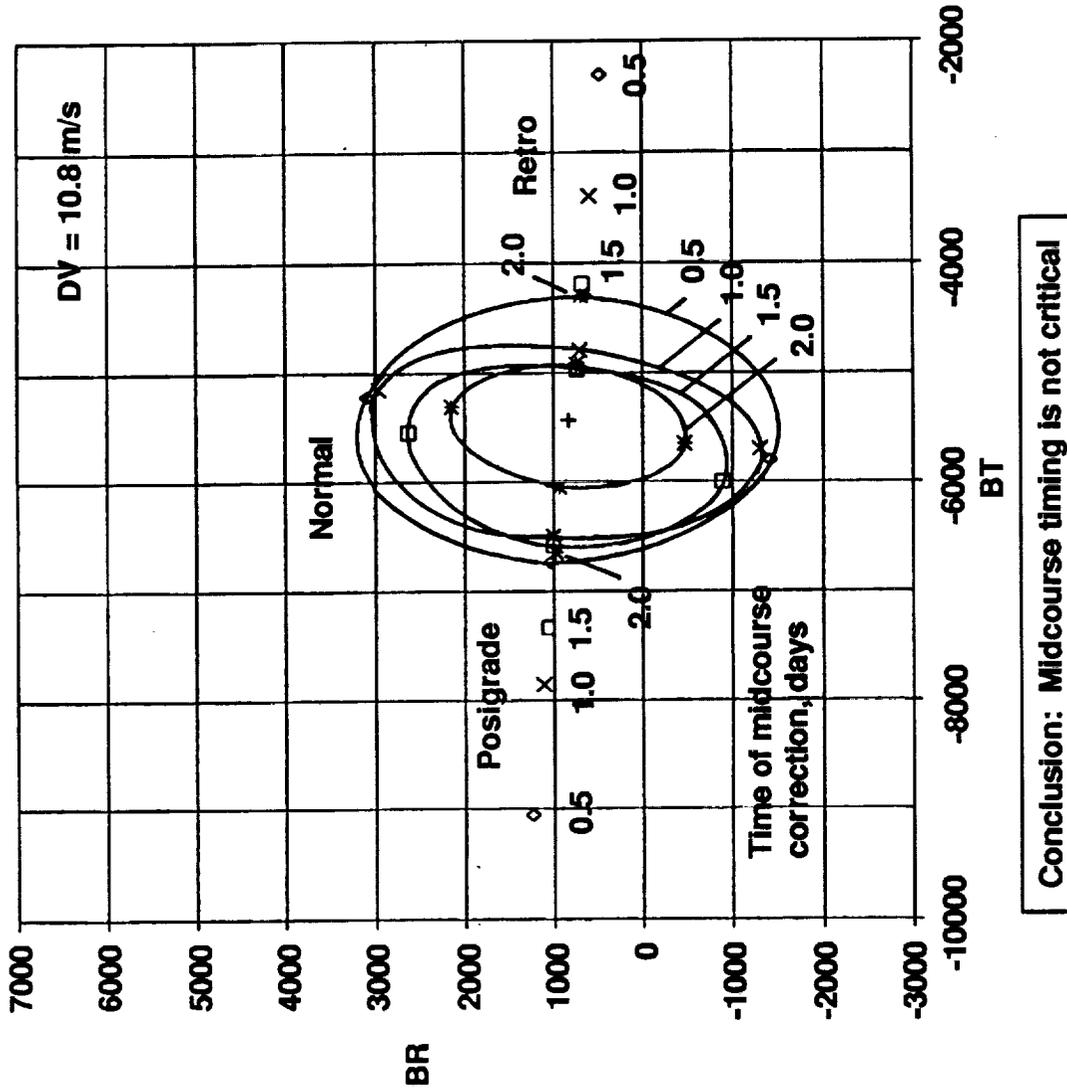
Figure 1-1.3.4.1-3.   Midcourse Retargeting Capability

Tangential corrections are effective, especially for changing the T component early in the flight. The effectiveness of normal corrections is more nearly constant. Figure 1-1.3.4.1-3 shows that the asymptote can be moved in the R direction about 100 km per m/s of correction ΔV up to 2 days after TLI or in the T direction about 45 km per m/s by a normally directed correction. The 1,700-km change in R, due to a one orbit delay discussed above, thus requires about 17 m/s to correct. Midcourse correction of a TLI timing error costs about 170 m/s per minute of error.

1-1.3.4.2 Opportunities From and to Space Station

Figure 1-1.3.4.2-1 is included to show the variation of the Moon's angular distance out of the Space Station orbit plane (dashed line). The time between zero crossings is noted by the values from 6 to 11 days.

If the Space Station orbit plane had a fixed orientation, the crossings would average every 13.6 days. But because of the Earth's oblateness, regression of the orbit plane (about 7 degrees per day opposite to the Moon's motion) reduces the spacing to 9 days on the average.

The angle between the orbit planes of the Moon and Space Station is shown as a solid line in the figure. High relative inclination is 47 to 57 degrees and low relative inclination is 0 to 10 degrees. Actual extremes depend on the Moon's orbit inclination, which varies from 18.5 to 28.5 degrees in a 19-year cycle.

Missions to the Moon must be based on the zero crossings, as can be seen in Figure 1-1.3.4.2-2. A descending zero crossing (from the north to the south side of the Space Station orbit plane) is illustrated, such as the circled point following the "9" in Figure 1-1.3.4.2-1 . Also shown in Figure 1-1.3.4.2-2 is a single arrowhead indicating one possible free return path. Other return paths are also possible by slight adjustment soon after leaving the vicinity of the Moon. All the possible return planes contain the Earth-Moon line but can have any inclination relative to the Earth-Moon plane.

Figure 1-1.3.4.2-3 shows that a free return path does not, in general, arrive at Earth in the plane of the Space Station. This is because of the regression of the

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Figure 1-1.3.4.2-1.   Lunar Launch Windows From SSF

- - - - Moon declination from Space Station Orbit

—— Angle between lunar & Space Stalon Orbit planes

⊙ Lunar mission launch opportunity from SSF

Out-of-plane angles (deg)

Time In days

BOEING

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D180-32040-2
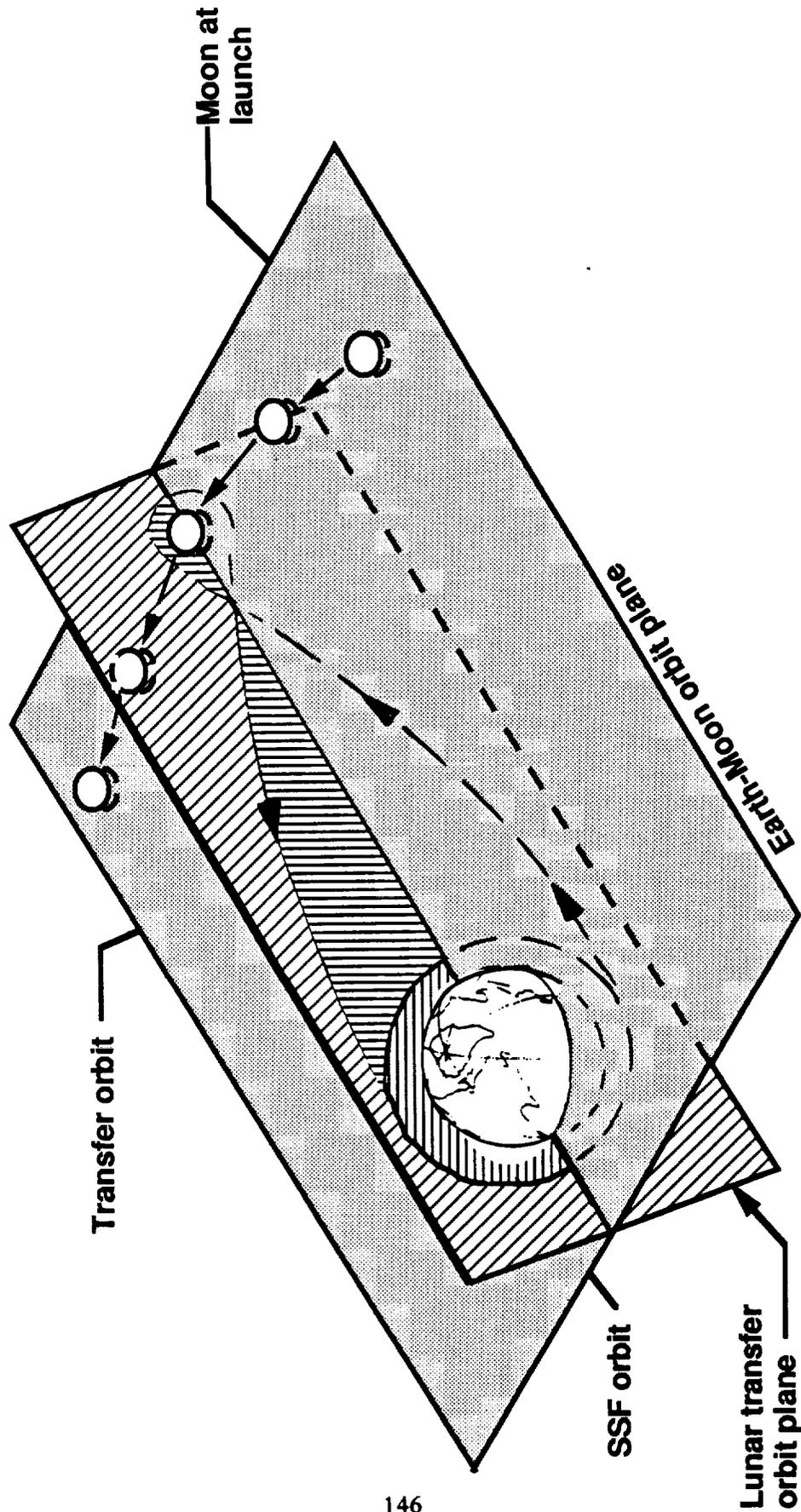
Figure 1-1.3.4.2-2. Lunar Transfer Geometry From SSF

BOEING

- Approximately 7°/day regression rate of SSF orbit WRT transfer plane
- After 6 day round trip "Fly by" SSF orbit is misaligned about 40°

Moon at launch

Earth-Moon orbit plane

Transfer orbit

SSF orbit at TLI

Lunar transfer orbit plane

SSF orbit at end of "Free Return" (~40° out of transfer plane)
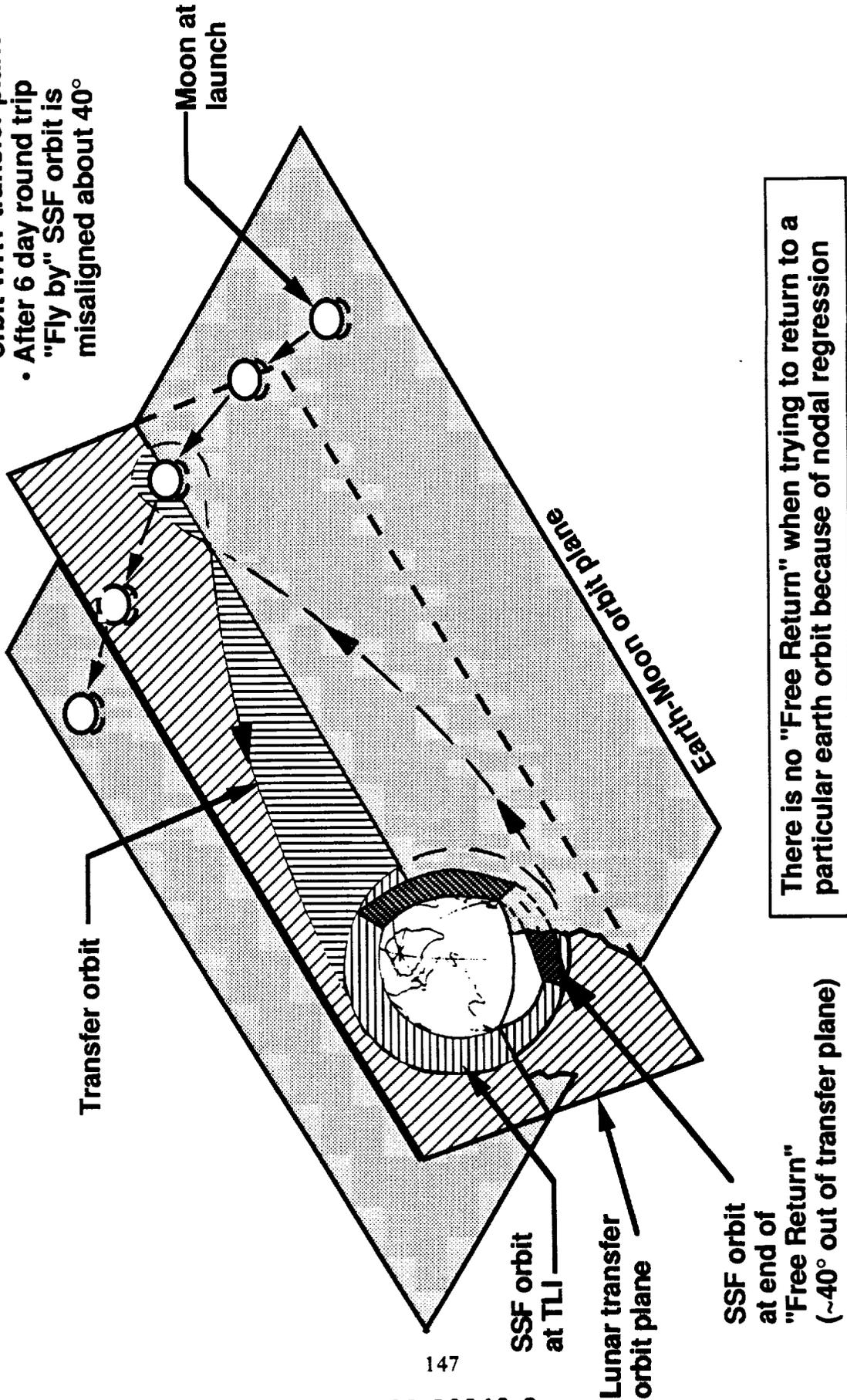
There is no "Free Return" when trying to return to a particular earth orbit because of nodal regression

Figure 1-1.3.4.2-3. Impact of SSF on Free Return

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D180-32040-2

orbit plane during the 6 days since departing from the Space Station. The heavy shading in the figure shows the final Space Station orbit plane.

Special situations can be found supporting free and near-free return to the Space Station, and these depend on near-zero angle between the orbit planes of the Moon and the Space Station. Such situations occur every 19 years, when the Moon's orbit inclination is near its maximum of 28.5 degrees.

1-1.3.4.3 Mission Abort Analysis

In the event of a need for earliest possible return to the Space Station, the overriding problem is the potentially large (up to 57-degree) angle of the Moon out of the plane of the Space Station's orbit. (Nominal mission event times are based on the passages of the Moon through this plane, and the opportunities average about 9 days apart.) Figure 1-1.3.4.3-1 reflects this worst case condition in the three upper solid "ΔV required" lines. Available ΔV is shown as dashed lines, decreasing in three phases with the nominal burn expenditures. Even a so called "free return" from translunar trajectory cannot avoid the problem of high ΔV because the Moon is, in general, out of the plane at the time of flyby. The point "B" chosen for the plane change maneuver is a location minimizing ΔV. Any approach azimuth at "A" is available through midcourse correction. Note that the data presented in Figure 1-1.3.4.3-1 were generated for the 90-day study reference vehicle (2.5 stage, LEV/LTV scenario, using LOR) and is presented here to provide visibility into concerns that must be addressed.

One way around the problem early in the mission, post-TLI burn, is shown as the "immediate" return. Here, a downward ΔV reverses the radial rate. The Space Station orbit thus has less time to regress, though the increasing plane change requirement is seen in the upturn of this line. A nominal mission can be planned that reduces the ΔV requirement for immediate and later aborts by launching when the SSF/lunar alignment favors the in-plane geometry.

Note that these free return issues are only applicable to STV concepts that use a LEO node. Both ground-based options (GB-1.5S and GO-1.5S) have a free return capability because the ballistic reentry crew module can return the crew to Earth.
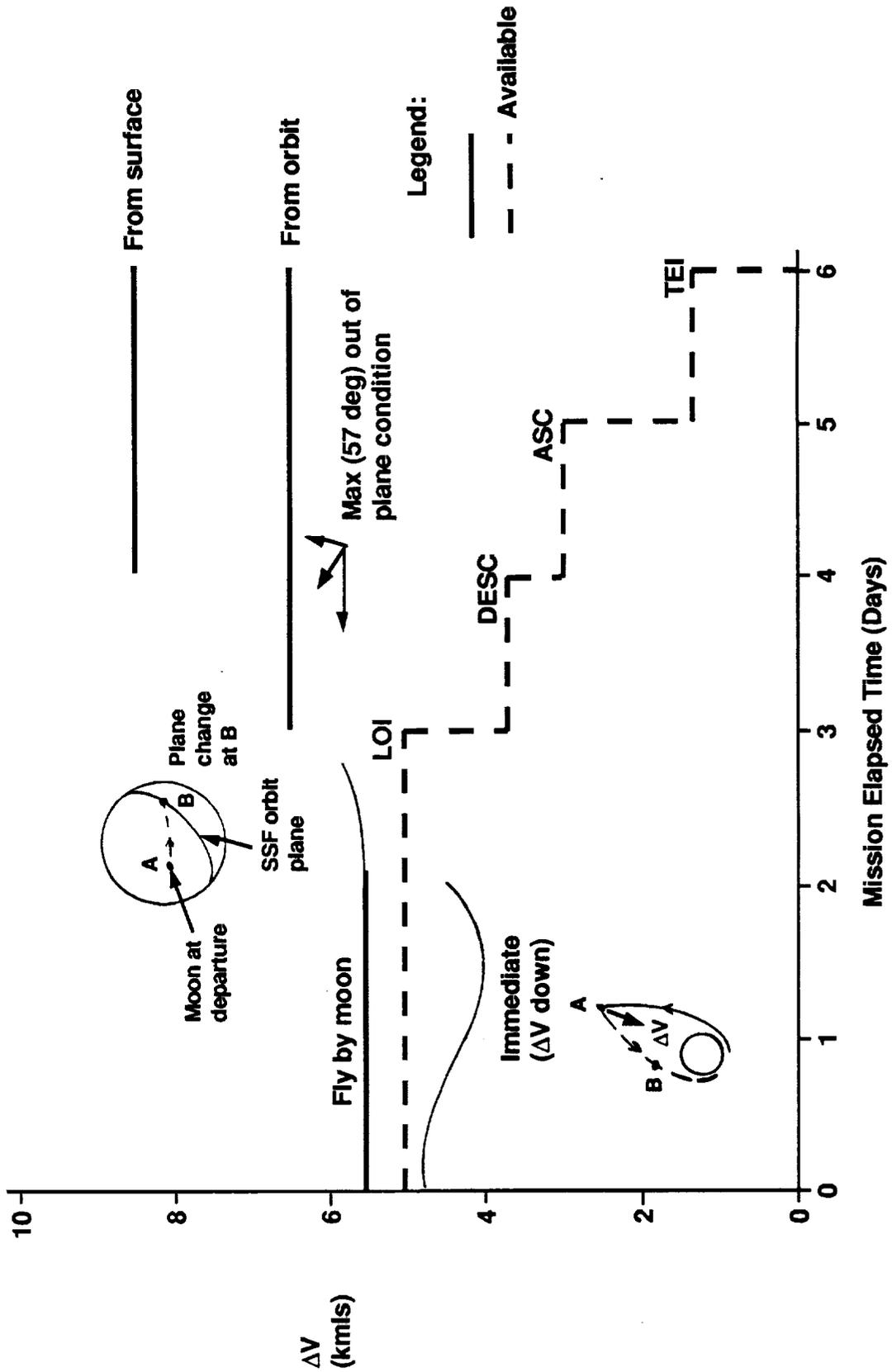
Figure 1-1.3.4.3-1. Abort Scenario Capability

Options for accommodating aborts with a LEO-based concept depend on the mission phase and situation requiring the abort. Options include inclusion of the necessary ΔV capability (large performance penalty); the use of a rescue vehicle to retrieve the crew from a LEO orbit (non-aligned with the LEO node) obtained after an abort return; or waiting until the LEO node orbit is in the necessary alignment either through (1) use of a LEO parking orbit to wait until the parking orbit and the LEO node orbit are aligned, (2) waiting in an LLO orbit, or (3) waiting on the lunar surface (either of these may require a long wait time that may be undesirable in emergency situations).

The operational scenarios that have a node to rendezvous with in low lunar orbit (LOR approach) were not selected. However, if the LOR approach is ultimately chosen, there are times when, depending on landing site latitude and lunar node orbit inclination, additional ΔV must be available for immediate return or else safe haven must be available while waiting on the surface for proper alignments. However, the non-optimum lunar orbit operations do not have a severe performance penalty associated with them as do the Earth orbiting node non-optimum operations.

As the Moon rotates on its axis and revolves around the Earth, the lunar orbiting segment will remain in a fixed inertial attitude. The orbit will not pass over the landing site and in fact can be some distance away depending on the site selection and node orbit. This is depicted in Figure 1-1.3.4.3-2 where the landing base and orbit are shown in their worst misalignment. To rendezvous with the orbiting element an LEV would have to ascend to orbit and then make a plane change to match orbits. The worst case ΔV to do this plane change is shown for a 10-degree orbit inclination and 10-degree landing site.

Aborts from the vicinity of the Moon are pictured in Figures 1-1.3.4.3-3 through 1-1.3.4.3-5 to address the three landing methods studied: one-burn direct, lunar orbit direct, and lunar orbit rendezvous.

For the direct landing, targeting for the landing site is nominally accomplished approximately 1 day prior to lunar arrival, and the STV is then on an impact trajectory. Using the descent and ascent fuel, an abort from the impact trajectory
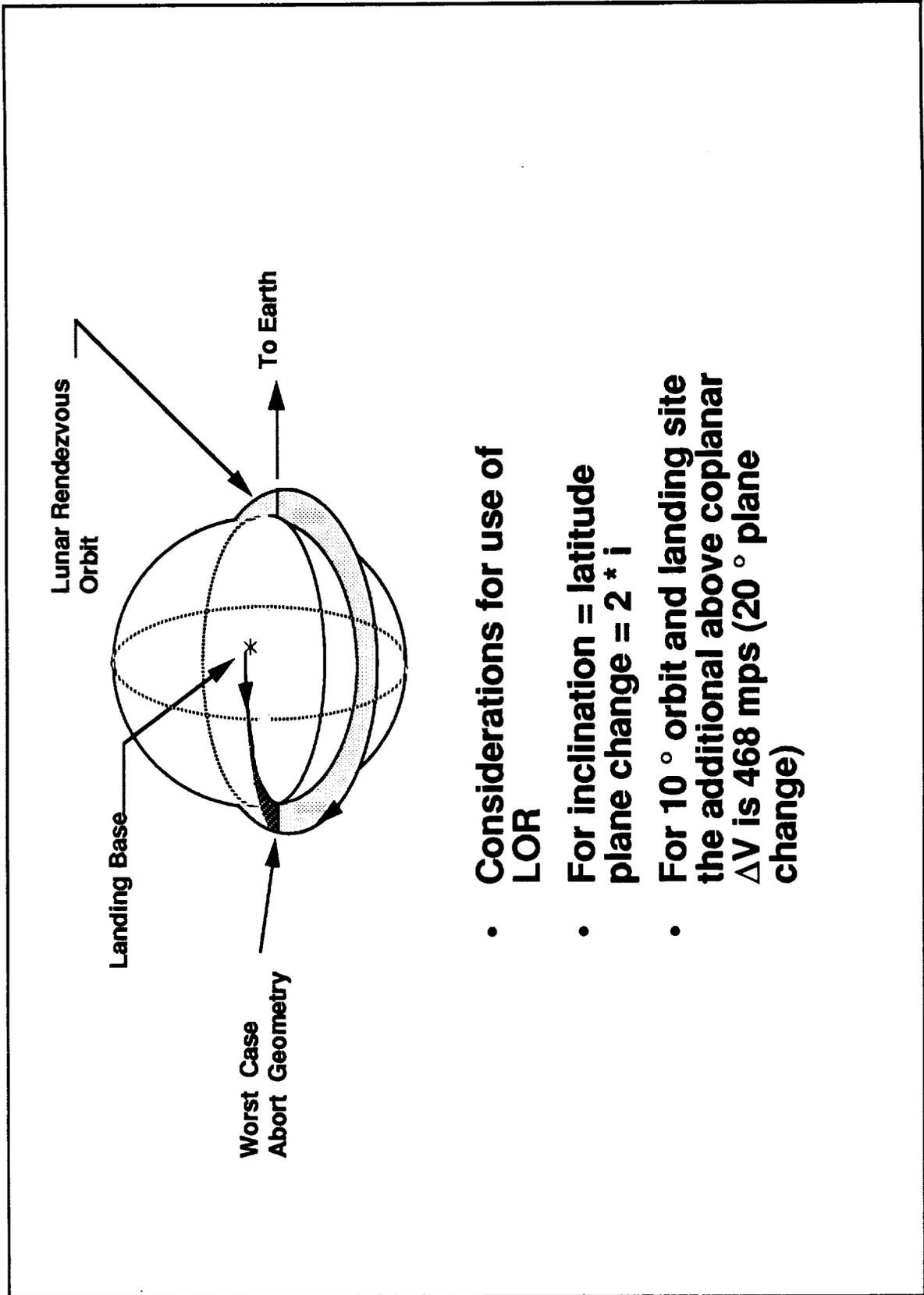
- **Considerations for use of LOR**

- **For inclination = latitude plane change = 2 * i**

- **For 10 ° orbit and landing site the additional above coplanar ΔV is 468 mps (20 ° plane change)**

Figure 1-1.3.4.3-2. Abort From Lunar Surface Considerations
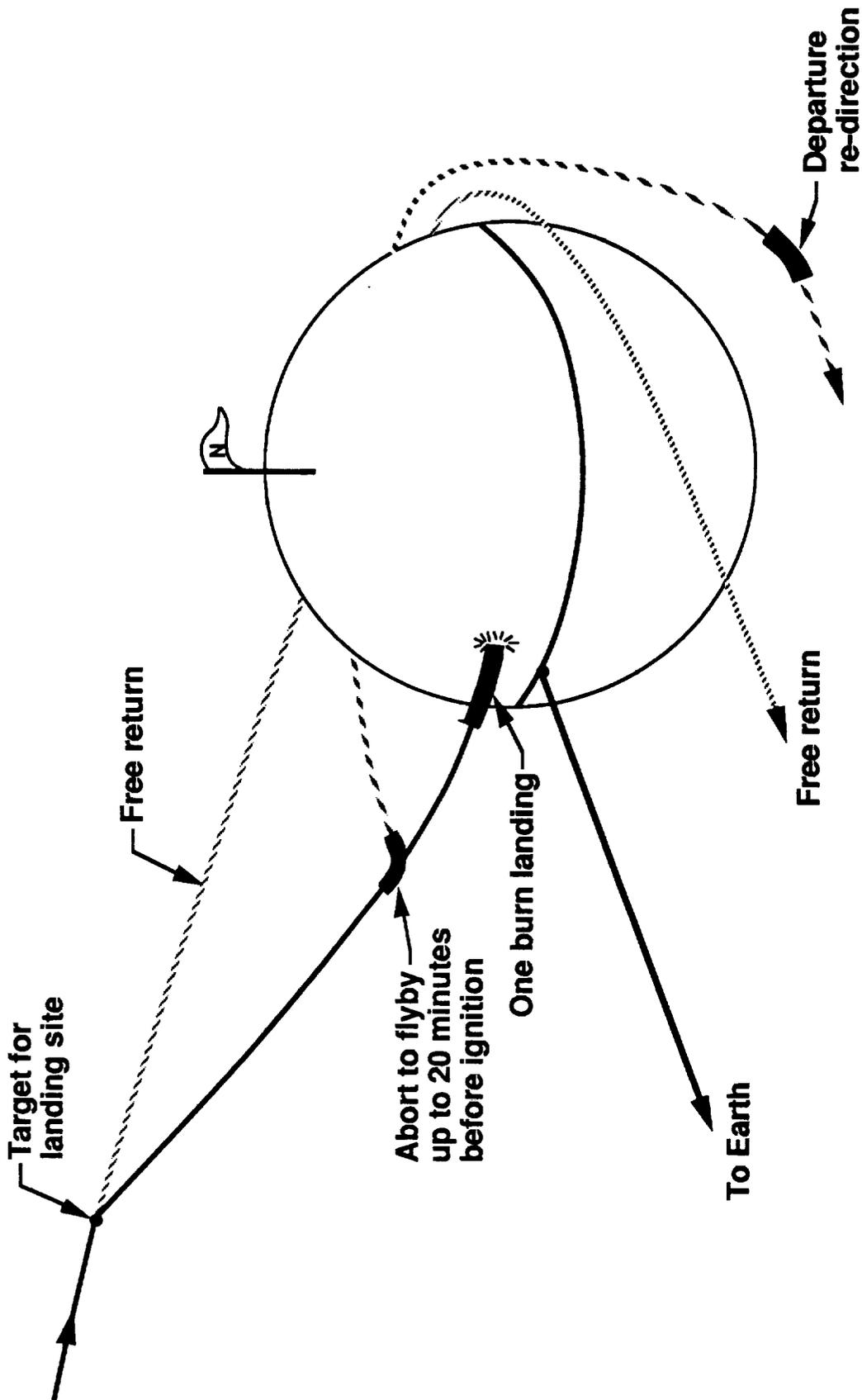
Figure 1-1.3.4.3-3. One-Burn Direct Aborts
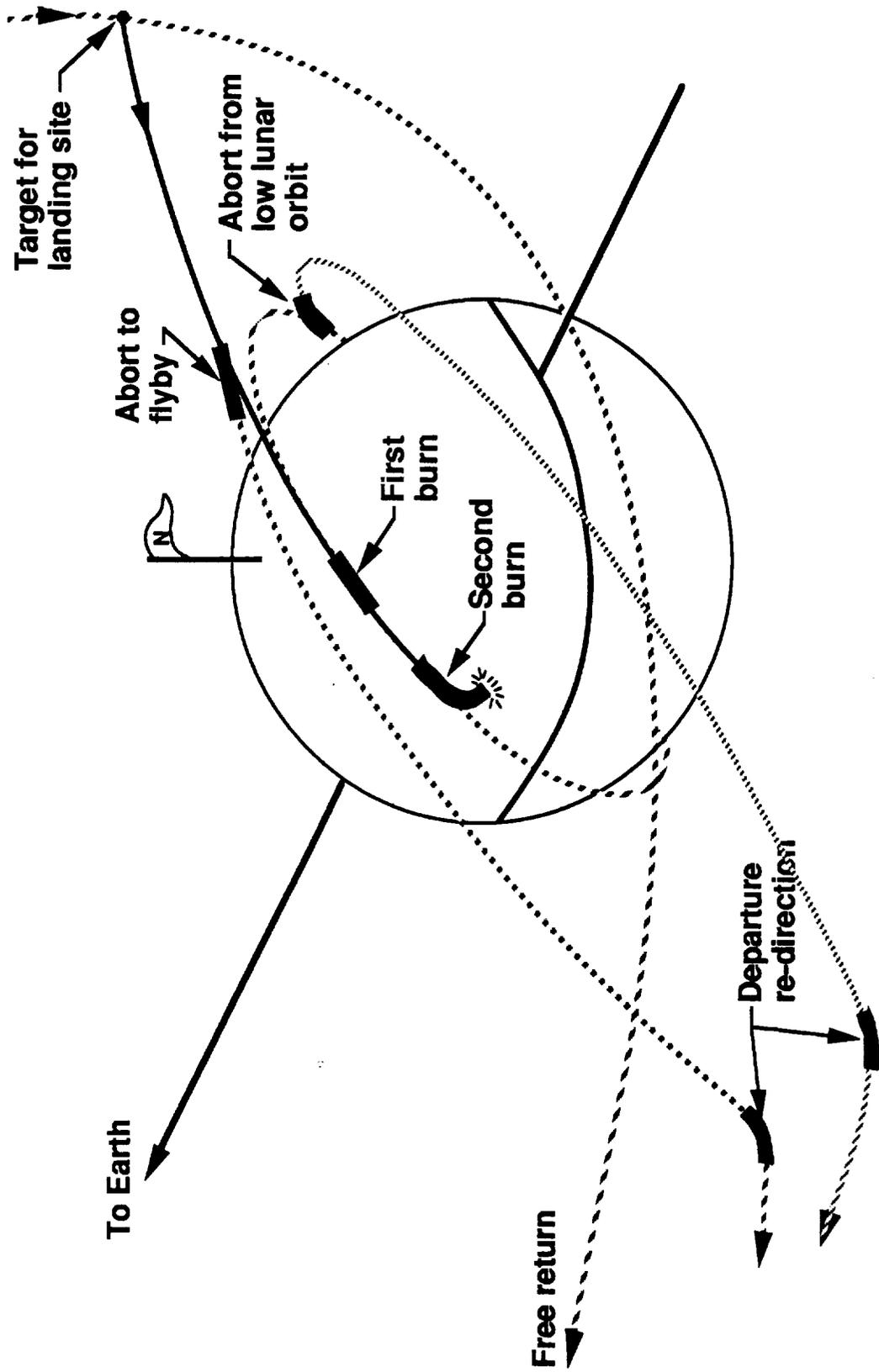
Figure 1-1.3.4.3-4. Lunar Orbit Direct Aborts
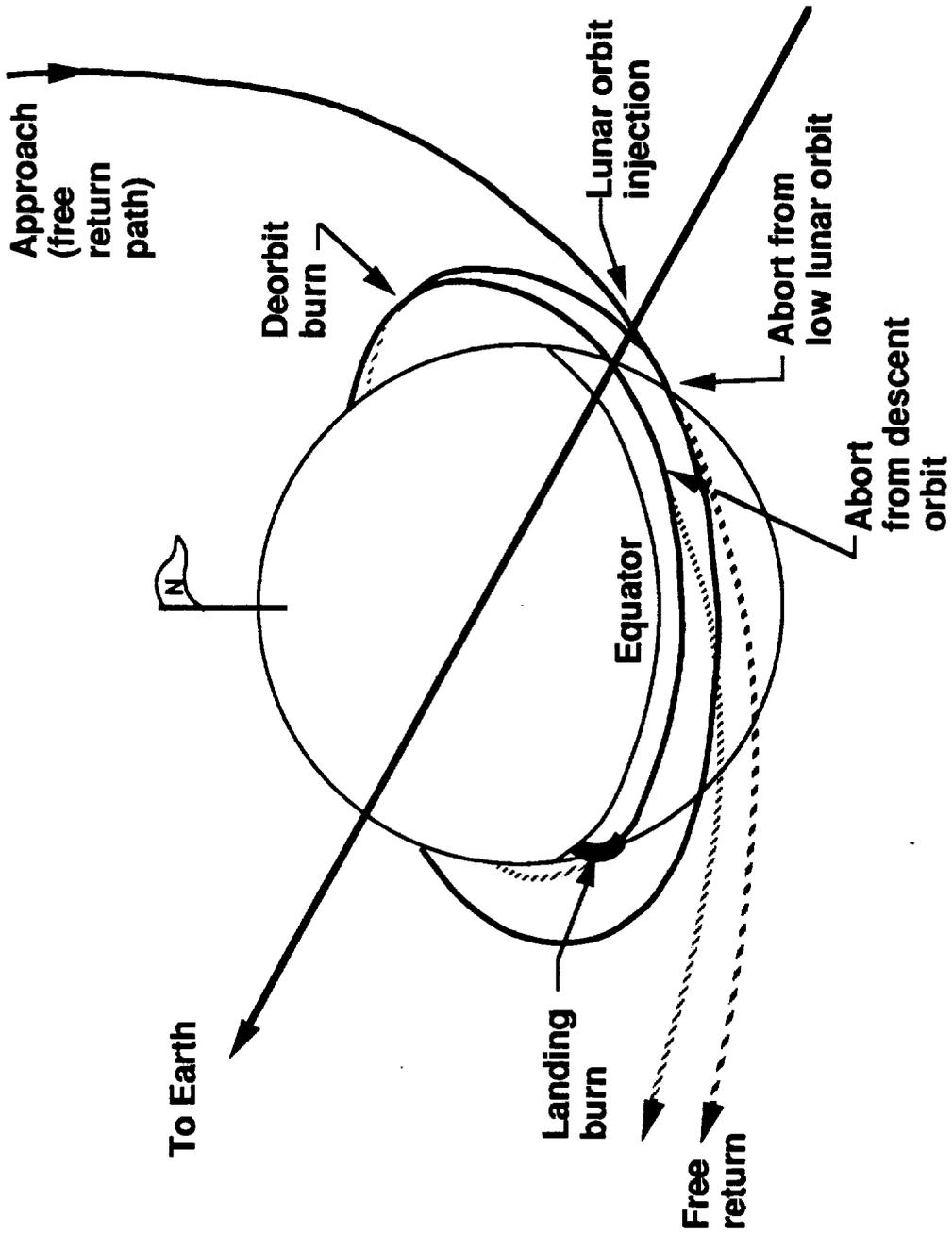
Figure 1-1.3.4.3-5. Lunar Orbit Rendezvous Aborts

to an Earth return can be accomplished up to 20 minutes prior to the scheduled final descent and landing burn (Figure 1-1.3.4.3-3).

Aborts for the lunar orbit direct approach can be accomplished from two coast phases: after retargeting from the initial free-return trajectory to the intermediate lunar orbit and after the first burn while in descent orbit (Figure 1-1.3.4.3-4).

Aborts for the lunar orbit rendezvous approach can be accomplished from two coast phases: from LLO after the lunar orbit injection burn and after the deorbit burn while in descent orbit. Note that an abort from the approach path is automatic because of the free-return targeting (Figure 1-1.3.4.3-5).

1-1.3.4.4 Lunar Orbit Stability

Missions featuring a lengthy stay in LLO require some degree of long-term orbit prediction. Accurate prediction prolongs the time between required orbit determination updates and gives more look-ahead time for making mission decisions. Conversely, inaccurate prediction requires a higher level of operational support in terms of tracking and orbit determination and can cause a mission problem to deteriorate more rapidly into an emergency. Also, more accurate knowledge of orbit stability can permit use of lower altitude orbits, enhancing payload performance.

Inherent in long-term orbit prediction is a modeling of the lunar gravitational model, typically in the form of a series involving latitude- and longitude-dependent harmonics.

Figure 1-1.3.4.4-1 presents five simple gravity models (four to eight coefficients) developed on the basis of one lunar orbiter and three early Apollo missions. Two of the coefficients, J20 and C22, are well known. Figure 1-1.3.4.4-2 shows the nodal regression - J20 relationship. Two others, J30 and C31, are fairly well known as shown in Figure 1-1.3.4.4-3 for C31. Models involving additional coefficients differ according to the orbit orientations used. None of the models are satisfactory for prediction of all orbit elements, as Figure 1-1.3.4.4-4 shows for three models predicting perilune attitude for a lunar orbiter. It is likely that the

	R-2	R-3	R-4	R-5	B-1	TGR-3
J_{20}	2.07108	2.1	2.1	2.1	2.07108	2.07108
C_{21}	—	—	—	.5	—	.417
S_{21}	—	—	—	-.43	—	-.45
C_{22}	.20716	.21	.21	.21	.20716	.20716
S_{22}	—	—	—	-1.465	—	—
J_{30}	-.21	-.16	-.10	-.16	-.21	-.21
C_{31}	.34	.42	.42	.42	.34	.34
S_{31}	—	—	—	—	—	—
C_{32}	—	—	.05	—	—	—
S_{32}	—	—	.05	—	—	—
C_{33}	—	—	—	—	.011	.02583
S_{33}	—	—	—	—	.035	—
J_{40}	—	—	—	—	—	-.6

Figure 1-1.3.4.4-1.   Gravity Models Evaluated for Apollo
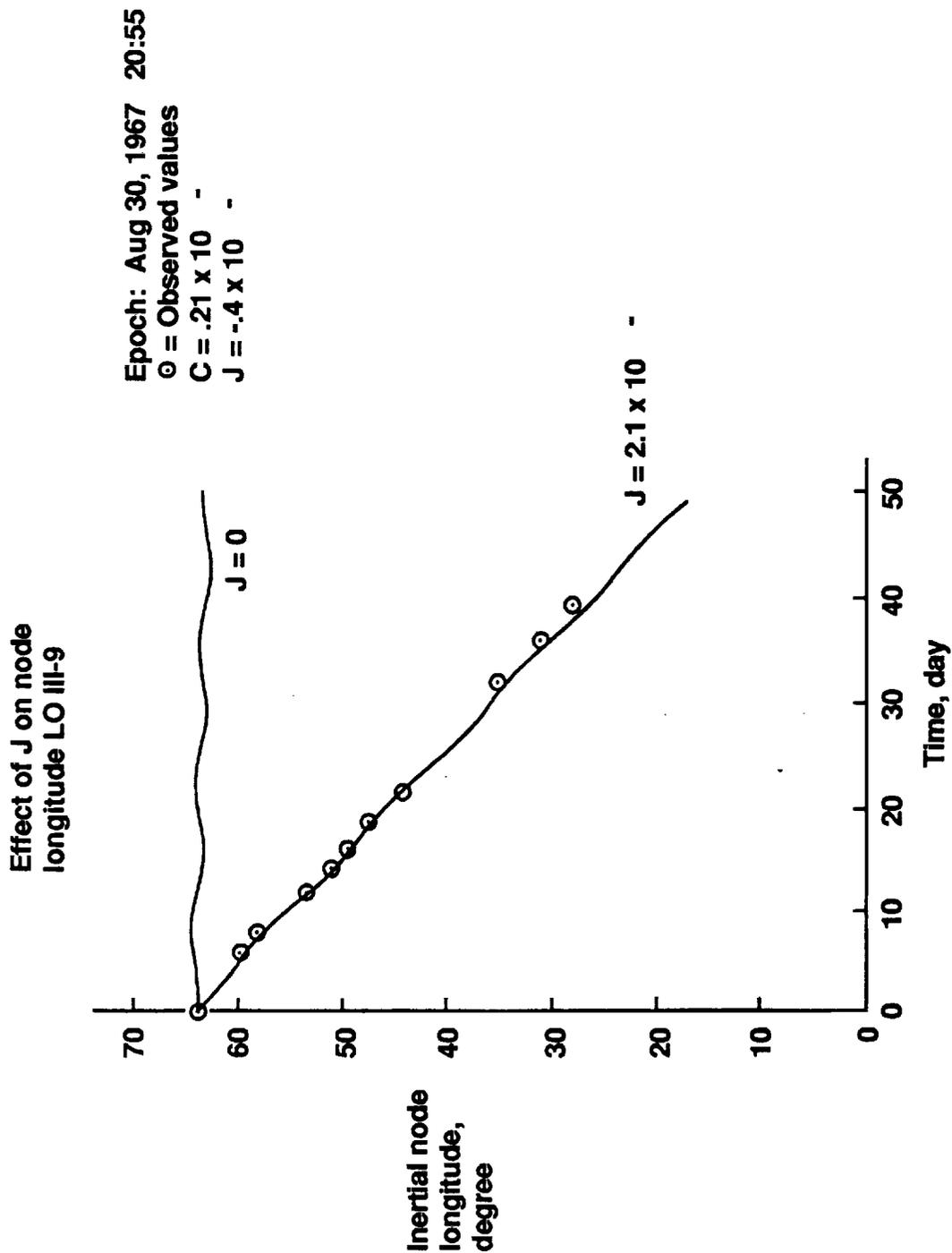
Figure 1-1.3.4.4-2. Simply Modeled Variations
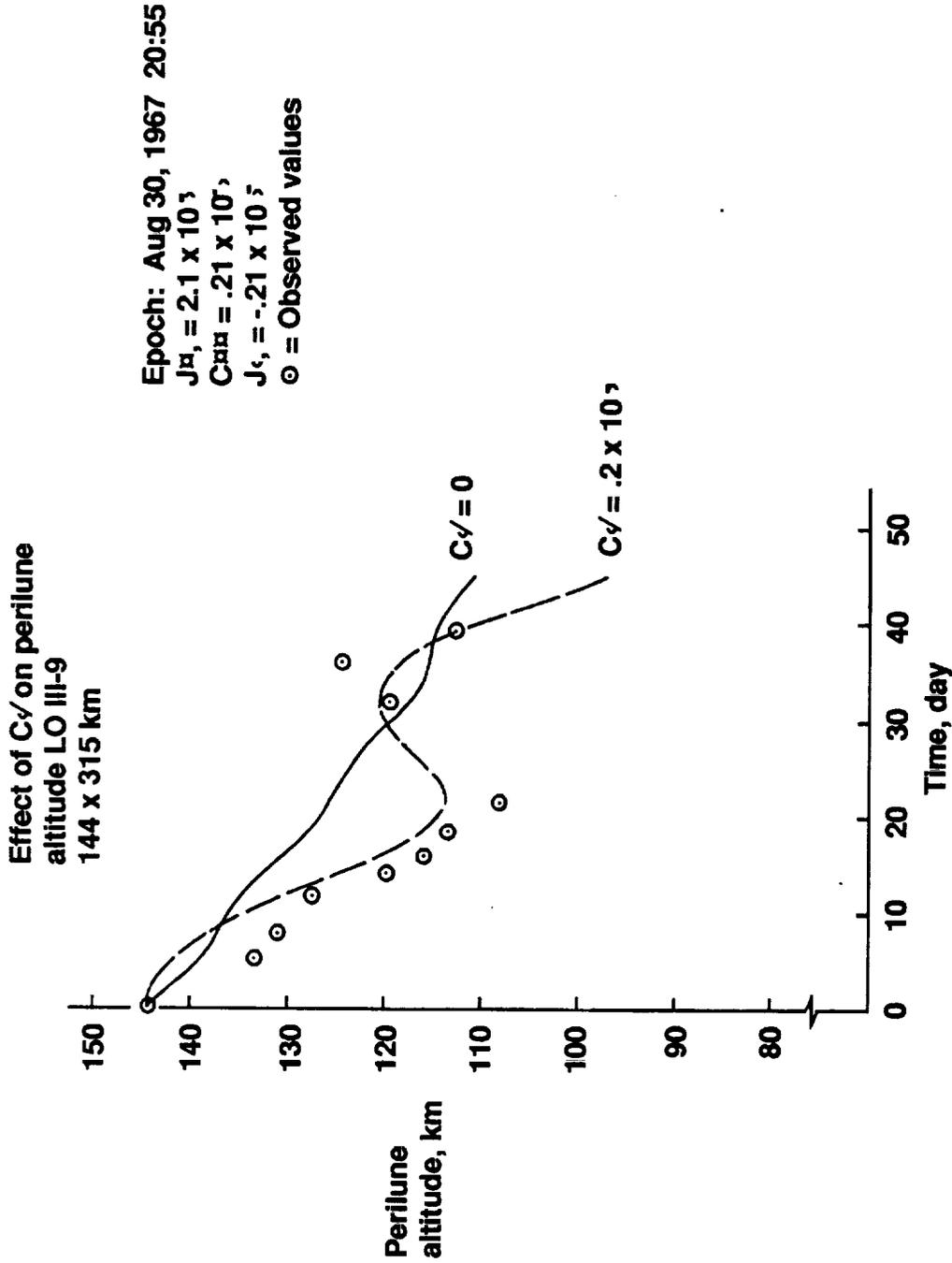
Figure 1-1.3.4.4-3. Model Variations
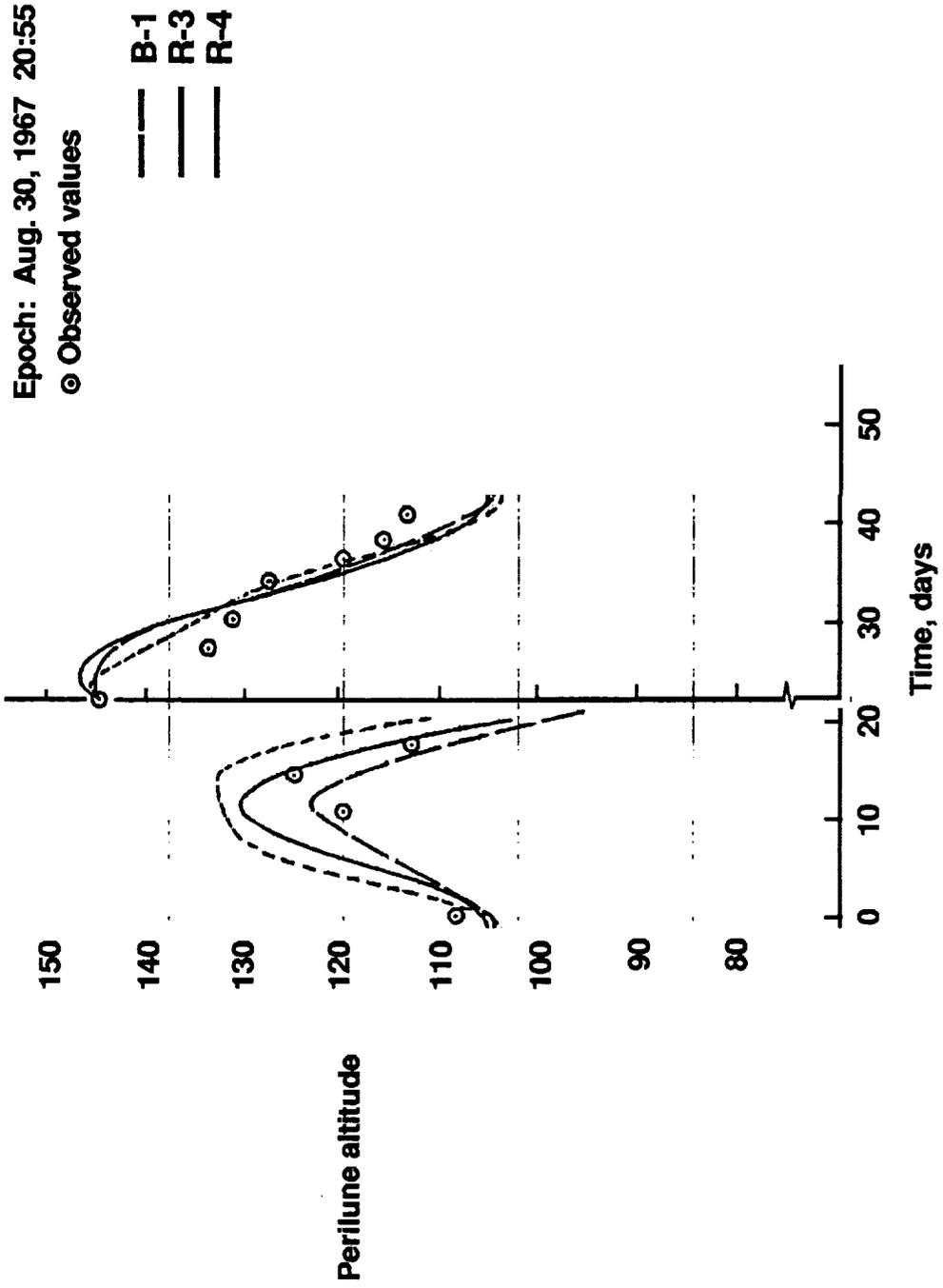
Figure 1-1.3.4.4-4.   Perilune Altitude Variation LO III Phase 9

dominance of a number of mass concentrations near the lunar surface has defeated attempts to solve for high-order coefficients.

1-1.3.4.5   Descent From Lunar Orbit

Two ways of descending from LLO to the surface have been analyzed to explore the range of choices possible for two-burn descent. Both start in 300-km lunar orbit with a short deorbit burn followed by a coast to near periapsis where the second and final burn starts.

The first method uses gravity turn for the final burn. Its coast periapsis altitude is about 50 km and the final portion of the descent is quite steep. The second method, low angle, uses the minimum possible coast periapsis altitude (2 km assumed) and a more nearly vertical altitude schedule during final burn to maintain a fairly flat descent.

Figure 1-1.3.4.5-1 shows these two trajectories starting from different first ignition points in orbit (above "B" for gravity turn and above "A" for low angle) to land at the same point on the surface on the other side of the Moon. This figure shows that the low-angle turn descent covers a smaller total angle about the Moon's center (about 165 degrees from "A" to landing site).

Figure 1-1.3.4.5-2 shows final burn descent profiles for the two types of descent. The difference in steepness of the final portion of the paths can be seen. Final conditions for both are 50m altitude and 5-m/s velocity. The low-angle descent requires about 40 m/s less total ΔV.

The capability in a two-burn descent to fly to landing sites other than the nominal is portrayed in Figure 1-1.3.4.5-3 for a low-angle descent. By steepening the descent, profile trajectories with range up to 2,200 km less than nominal are possible for an additional ΔV of 64 m/s. Introducing yaw into the deorbit burn, sites up to 197 km to the left or right of the nominal track are available. The resulting ellipse is shown in the figure. The note at the right shows that a landing site is attainable from a number of successive orbital passes by including yaw maneuvering.

Deorbit for low
angle descent

Deorbit for gravity
turn descent

A

B

Moon

Descent
burn

Landing
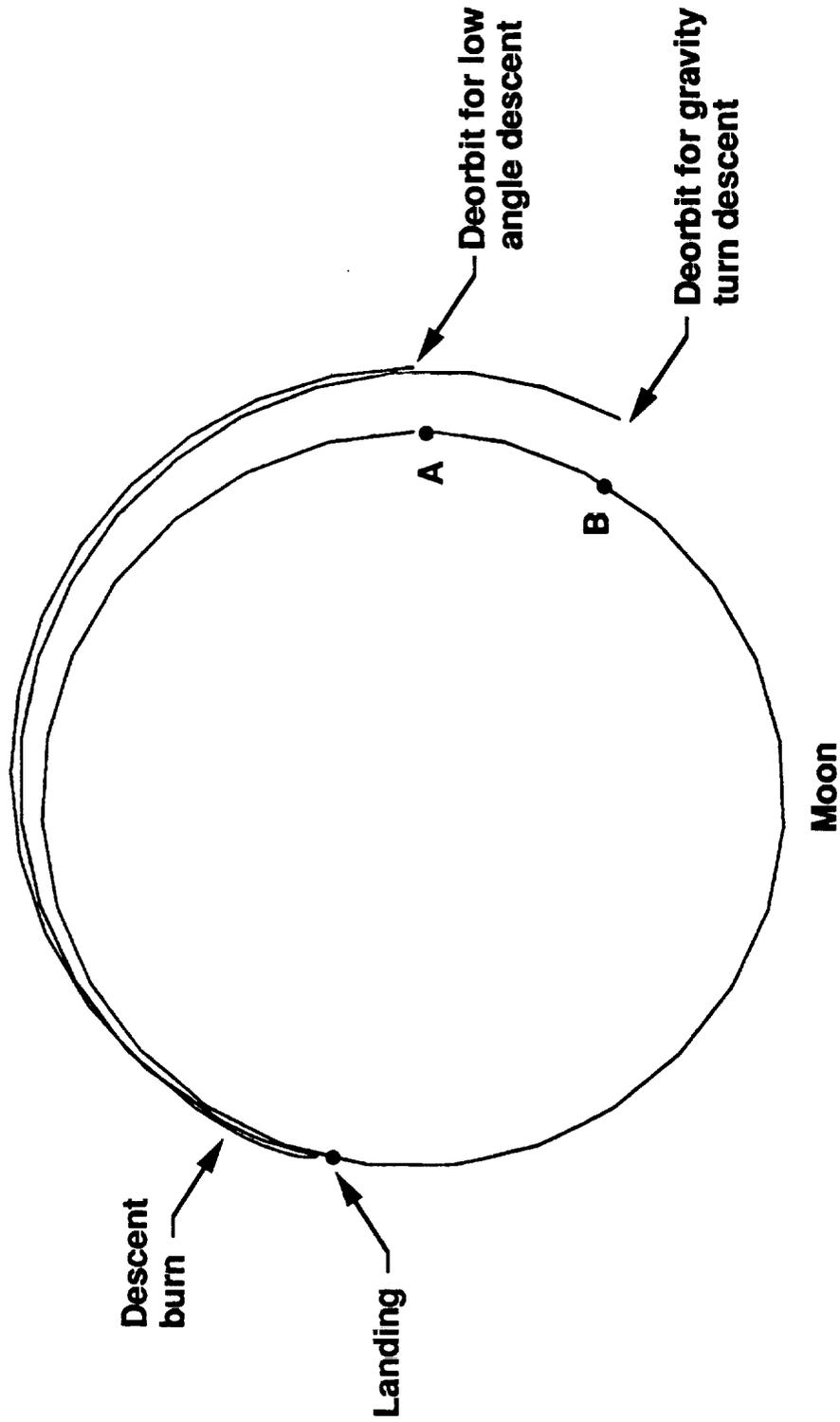
To scale

Figure 1-1.3.4.5-1. Descent From Low Lunar Orbit

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Figure 1-1.3.4.5-2. Descent Trajectory Options
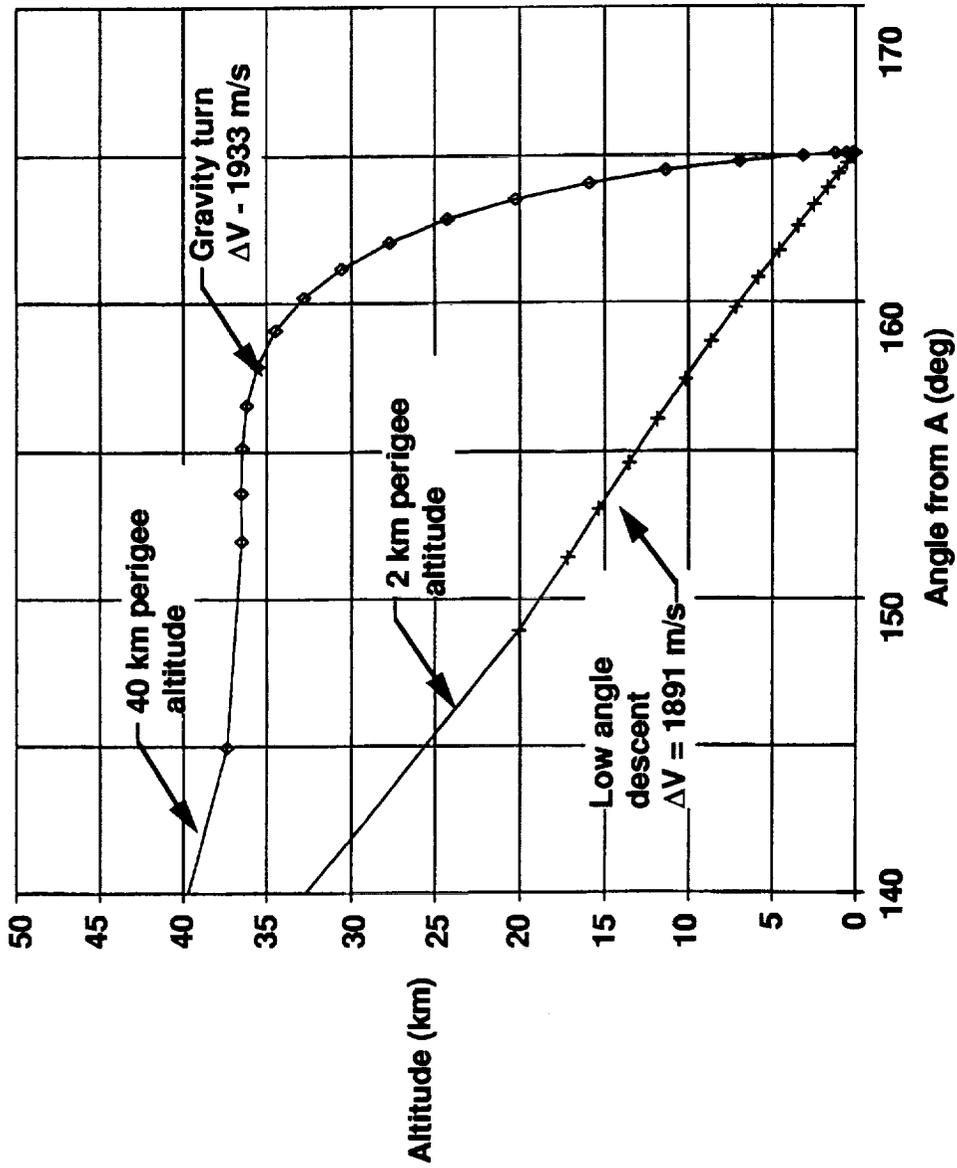
Conclusion: Low angle descent requires less ∆V than gravity turn

- Landing footprint for 64 m/s extra delta V

- Half width of ellipse equals 197 km (6.5 degree of central angle)
  Covers 5 i = 90 orbit tracks
  or 15 i = 20 orbit tracks
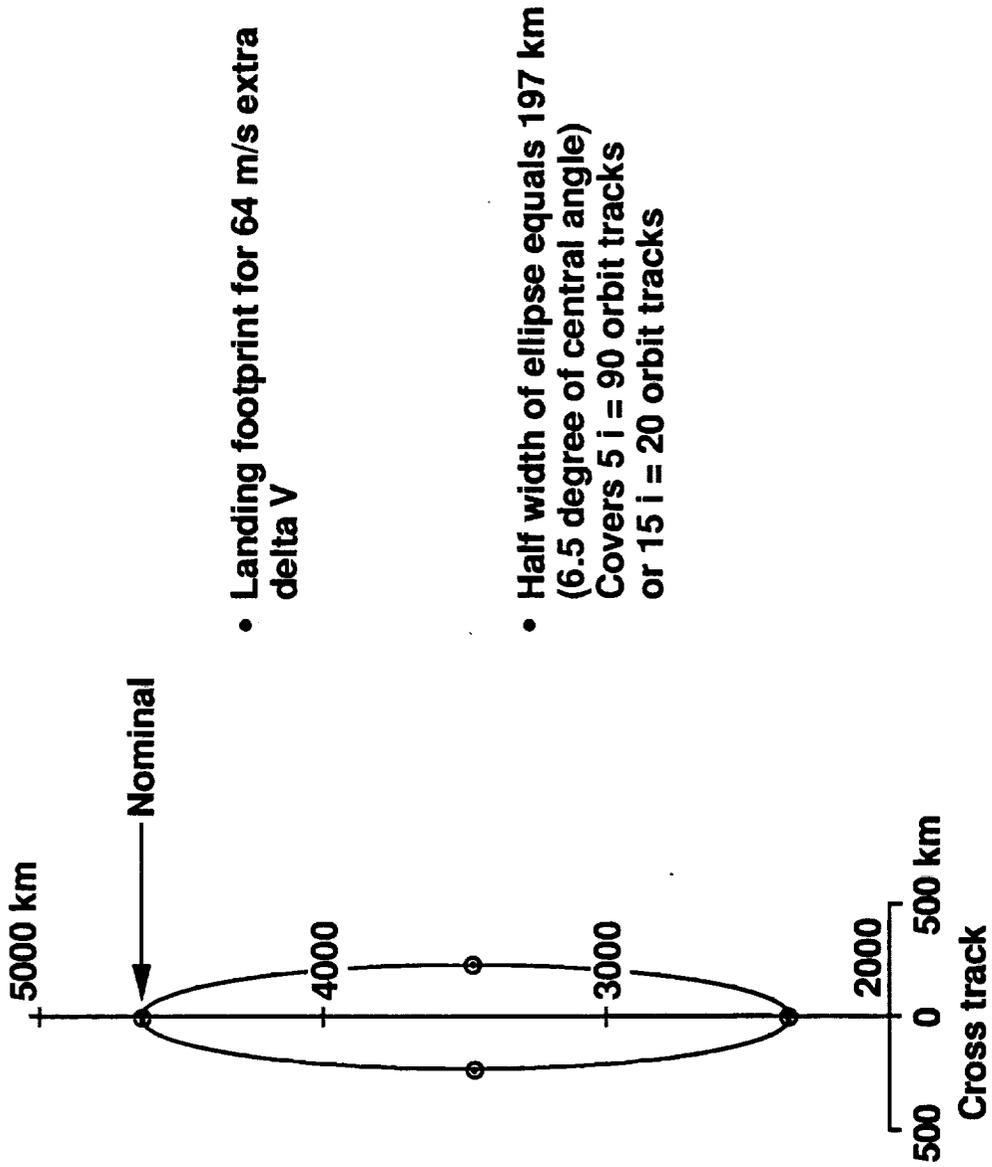
Conclusion: Landing footprint requires little extra propellant

Figure 1-1.3.4.5-3. Landing Footprint Capability In Orbit

1-1.3.4.6   Ascent to Lunar Orbit

In a manner similar to two-burn descents discussed in the previous section, alternatives for two-burn ascent to 300-km lunar orbit include low angle and gravity turn. These are illustrated in Figure 1-1.3.4.6-1 for an initial thrust-to-weight ratio of 0.2. This is less than the higher thrust descent analyzed above (thrust-to-weight ratio = 0.33), so the low-angle case requires nearly as much ΔV as the gravity turn (8-m/s difference).

1-1.3.4.7   Trans-Earth Injection

The TEI ΔV required to return to Earth from LLO was analyzed for effect of different landing site latitudes. The analysis assumed the first opportunity in January 1990 from a space station with ascending node on the vernal equinox, which has LOI on January 4. The first two TEI opportunities after that are on January 7 and January 18. Assuming equatorial lunar orbit as a reference case, Figure 1-1.3.4.7-1 shows the time and ΔV for LOI (circled) and for the first two TEI opportunities (X's). The reference points are joined by dashed line for clarity. These data pertain to any landing site on the equator. The ΔV variation reflects the changing hyperbola energy and latitude of the approach asymptote.

Additional data were generated for lunar orbit inclination of 10 degrees, with a landing site on the lunar prime meridian and an assumed 1 day from LOI to site overfly. Solid lines connect the points for site latitudes of 5 , 0, and -5 degrees. There is a total variation of 400 m/s, from 850 to 1250 m/s, in the LOI and TEI ΔV data.

1-1.3.4.8   Aerobrake L/D Required

One method of returning to the Space Station from the Moon is to aerobrake at perigee of the Moon-to-Earth trajectory, decelerating to an orbit with apogee at the Space Station altitude. A burn is then performed at apogee to match with the Space Station orbit for docking.

The concept of a lifting aerobrake maneuver provides a way to correct for uncertainties that exist at the time of atmospheric entry. Ignoring all error
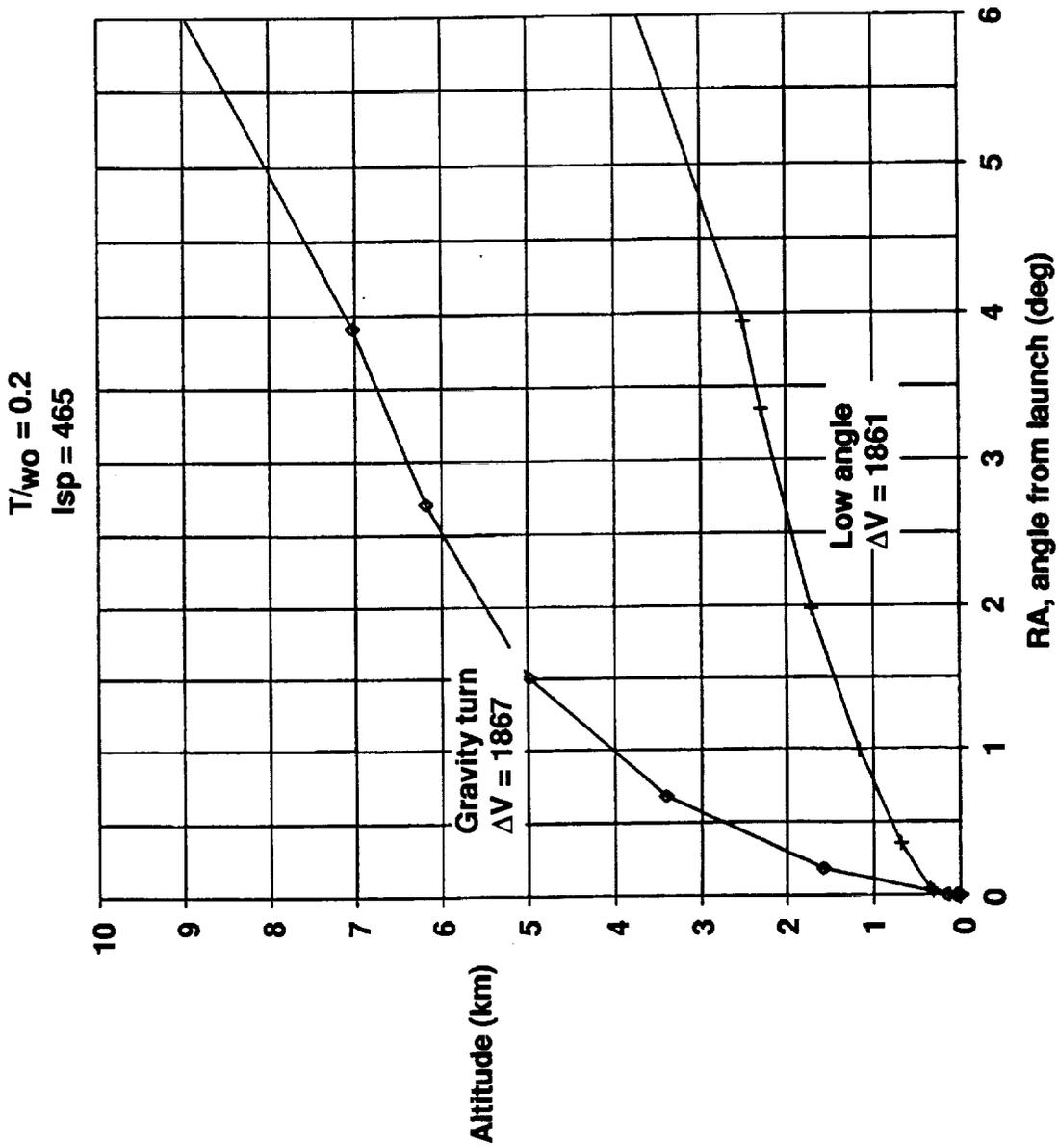
Figure 1-1.3.4.6-1. Low-Angle Ascent Trajectory - ΔV

LMS 315-3

- SSF RAAN = 0
- Three-day trips
- 300 km lunar orbit
- One day to overfly site
- No ascent
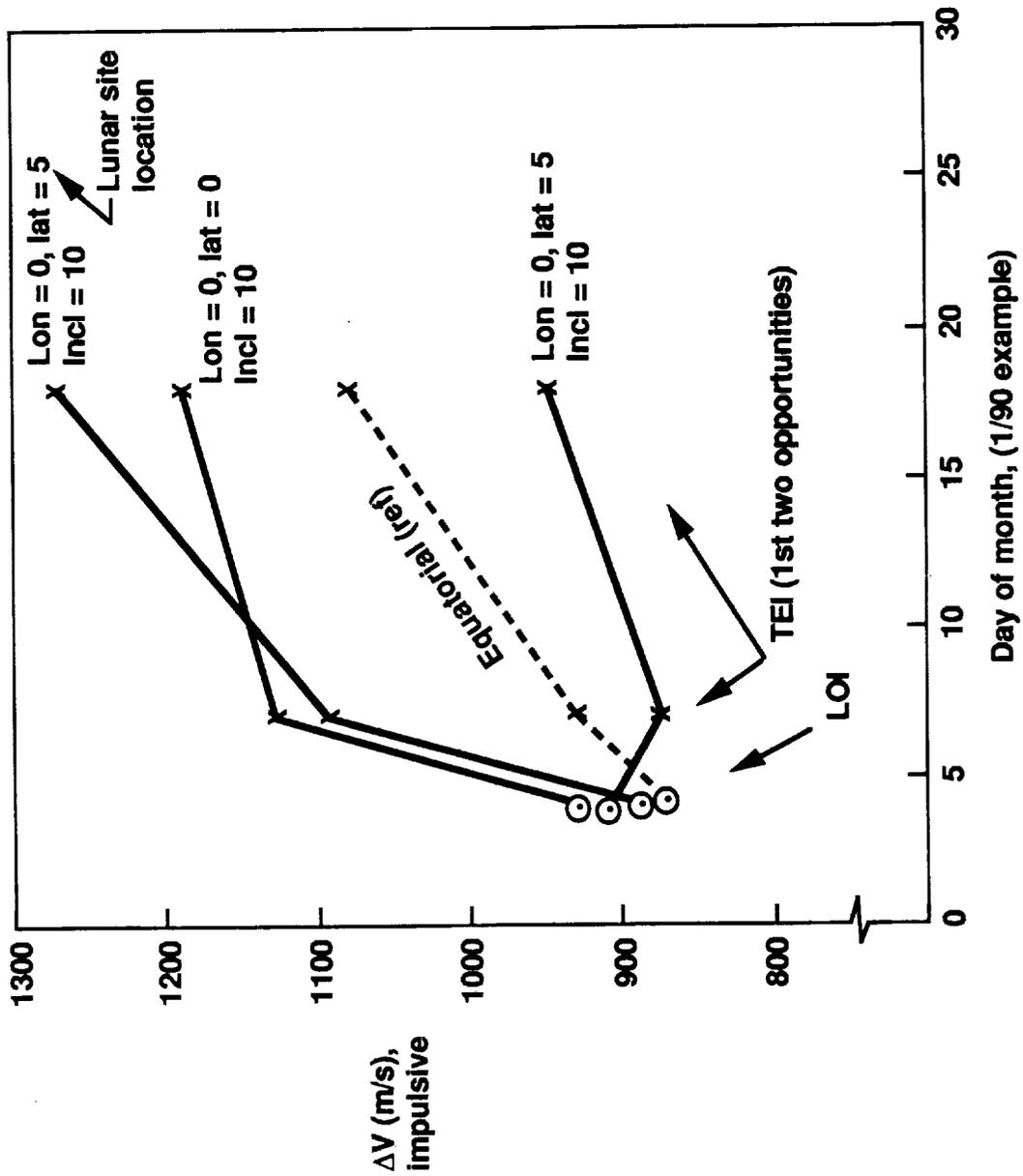
Figure 1-1.3.4.7-1. Effect of Lunar Orbit Inclination and Site Location

Lon = 0, lat = 5
Incl = 10

Lunar site location

Lon = 0, lat = 0
Incl = 10

Equatorial (ref)

Lon = 0, lat = 5
Incl = 10

TEI (1st two opportunities)

LOI

Day of month, (1/90 example)

ΔV (m/s), impulsive

1300
1200
1100
1000
900
800

0   5   10   15   20   25   30

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contributors except knowledge of atmospheric density, a study was made assuming the use of in-plane, normally directed (up or down) lift during the aerobrake maneuver to correct back to the same final apogee altitude. The following flight schedule was assumed starting at the atmospheric entry point (120-km altitude).

1. Fly at zero lift for 36 seconds to calibrate the atmospheric density (sensed as drag or drag rise).
2. Within 4 seconds acquire a new attitude as needed to attain a lift-to-drag (L/D) ratio appropriate for the density sensed.
3. Maintain constant L/D until the flight again reaches a 120-km altitude.

This approach is optimum in that a borderline high- or low-density atmosphere makes use of the L/D capability of the vehicle for the maximum period of time.

In the study, atmospheric density was based on the US Standard 1962 atmosphere, using a constant multiplier M at all altitudes (nominal M=1). Typical three-sigma high variation in the neighborhood of M=1.6 for the altitude range of interest is seen in the MSIS 1986 model. A range of M=0.5 to M=2.0 was assumed here. The fact that M will vary with altitude, as modeled in the Gram atmosphere for example, was ignored in this study as a second order effect. Attitude, navigation, and L/D errors were ignored also.

The study included the minimization of an entry flight path angle to minimize the Space Station orbit matching ΔV. This implies a similar capability on board for the midcourse correction targeting prior to entry and use of a defined "least dense" atmosphere (M=0.5 here) in the minimization process.

Figure 1-1.3.4.8-1 presents the study results in a plot of required L/D (minus downward and plus upward) as a function of the sensed density. Five vehicle configurations characterized by ballistic coefficients from 10 to 30 lb/ft^2 are shown. It is assumed that the deceleration is sensed (through accelerometer for example) and its rate of change at t=36 is calculated and made available on board in time for the maneuver to attitude to be completed by t=40. Symbols show the data points at M=0.5, M=1, and M=2 for each ballistic coefficient. Required L/D varies from about -0.15 to +0.10.
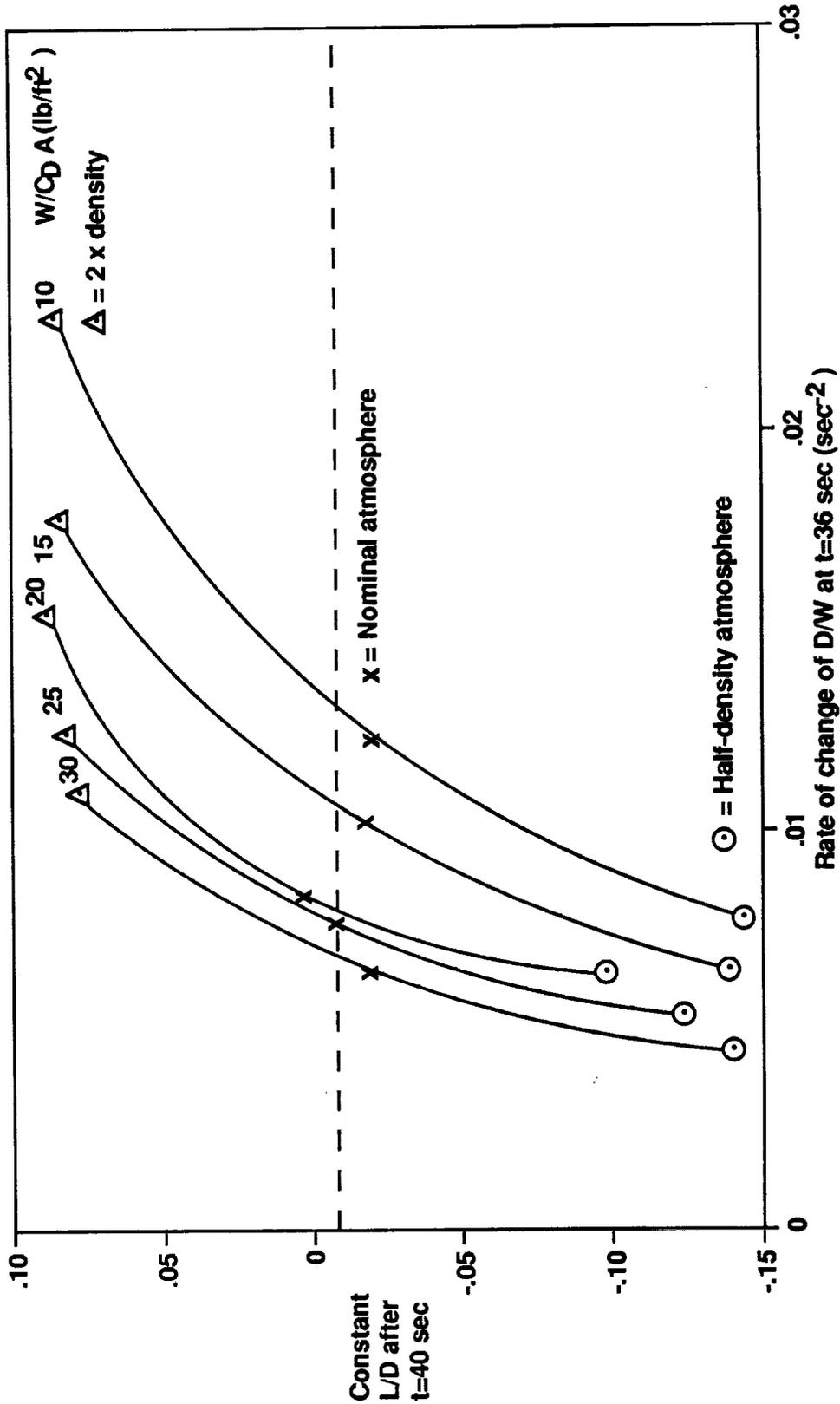
Figure 1-1.3.4.8-1. Aerobrake L/D Required

This study shows that regardless of ballistic coefficient, the required L/D is about 0.15 to correct for first order atmosphere variations. To this must be added the effects of side maneuvering as needed for return time flexibility, vehicle performance uncertainties, and altitude-dependent density variation.

1-1.3.4.9 Trajectory Options

Three options in targeting of the near-Moon portion of the trajectory were studied. Figures depicting these options are presented in section 1-1.3.4.3 in connection with abort considerations.

The first option is lunar orbit rendezvous (LOR) where LOI burn establishes low-inclination circular lunar orbit from the free-return approach path. The ascent vehicle returns to the vehicle in LLO prior to TEI.

The second option is lunar orbit direct (LOD), which departs from the free-return path about 1 day before arrival to target to a LLO having possibly high inclination passing over the landing site. The LLO is elliptical having minimum periapsis altitude (about 5 km), with the orbit oriented to put periapsis over the landing site. The descent is a single burn following fractional or multiple orbit coast. Similarly, ascent is a single burn to low periapsis, with possibly high inclination orbit from which TEI may occur after fractional or multiple revolutions. There is no relation between the lunar orbit used for descent and ascent in the LOD option.

The third option is direct, which departs from the free-return path about 1 day before arrival to target the approach hyperbola to the landing site. A single-burn descent is initiated from the hyperbola approach path. Similarly, ascent is a single burn that establishes a hyperbola departure orbit returning to Earth.

Site accessibility for LOR is limited to low latitudes due to the low-inclination free-return path. Direct cannot land on most of the far side and parts of the eastern and western limbs because of the fixed approach and departure hyperbola orientation. LOD can go to any site.

The ΔV requirement for LOD and LOR are similar. Direct landing and ascent incurs high gravity losses, especially over sites requiring a nearly vertical trajectory.

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1-2.0   INITIAL AND EVOLUTIONARY STV CONCEPT DEFINITION

1-2.1   INTRODUCTION

The purpose of this STV study was to identify and study a transportation system from Earth orbit to the lunar surface and look at its applicability to other orbital transportation needs. This section reports the recommended STV concepts resulting from system architecture trades, mission analyses, and subsystem trades as shown in Figure 1-2.1-1. These trades and analyses, reported in section 3.0 of book 1 and in section 1 of book 2, address issues including basing location, type of lunar transfer orbit, the optimum number of stages, the number of crew modules, and the method of recovery.

The selected concepts are shown in Figure 1-2.1-2 and include a space-based, single-stage vehicle with expendable droptanks; a ground-based, single-stage multiple-launch vehicle with expendable droptanks and lunar lander; and a ground-based, single-stage single-launch vehicle with expendable droptanks and lunar lander. The two ground-based concepts are similar in design, but the multiple-launch concept includes a LO2 tanker for filling vehicle LO2 tanks on orbit. All concepts have a single crew module for piloted missions and use a lunar-surface-direct transfer, requiring no rendezvous in lunar orbit. The space-based core vehicle uses an aerobraking maneuver to return the crew module and core stage to the Space Station or other LEO node, but on the ground-based vehicle, only the crew module returns to the ground and is recovered. These three concepts satisfy current study requirements and were chosen to carry forward for further study.

General STV design requirements and those that apply specifically to the lunar missions are given in Figures 1-2.1-3 and 1-2.1-4, respectively. The requirements shown are those that primarily affect the flight system.

Figure 1-2.1-1. System Architecture Evaluation Flow

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**Ground Based
Single Launch**

**Ground Based
Multiple Launch**

Space Based (SB)

**1 Launches
250 mt booster
No assembly
required**
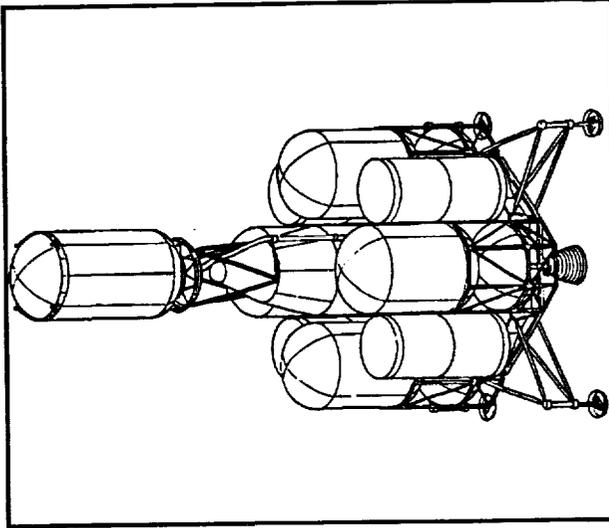
**2 Launches
125 mt booster
Docking and
refuelling**
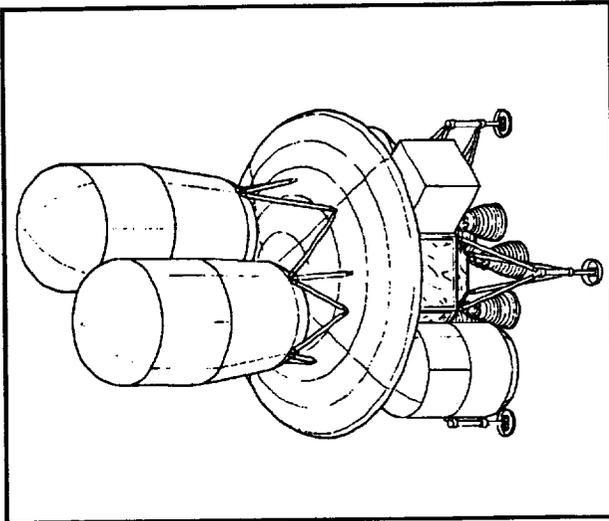
**5 Launches
70 mt booster
Assembly
required**

Figure 1-2.1-2. STV Concepts

	Basing		Requirement Source		
	Space	Ground	Given	Derived	Provis.
General Requirements					
Safety / Abort					
Free return aborts during all mission phases	X	X	X		
Safe haven capability at outpost	X	X	X		
Critical functions affecting crew safety to be two failure tolerant	X	X	X		
Provide two means of ingress / egress at all times	X	X		X	
EVA requirements					
Minimum of two crew members shall perform each scheduled EVA.	X	X	X		
In-space and surface EVA for each crew (not simultaneously).	X	X	X		
Service life / Life extension					
Minimum 5-mission life on reusable flight elements	X	X	X		
Initial flights of reusable vehicles will be in expendable mode	X	X	X		
Maintainability					
Shall be maintained, mated, serviced at Space Station	X		X		
Design for replacement at functional component level	X	X	X		
Provide for checkout tests of critical functions	X	X	X		
Robotic or EVA-maintained systems external to pressurized environ.	X	X	X		
Flight Performance Reserves					
Main propulsion - 2% FPR on each delta V maneuver	X	X			X
Reaction Control - 10% FPR of mission nominal propellant	X	X			X
Electrical Power - 20% FPR of mission nominal reactants	X	X			X
Hardware Disposal					
Disposal by controlled deorbit or other non-interference mode	X	X		X	
ETO Capabilities					
Ground-services will be through launch vehicle interface	X	X			X
HLLV capability - at least 58 tonne cargo or 68 tonne propellant	X	X	X		
10 m dia X 30 m payload shroud for cargo flights	X	X	X		
Maximum 6 cargo/ propellant launches per 12-month period	X	X	X		
Personnel transport capability - 4 crew, 20 tonne cargo per flight	X	X	X		
Maximum 2 crew/cargo flights per year	X	X	X		
HLLV Launch from ETR	X	X		X	

Figure 1-2.1-3. General STV Design Requirements

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BOEING

	Basing		Requirement Source		
	Space	Ground	Given	Derived	Provis.
Lunar Mission Requirements					
Schedule					
1999 - IOC	X	X	X		
2002 - First Cargo lunar flight, ETO Cargo delivery	x	x	x		
2004 - First piloted lunar flight, ETO crew delivery	x	x	x		
Lunar Mission Duration					
Nominal mission - 12 day support of personnel	X	X		X	
Abort contingency - TBD days	x			x	
Lunar Landing					
Initial and far side autonomous landing on unprepared surface.	X	X		X	
TBD m diameter landing site					
TBD degree slope					
TBD surface irregularity					
Piloted - landing on prepared surface	X	X	X		
50 m diameter landing site					
<2 degree slope					
< 0.2 m surface irregularity					
Capable of cargo jettison in case of landing abort.	X	X		X	
Lunar Stay					
Capable of offloading cargo on lunar surface.	X	X		X	
Lunar vehicle for up to 48 hours duration	x	x	x		
Power - 2 kW (ave), 3 kW (peak)					
Remote payload release latches					
Life support for crew for minimum 48 hours post-landing					
1 kW heat rejection					
200 kb/s data communication					
Planetary Surface Support (PSS) for up to 6 months duration	X	X	X		
Power - 2 kW					
Thermal control and Protection: 3 kW Heat rejection					
Propellant Conditioning to accomodate 4% boiloff per month					
Structural support for LEV maintenance					
Data communications w/ 200 kb/s fdata link to LEV					
Payload offloading					
Provide navigation aids					
Recovery					
Dry landing and recovery of Earth return hardware		X		X	
Aerobrake at Earth return (11.1 km/s max)	x		x		
Lunar Payload Capabilities					
Capable of transporting 4 people to lunar surface and back.	x	x	x		
100 - 500 kg return sample capability from the lunar surface.	x	x	x		
16.2 tonne cargo with piloted mission	x	x	x		
33 tonne cargo delivery capability	x	x	x		
61.3 tonne cargo to surface in first three flights (2 cargo, 1 piloted)	x	x	x		
418 tonne total cargo to lunar surface	x	x	x		

Figure 1-2.1-4. Lunar Mission STV Design Requirements

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1-2.2   SPACE-BASED CONCEPT DESCRIPTION

This section discusses the configuration of a space-based STV with selected flight elements based at the Space Station or other LEO node. It includes a top-level description of the core stage, crew module, and droptank sets and gives mass properties, performance, launch and recovery operations, and the use of lunar-designed flight elements for capture of other non-lunar missions.

1-2.2.1 System Design and Operation

A few of the issues addressed by the current space-based concept include the following:

1. Two engine-out operation capability.
2. Fit within the launch shroud diameter.
3. Visibility of lunar landing pads and horizon.
4. Aeromaneuver capability, including minimizing wake impingement, meeting L/D requirements, and keeping within TPS limitations.
5. Vehicle reusability.

The selected space-based concept is a cryogenic vehicle with a reusable core stage and two pairs of expendable droptanks, as shown in Figure 1-2.2.1-1. 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 droptanks for both missions include a pair of tanksets holding translunar 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.

The lunar mission sequential configuration of the vehicle is depicted in Figure 1-2.2.1-2. The aerobrake must be launched in sections to fit in the launch shroud, 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 aeromaneuver, as shown in Figure 1-2.2.1-3. The TLI tanksets, lunar descent tanksets, and cargo are launched in

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

TLI drop tanks

Cargo volume (2)

Descent tanks (2)
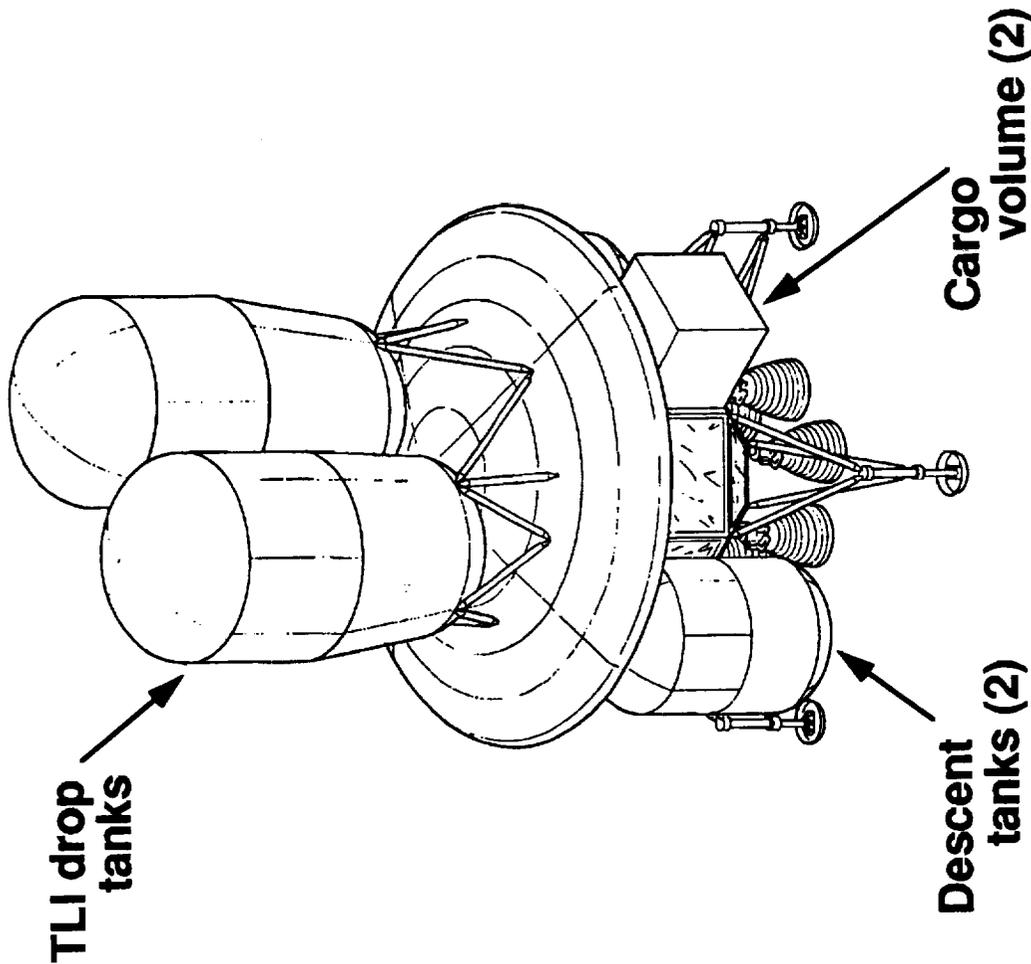
Figure 1-2.2.1-1. Space-Based Vehicle
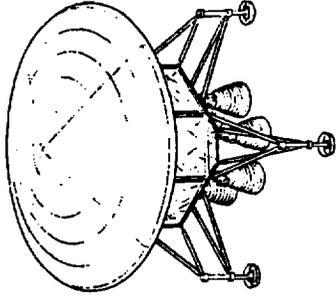
Lunar ascent
Vehicle core and aerobrake
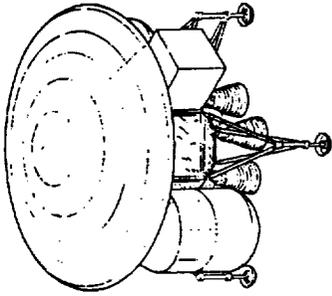
Lunar arrival
TLI tanks discarded
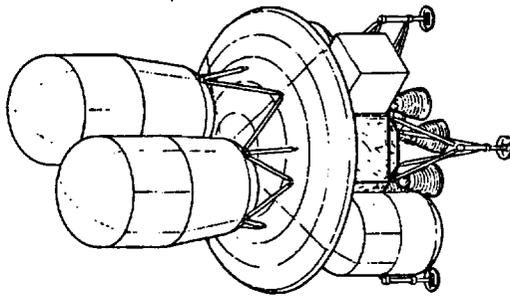
Mission start
Fully assembled configuration

Figure 1-2.2.1-2. Space-Based Vehicle Configuration Sequence

Crew module

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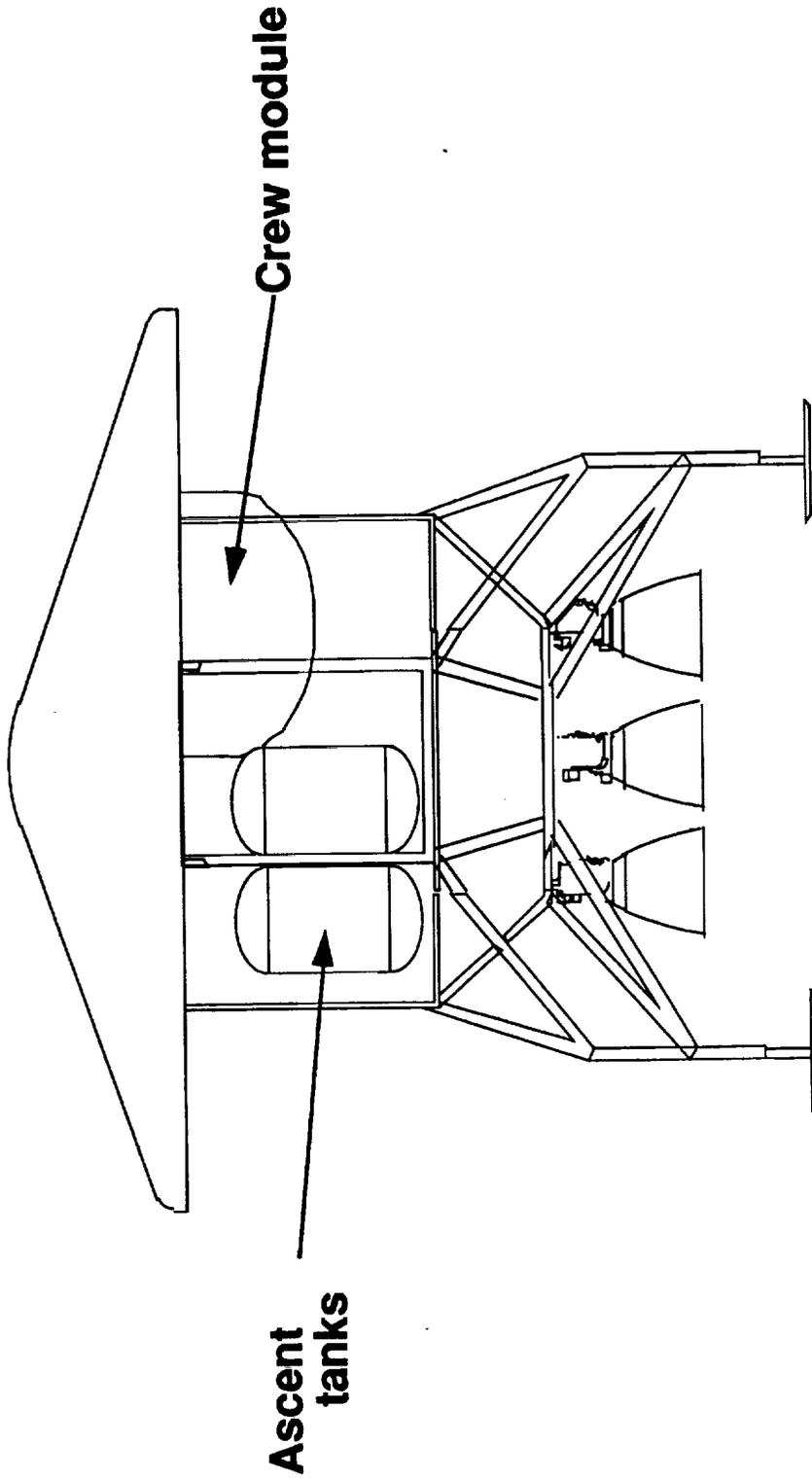
The returning vehicle is asymmetric and has an offset
crew cab to provide the cg offset required for
aerobraking.

Figure 1-2.2.1-3. Space-Based Crew Module Integration

three to four launches and integrated with the core, and the core tanks are filled from a LEO tank farm prior to each mission.

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, 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. Because of the aerobrake overhang, cargo must be unloaded from the side of the core and moved to the base, either with built-in provisions or using a lunar flatbed trailer, as shown in Figure 1-2.2.1-4. At the end of the lunar stay, the crew loads return cargo, boards, and checks out the vehicle, then the core vehicle ascends and returns to the LEO node, using an aeromaneuver, where it is inspected and refurbished for the next flight.

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

1-2.2.2 Subsystem Overview

The space-based vehicle is made up of the following subsystems, as shown in Figure 1-2.2.2-1:

1. Structures and Mechanisms - Includes a core stage with external load-bearing body structure and landing gear, a rigid aerobrake, a pressurized crew module, two sets of TLI droptanks, and two sets of descent droptanks.
2. Main Tankage - The core stage has two cylindrical LO2 tanks and two cylindrical LH2 tanks with associated propellant acquisition devices. Each droptank set has a single LO2 tank and a single LH2 tank with associated slosh baffles and propellant acquisition devices.
3. Protection - Includes thermal control and damage protection of the main cryogenic tanks, thermal control of avionics and power equipment, thermal and radiation protection of the crew during long-duration exposure in

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TLI tanks actually
jettisoned before landing
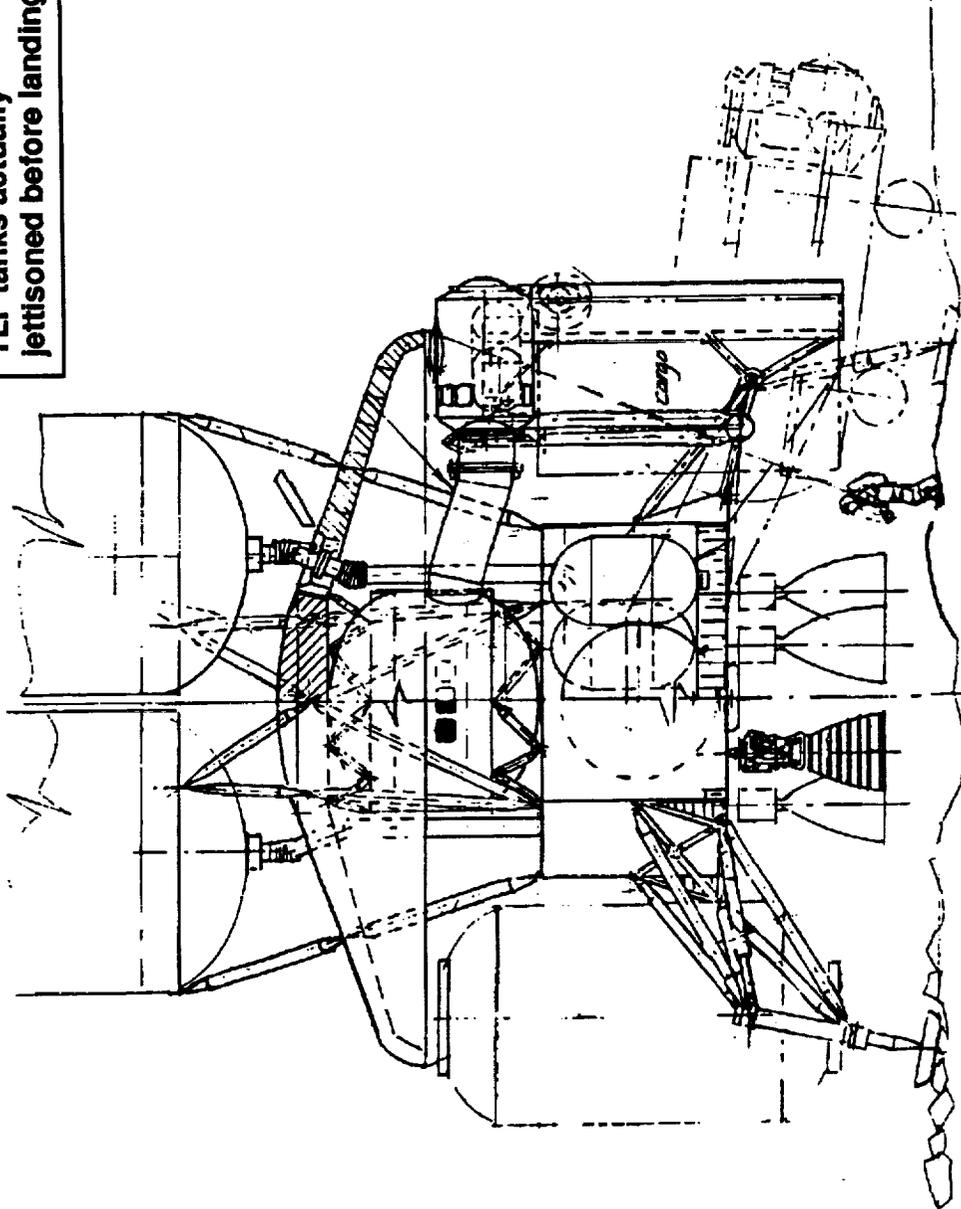
Figure 1-2.2.1-4. Space-Based Payload Unloading

Aerobrake
- 15.3 m dia
- GR/PI structure
- TPS
  - Fibrous ceramic
  - Mechanical Attach.
  - Ti fittings

Structures & Mechanisms
- Aluminum
- Graphite composites
- Gas filled shock struts on landing gear

Propulsion
Advanced Space engine
- Thrust = 15000 lbf

Drop Tanks
Aluminum-Lithium
Titanium fittings
MLI blankets
Meteroid protection

Crew Module
- Four crew
- Two EVA hatches
- Open ECLSS
- ALSPE shelter

Power
- GH/GO fuel cells
- Ni/H2 batteries

Reaction Control
- GO2/GH2 thrusters
- Accumulators fed from main tanks

Figure 1-2.2.2-1. Space-Based Subsystem Overview

space, and thermal protection of the aerobrake during the aerobraking maneuver.

4. Main Propulsion - Consists of six advanced expander-cycle engines with electromechanical actuation and propellant delivery, pressurization, fill, and vent systems.

5. Reaction Control - Includes four GO2/GH2 thruster modules and associated accumulators, pressurization, and control.

6. Electrical Power - 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.

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

8. Communication and Data Handling - Provisions for communication, vehicle health maintenance, and data handling, with audio and video interfaces for piloted operations and instrumentation for droptank monitoring and control.

9. Displays and Controls - Provisions on the crew module for limited crew control and status monitoring of the vehicle during critical phases of the mission.

10. Environmental Control - Provisions on the crew module for atmosphere supply and control, internal equipment cooling, and metabolic and equipment heat rejection.

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

1-2.2.3 Mass Properties

Mass summaries for the space-based STV concept are given in Figures 1-2.2.3-1 and 1-2.2.3-2 for the piloted and unpiloted lunar missions, respectively. A weight growth margin of 15% 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.

The current space-based vehicle concept can either deliver 9,870 kg of 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 tons of cargo is delivered to the lunar surface over

Lunar Piloted Mission

	Lander	Core Stage Crew Module	Aero-Brake	TLI Drop-Tankset Module #1	Module #2	LD Drop-Tankset Module #1	Module #2	Lunar Surface Cargo
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	-	15	23	23	23	23	
Communication & Data Handling	422	124	-	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	
	29974	7383	4110	71034	71034	30801	30801	9870
	41467			142068		61603		
	255007							
Total LEO-Assembled Mass								9870

Figure 1-2.2.3-1. Space-Based STV Mass Summary - Lunar Piloted
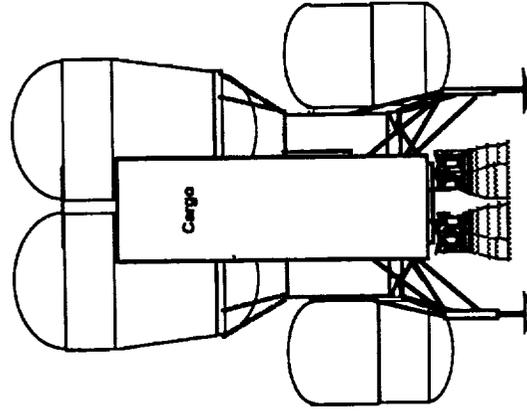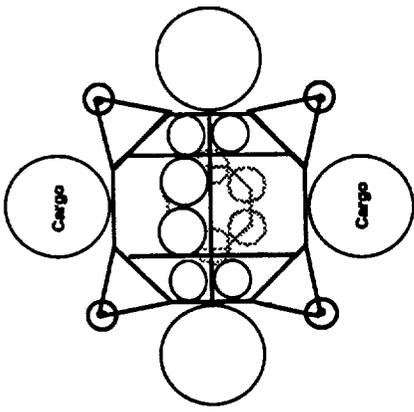
	Core	TLI Drop-Tankset		LD Drop-Tankset		Lunar Surface Cargo
	Lander	Module #1	Module #2	Module #1	Module #2	
Structure and Mechanisms	1935	884	884	474	474	
Tankage - Main	583	1089	1089	597	597	
Protection	437	642	642	385	385	
Propulsion - Main	2126	393	393	257	257	
Propulsion - Reaction Control	347	-	-	-	-	
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	-	-	-	-	-	0
Weight Growth Margin	1068	460	460	266	266	
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				

Lunar Cargo Mission - Unmanned

Figure 1-2.2.3-2. Space-Based STV Mass Summary - Lunar Cargo

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21 piloted and 4 cargo-only missions, and the size of the vehicle is common to both piloted and cargo-only missions.

Summary and sequential mass properties for the space-based STV are shown in Figure 1-2.2.3-3. The longitudinal center of mass ranges from 7.66m from the engine gimbal plane at startburn to 4.73m from the engine gimbal plane prior to lunar orbit insertion. At lunar landing, the cg is about 8.75m from the landing pad plane.

1-2.2.4 Performance

The selected space-based STV main propulsion system is a LO2/LH2 system and uses advanced engines with a vacuum thrust of 15,000 lb per engine and an assumed specific impulse of 481 seconds. The reaction control system is a gaseous O2/gaseous H2 system with an assumed specific impulse of 410 seconds. The selected vehicle concept is designed to satisfy the piloted and unpiloted lunar missions, with flight elements capable of capturing other non-lunar missions.

Lunar Mission Performance. A mission timeline for the piloted lunar mission is given in Figure 1-2.2.4-1. Sequential mass and fluid inventories for the lunar piloted and lunar cargo-only missions are given in Figures 1-2.2.4-2 and 1-2.2.4-3, respectively. Included are main and auxiliary propulsion fluids, non-propulsive consumables, waste fluids, and sequential time and power levels.

Evolutionary Mission Capture. For capture of unpiloted non-lunar missions, the core stage of the lunar-designed vehicle without landing gear and with fewer engines can be used as a delivery stage. Prior to advanced engine development, the stage can be used with RL10 engines. Specific mission requirements based on the 1989 civil needs database (CNDB) are given in Figure 1-2.2.4-4. As was mentioned before, the delivered cargo for lunar missions has been optimized for the specific vehicle design and is different from the CNDB design payloads. A configuration and mass summary of the current space-based lunar vehicle flight elements used for capture of the CNDB missions is shown in Figure 1-2.2.4-5. The core stage with crew module and

Boeing

	ITEM	Mass - kg W3	W2	CENTER OF MASS - m Xcg	Ycg	Zcg	MOMENTS (KG-M^2) IXX	IYY	IZZ
Mass Properties Summary									
CARGO			9870	3.81	0.00	0.00	3.225E+05	3.295E+05	2.221E+04
	CARGO 1	4935		3.81	0.00	5.58	7.588E+03	1.110E+04	1.110E+04
	CARGO 2	4935		3.81	0.00	-5.58	7.588E+03	1.110E+04	1.110E+04
CREW MODULE			5584	6.35	0.00	2.16	9.079E+03	9.079E+03	9.079E+03
	CREW MODULE	4493		6.35	0.00	2.16	8.806E+03	8.806E+03	8.806E+03
	CREW	400		6.35	0.00	2.16	1.000E+02	1.000E+02	1.000E+02
	EVA SUITS	400		6.35	0.00	2.16	1.000E+02	1.000E+02	1.000E+02
	CREW CONSUMABLES - total	291		6.35	0.00	2.16	7.275E+01	7.275E+01	7.275E+01
STAGE- (D&A Lander) P/A MODULE			29974	4.24	0.00	-0.53	1.602E+05	1.696E+05	2.175E+05
	STAGE INERT	8522		2.22	0.00	-0.04	1.025E+05	1.047E+05	1.107E+05
	PROPELLANT	21452		5.04	0.00	-0.73	5.477E+04	1.343E+04	5.815E+04
AEROBRAKE			4109	7.87	0.00	0.00	1.220E+05	8.028E+04	8.028E+04
DROP TANK SET-TLI (2 SETS)			143856	9.94	0.00	0.00	1.491E+06	1.021E+06	2.476E+06
	TANKSET INERT	4301		13.33	3.18	0.00	1.826E+04	5.975E+04	5.975E+04
	PROPELLANT	67627		9.73	3.18	0.00	0.000E+00	3.982E+05	3.982E+05
	TANKSET INERT	4301		13.33	-3.18	0.00	1.826E+04	5.975E+04	5.975E+04
	PROPELLANT	67627		9.73	-3.18	0.00	0.000E+00	3.982E+05	3.982E+05
DROP TANK SET-Descent (2 SETS)			61618	4.71	0.00	0.00	2.746E+06	2.624E+05	2.703E+06
	TANKSET INERT	2424		3.18	6.48	0.00	1.644E+04	2.145E+04	1.120E+04
	PROPELLANT	28385		4.84	6.48	0.00	6.295E+04	1.036E+05	4.065E+04
	TANKSET INERT	2424		3.18	-6.48	0.00	1.644E+04	2.145E+04	1.120E+04
	PROPELLANT	28385		4.84	-6.48	0.00	6.295E+04	1.036E+05	4.065E+04
SEQUENCED MASS DATA									
LEO ASSEMBLY			255011	7.66	0.00	-0.02	4.886E+06	3.698E+06	7.300E+06
START TRANS-LUNAR INJECTION			249013	7.57	0.00	0.00	4.815E+06	3.614E+06	7.124E+06
PRIOR TO LUNAR ORBIT INSERTION			110373	4.73	0.00	0.00	3.284E+06	9.623E+05	2.990E+06
LLO OPERATIONS			87860	4.74	0.00	0.00	2.280E+06	8.663E+05	2.002E+06
LUNAR LANDING			58389	4.75	0.00	0.01	9.669E+05	7.408E+05	7.095E+05
BEGIN LUNAR ASCENT			42029	4.99	0.00	0.03	3.406E+05	3.738E+05	3.763E+05
START TRANS EARTH INJECTION			28476	5.35	0.00	0.30	2.619E+05	2.794E+05	2.666E+05
START AEROMANEUVER			20474	5.64	0.00	0.37	2.012E+05	2.076E+05	1.960E+05
EOM MASS			19050	5.75	0.00	0.43	1.924E+05	1.953E+05	1.827E+05

Figure 1-2.2.3-3. Summary Space-Based Mass Properties

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Event	Event Duration (Hours)	Mission E.T. (Hours)	Mission E.T. (Days)	Sequenced Mass (kg)	ΔV (m/s)
Launch #1 - Core, crew mod., and aerobrake	0.0	0.0	0.0	16791	
Storage at SSF (21 days between launches)	504.0	504.0	21.0	16791	
Launch #2 - LD droptanks, and PSS cargo	0.0	504.0	21.0	88264	
Storage at SSF	504.0	1008.0	42.0	87921	
Launch #3 - First TLI tankset	0.0	1008.0	42.0	158955	
Storage at SSF	504.0	1512.0	63.0	156500	
Launch #4 - Second TLI tankset	0.0	1512.0	63.0	227534	
Final vehicle integration	72.0	1584.0	66.0	227534	
Add protective water	0.0	1584.0	66.0	229334	
Fill core vehicle propellant tanks	4.0	1588.0	66.2	251119	
Flight readiness verification	96.0	1684.0	70.2	251119	
Vehicle closeout	192.0	1876.0	78.2	251119	
Launch #5 - Crew, crew consum. (STS)	0.0	1876.0	78.2	246788	
Crew ingress	1.0	1877.0	78.2	247879	
Final vehicle checkout	4.0	1881.0	78.4	247879	
Separate from LEO node	0.0	1881.0	78.4	247879	
Departure prox ops	4.3	1885.3	78.6	247879	
Wait for TLI node (up to one revolution)	1.5	1886.8	78.6	247879	
Trans Lunar injection (TLI) burn	0.2	1887.0	78.6	120761	3300
Stage TLI droptanks	0.0	1887.0	78.6	110585	
TCM to "free return" trajectory	0.1	1887.1	78.6	110585	
Lunar transit	72.0	1959.1	81.6	110585	
TCM (target to landing site)	0.0	1959.1	81.6	110585	95
Lunar approach	12.0	1971.1	82.1	110585	
First lunar descent burn	0.2	1971.3	82.1	87085	1075
Low Lunar orbit coast (up to one revolution)	2.0	1973.3	82.2	87085	
Lunar landing burn	0.2	1973.5	82.2	54467	1920
Crew remains in transit module	48.0	2021.5	84.2	54467	
Crew transfer to habitat module	0.0	2021.5	84.2	53667	
Unload cargo	0.0	2021.5	84.2	43797	
Lander on surface with no surface support	672.0	2693.5	112.2	43497	
Lander activation, crew ingress with return P/L	4.0	2697.5	112.4	44797	
Transfer LD droptanks residuals to core tanks	1.0	2698.5	112.4	44797	
Drop descent tanks	0.0	2698.5	112.4	39967	
Lunar ascent burn	0.2	2698.7	112.4	26585	1822
Low Lunar orbit coast	2.0	2700.7	112.5	26585	
Second Lunar ascent burn	0.2	2700.9	112.5	19894	1075
Earth transit	72.0	2772.9	115.5	19894	16
TCM	0.0	2772.9	115.5	19894	
Earth approach	12.0	2784.9	116.0	19894	
Dump protective water	0.1	2785.0	116.0	18094	
Final TCM	0.1	2785.1	116.0	18094	
Aeroassist maneuver	0.1	2785.2	116.0	18094	
Coast	0.8	2786.0	116.1	18094	
Earth orbit circularization burn	0.1	2786.1	116.1	16821	310
LEO node rendezvous orbital maneuvers	48.0	2834.1	118.1	16821	
LEO node arrival prox ops	4.3	2838.4	118.3	16821	
Vehicle closeout	1.0	2839.4	118.3	16821	
Crew egress	1.0	2840.4	118.3	15730	
Remove Lunar payload	1	2841.4	118.4	15230	

E.T. - Elapsed Time
TCM - Trajectory Correction Maneuver

Figure 1-2.2.4-1. *Space-Based Nominal Timeline*

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Figure 1-2.2.4-2. Sequential Mass and Fluid Inventory Lunar-Piloted Mission

The table below records mass and fluid inventory data across the lunar mission sequence. Column groups: Delta V (m/sec) [MPS, RCS], Time (Hrs), Pwr [kW, kW-hr], Fluids (kg) [MPS nom, MPS Losses, RCS, Pwr Fluids], ECLSS Consumables [Food, Water, O2, N2, Metabolic], Stored Waste [Fluids], Inert Mass (kg), Delta Mass (kg), Total Mass (kg).

LUNAR TRANSFER SEQUENCE

Sequence	Delta V MPS	Delta V RCS	Time Hrs	Pwr kW	Pwr kW-hr	MPS nom	MPS Losses	RCS	Pwr Fluids	Food	Water	O2	N2	Metabolic	Waste Fluids	Inert Mass	Delta Mass	Total Mass
LEO Mass (when applicable)																17124	17124	17124
Fill Core Vehicle			0.0	1.0	0.0	+21453		0	0	0	0	0	0	0	0	0	+21453	38577
LEO stay prior to Launch #1			0.0	1.0	0.0	+56769		0	0	0	0	0	0	0	0	+14718	+71487	110064
LEO stay prior to Launch #2			504.0	1.0	504.0	+67627	-1080	0	0	0	0	0	0	0	0	+4302	+70848	180912
LEO stay prior to Launch #3			504.0	1.0	504.0	+67627	-1999	0	0	0	0	0	0	0	0	+4302	+69929	250841
LEO stay prior to Launch #4			504.0	1.0	504.0		-2905	0	0	+109	+52	+25	+104	0	+2119	+800	+304	251145
Departure Preparations			370.0	4.0	1480.0	0	-2104	0	0	-7	-22					-2133	249012
Start Trans-lunar Injection						203710		645	1033	109	312	78	82	0	1800	41245		249012
Coast		3	4.5	5.0	22.5	0	-25	-186	-8	-2	-4	-1	0	+1	0		-224	248788
Trans-lunar Injection	3300		0.2	5.0	1.0	-125187	-22	0	0	0	0	0	0	+0	0	-8603	-125210	123578
Drop TLI Tankset				5.0	0.0	-1717		-19	-1								-10340	113238
Trans-lunar Coast		3	72.0	5.0	360.0		-154	-84	-131	-27	-65	-11	-1	-22	0		-453	112786
Mid-Course Correction	95	0	0.0	5.0	0.0	-2249	-5	-82		-6	-11	-2	0	+4	0		-2270	110516
Coast		3	12.0	5.0	60.0	0	-25		-22						0		-143	110373
Prior to Lunar Orbit Insertion	1075		0.0	5.0	0.0	-22492	-21	0	0	0	0	0	0	0	0		-22513	87860
Lunar Orbit Insertion	0	0	0.0	5.0	0.0	0	0	0	0	0	0	0	0	0	0		0	87860
Orbital Rendezvous			0.0	0.0	0.0													87860
Add LLO Mass																	-87860	0
Lunar Orbit Operations			0.0	5.0	0.0	-51796		-273	-871	-76	-232	-64	-80	-27	-1800	-32642	-87860	0
Drop LEV, Payload			0.0	0.0	0.0													28448
On-orbit Stay	95	0	0.0	5.0	0.0	+7590	0	+66	+227	+53	+179	+48	+57	+3	+1800	+18424	+28448	28448
Add LEV						0		0	0	0	0	0	0	0	0		0	28448
Start Trans-Earth Injection								-21	0	0	0	0	0	0	0		-21	28427
Trim Burn	1075	3	0.0	4.0	0.0	-5793	0	0	0	0	0	0	0	+0	0		-5814	22612
Trans-Earth Injection		3	0.2	4.0	0.8		-21	-17	-122	-32	-76	-13	-2	+25	0		-240	22372
Trans-Earth coast	16	0	84.0	4.0	336.0		-5	0	0	0	0	0	0	+0	0		-97	22274
Mid-course Correction			0.2	4.0	0.8	-76	-21							+0	0		-1800	20474
Dump Protection Water			0.0	4.0	0.0	0		0	0	0	0	0	0	0	-1800			20474
Start Aeromaneuver								0	0	0	0	0	0	0	0		0	20474
Dump Protection Water			0.0	4.0	0.0			-15	-83	-22	-51	-9	-1	+17	0		-1490	18985
Post-aero Correction	310	3	57.0	4.0	228.0	-1302	-24	-12	-22	0	-52	-25	-54	-45	0	-1300	-1861	17124
Drop Residuals, Cargo, Crew						-349									0			
EOM Mass	5871	18	2112		4001	0	0	0	0	0	0	0	0	0	0	0	0	17124

LUNAR EXCURSION SEQUENCE

Sequence	Delta V MPS	Delta V RCS	Time Hrs	Pwr kW	Pwr kW-hr	MPS nom	MPS Losses	RCS	Pwr Fluids	Food	Water	O2	N2	Metabolic	Waste Fluids	Inert Mass	Delta Mass	Total Mass
Begin Lunar Descent						51796		273	871	76	232	64	80	27	1800	32842	87860	87860
Corrections		5	0.0	5.0	0.0	0	0	-109	0	0	0	0	0	0	0	-109	87751
Lunar Descent	1920		0.2	6.0	1.2	-29341	-21							+0	0	-29362	58389
Lunar Surface Operations																		58389
Lunar Ops - Crew in Cab			48.0	4.0	192.0	0	-30	0	-70	-18	-43	-8	-8	+14	0	-9870	-10025	48364
Lunar Ops - Stage Active			672.0	2.0	1344.0	0	-413	0	-488			-7	-22	-41	0	+500	-471	47893
Lunar Ops - External Support			3600.0	0.0	0.0	0		0	0									47893
Lunar Ops - Drop Tankset			0.0	4.0	0.0	-918		-28	-71	-3	-7	-1	0	+2	0	-4848	-5864	42029
Drop Landing Gear			8.0	4.0	32.0	0	-5	0	-12	0	-7	0	0	+0	0		-25	42003
Lunar Ascent	1822		0.2	4.0	0.8	-13457	-21	0	-3	-1	-2	0	0	+1	0		-13479	28524
Trim, Rendezvous		10	2.0	4.0	8.0	0	0	-71							0		-76	28448
LLO Rendezvous			0.0	4.0	0.0	0	0	0	0	0	0	0	0	0	0	0	0	28448
Transfer Crew, Cargo																		
Burnout Mass	3742	15	4330		1578	7590	0	66	227	53	179	48	57	3	1800	18424	0	28448

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CARGO EXPENDABLE MISSION LUNAR TRANSFER SEQUENCE	Delta V (m/sec) MPS	RCS	Time Hrs	Pwr kW	kW-hr	Fluids (kg) MPS nom	MPS Losses	RCS	Pwr Fluids	ECLSS Consumables Food	Water	O2	N2	Stored Waste Metabolic	Fluids	Inert Mass (kg)	Delta Mass (kg)	Total Mass (kg)
ETO Launch #1						0		0	0	0	0	0	0	0	0	61205	61205	61205
Fll Core Vehicle			0.0	1.0	0.0	+21453	0	0	0	0	0	0	0	0	0	0	+21453	128229
LEO stay prior to Launch #2			0.0	1.0	0.0	+56769	0	0	0	0	0	0	0	0	0	+4848	+61617	189846
LEO stay prior to Launch #3			504.0	1.0	504.0	+67627	-1080	0	0	0	0	0	0	0	0	+4302	+70848	260694
LEO stay prior to Launch #4			504.0	1.0	504.0	+67627	-1999	0	0	0	0	0	0	0	0	+4302	+69929	306623
Departure Preparations			370.0	2.0	740.0		-2133	0	0							0	-2133	328490
Start Trans-lunar Injection						206586		645	1033	0	0	0	0	0	0	74656	282919	282919
Coast		3	4.5	2.0	9.0	0	-13	-211	-3								-227	282692
Trans-lunar Injection	3300		0.2	2.0	0.4	-142247	-22	0	0								-142269	140423
Drop TLI Tankset						-1951		-21	0							-8603	-10576	129847
Trans-lunar Coast		3	72.0	2.0	144.0	0	-63	-97	-52								-212	129635
Mid-Course Correction	95		0.0	2.0	0.0	-2585	-21	0	0								-2606	127030
Coast		3	12.0	2.0	24.0	0	-10	-95	-9								-113	126916
Prior to Lunar Orbit Insertion																		126916
Lunar orbit insertion	1075	0	0.0	2.0	0.0	-25863	-21	0	0								-25884	101032
Orbital Rendezvous	0	0	0.0	2.0	0.0	0	0	0	0							0	0	101032
Drop Stage 2						0		0	0								0	101032
Lunar Orbit Operations																		101032
Corrections		5	0.0	2.0	0.0	0	0	-126	0								-126	100906
Lunar Descent	1920		0.2	2.0	0.4	-33739	-21	0	0								-33760	67146
Lunar Surface Operations																		67146
Lunar Ops			0.0	2.0	0.0	-1055		-32	-6							-52683	-53776	13370
EOM Mass	6390	14	1467		1926	0		0	0	0	0	0	0	0	0	13370		13370

Figure 1-2.2.4-3. *Sequential Mass and Fluid Inventory Lunar Cargo Mission*

Payloads
- ● CNDB Delivered Payload
- ● Return mass, incl return payload and GB Crew Module

Mission Types
- ▦ Unpiloted Delivery
- ▨ Piloted or Servicing with Return
- ▧ Lunar Outpost Support

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 1-2.2.4-4. CNDB Mission Requirements

Flight Element Mass Summary

Flight Element	Inert Mass (kg)	Propellant Mass (kg)
Crew Module	5583	0
- with Protection	7383	0
Aerobrake	4110	0
Core Stage	8520	21450
Descent Tankset (1)	2415	28386
TLI Tankset (1)	4307	66727

Figure 1-2.2.4-5. Space-Based Concept Definition

aerobrake can be used for most non-lunar piloted missions, but descent and TLI tanksets can be added for greater capability.

Cargo delivery capabilities of various configurations of the space-based STV concept are given in Figure 1-2.2.4-6. Also shown are the CNDB mission payloads and ΔV's. Unpiloted 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 captured completely by the core stage. 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.

1-2.2.5   Launch and Recovery

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 droptanks are launched fully loaded aboard three heavy-lift launch vehicles and 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.
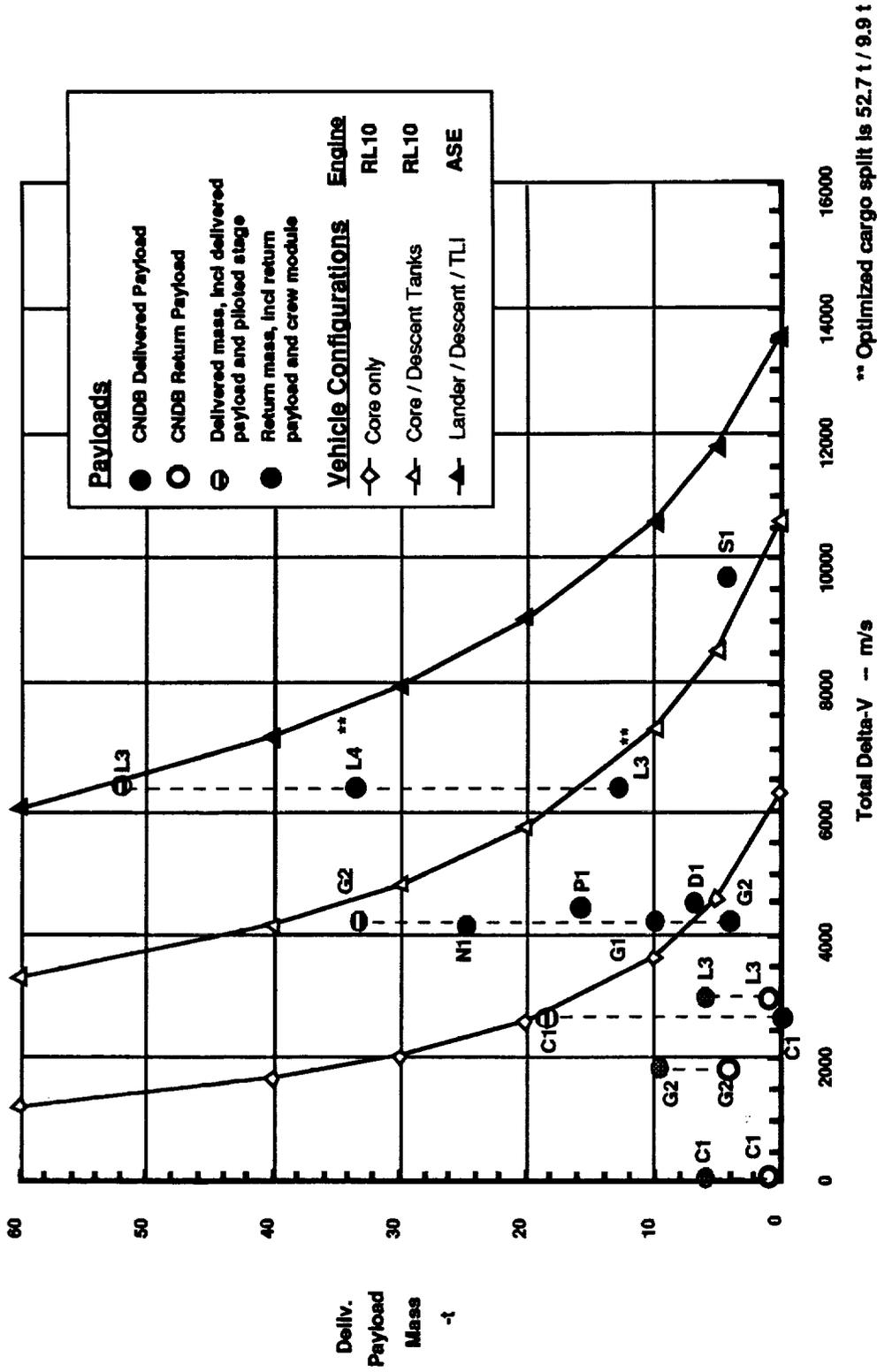
Figure 1-2.2.4-6. Space-Based Vehicle Element Capabilities

CNDB Missions are Captured by Components of a Lunar Mission-designed Vehicle.

1-2.3   GROUND-BASED CONCEPT

This section discusses the configuration of a ground-based STV, including a single-launch concept and a multiple-launch concept. It includes a top-level description of the core stage, crew module, delivery segment, and droptank sets and gives mass properties, performance, launch and recovery operations, and the use of lunar-designed flight elements for capture of other non-lunar missions.

1-2.3.1   System Design and Operation

A few of the issues addressed by the current ground-based concepts include the following:

1. Minimization of on-orbit assembly.
2. Two engine-out operation capability.
3. Crew launch-escape capability in the case of an on-pad emergency.
4. Visibility of lunar landing pads and horizon.
5. Payload accessibility.
6. Lunar surface crew access.
7. Lunar surface staging (i.e., liftoff from a stable platform).
8. Capture of non-lunar CNDB missions.

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 droptank sets, a pair of expendable delivery stages, and an expendable lunar lander platform. Figure 1-2.3.1-1 shows a single-launch concept in which all flight elements are launched full in a single HLLV launch. Figure 1-2.3.1-2 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.

The on-orbit operations of the multiple-launch vehicle are depicted in Figure 1-2.3.1-3. The LO2 tanker is launched initially and remains on-orbit until the core

BOEING
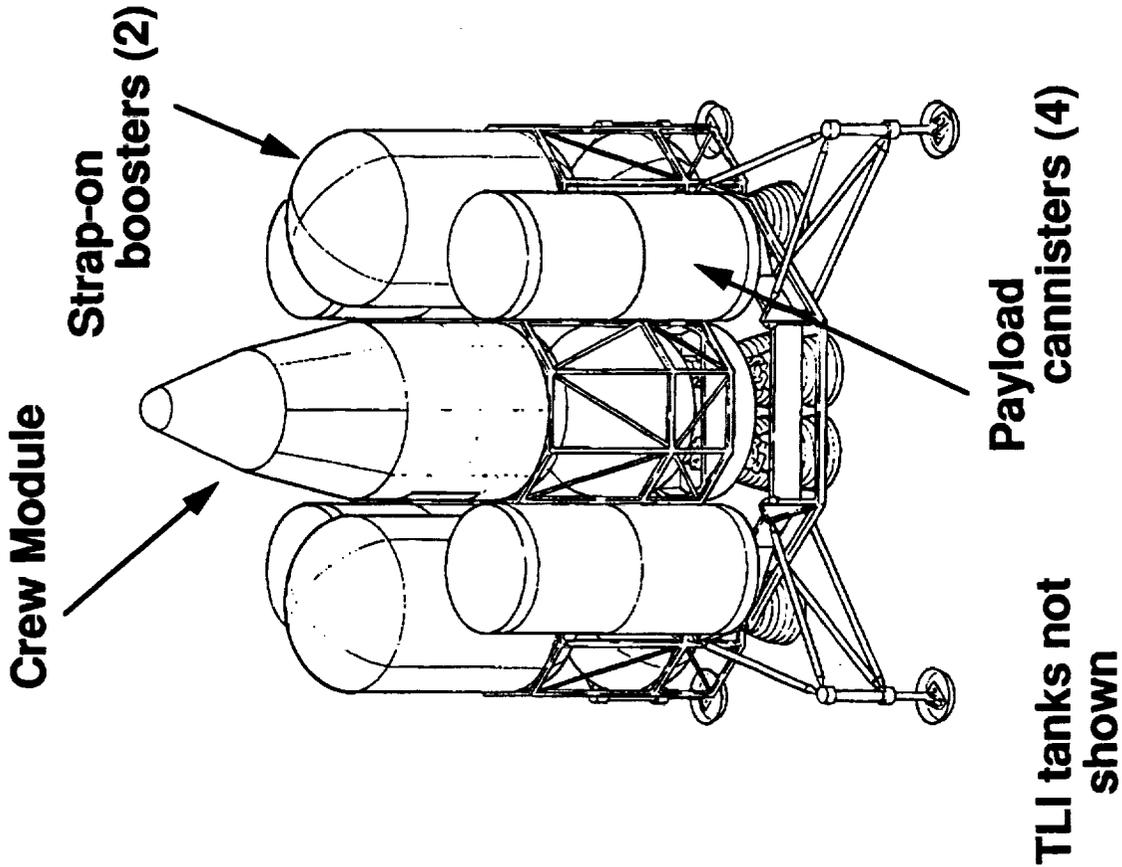
Features

- 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

Crew Module

Strap-on boosters (2)

Payload cannisters (4)

TLI tanks not shown

Figure 1-2.3.1-1. Ground-Based Single Launch Vehicle
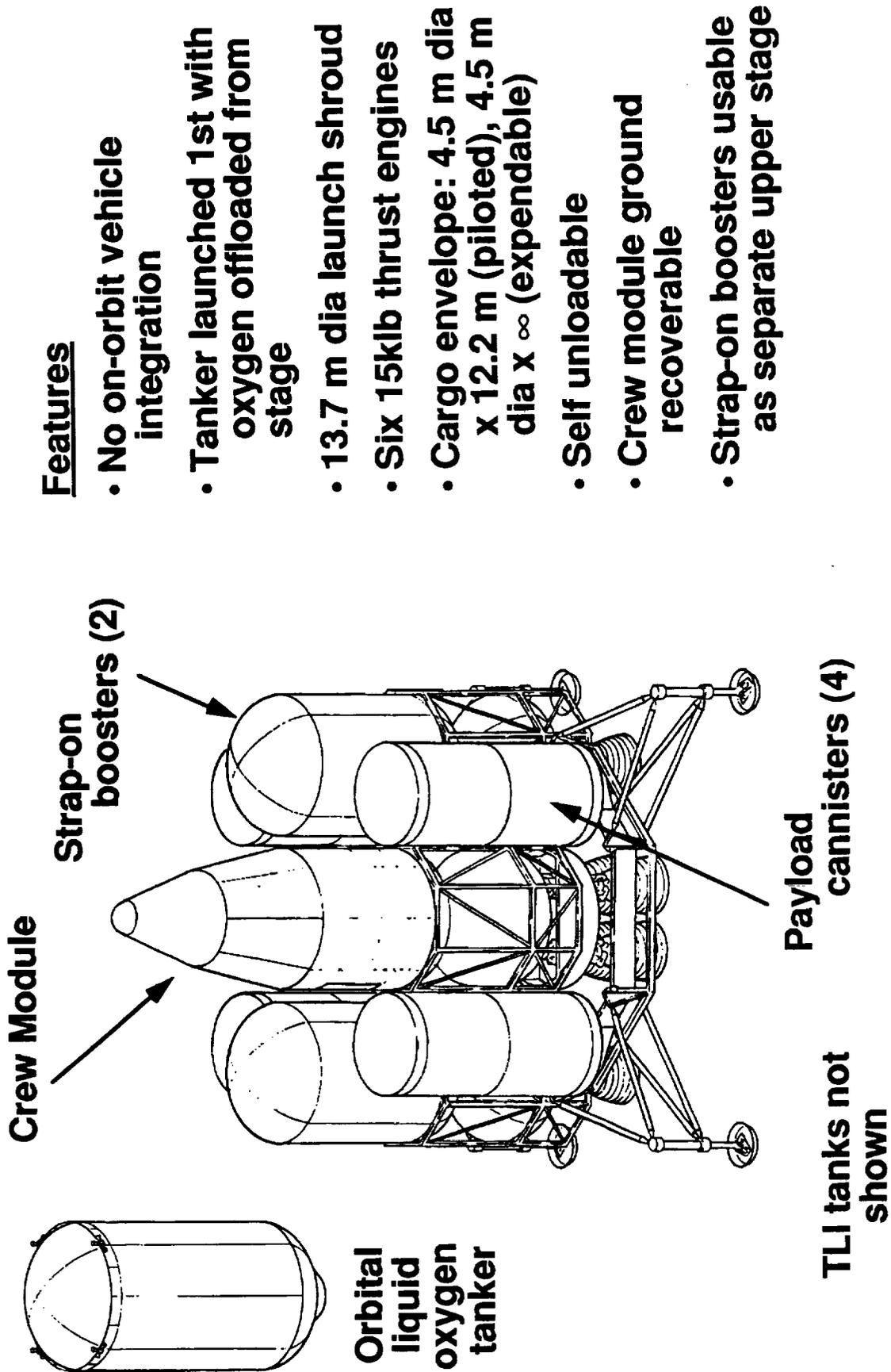
Features

- 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

Crew Module

Strap-on boosters (2)

Payload cannisters (4)

Orbital liquid oxygen tanker

TLI tanks not shown

Figure 1-2.3.1-2. Ground-Based Multiple Launch Vehicle
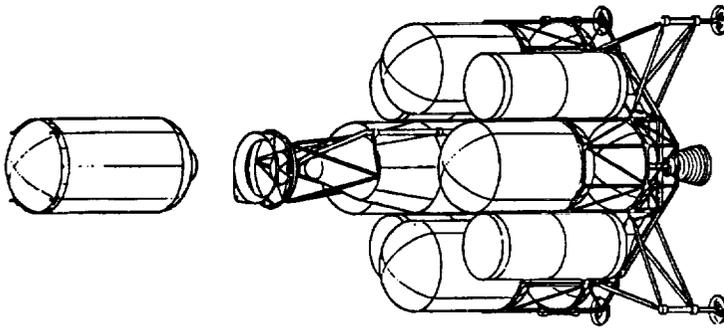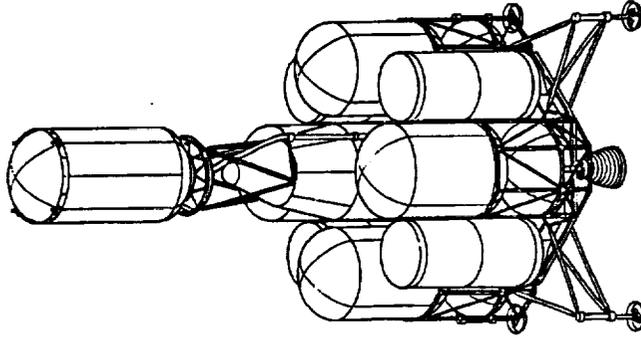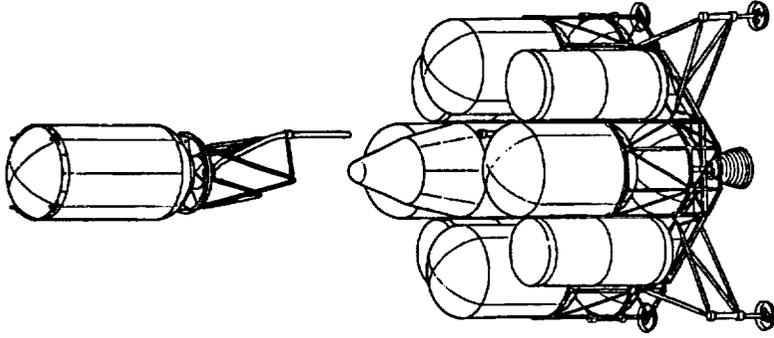
- **Perform settling burn**
- **Transfer liquid oxygen**

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

Figure 1-2.3.1-3. Ground-Based Orbital Refueling Operation

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, and then jettisons the tanker, escape structure, and 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 1-2.3.1-4. The TLI tanks are dropped after the TLI burn and the vehicle descends to the lunar surface following lunar injection with the 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 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 1-2.3.1-5, and moved to the base, either with built-in provisions or using a lunar flatbed trailer. At the end of the lunar stay, the crew loads return cargo, boards using a hoist, and 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 1-2.3.1-6, 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 the avionics pallet is left on the lunar surface with the lander and delivery stages.

1-2.3.2   Subsystem Overview

The ground-based vehicle includes the following subsystems, as shown in Figure 1-2.3.2-1:

1. Structures and Mechanisms - Includes a core stage with external load-bearing body structure, a lunar lander with landing gear, a pressurized crew module with an external aerodynamic shell, two sets of TLI droptanks, and two sets of delivery dropstages.
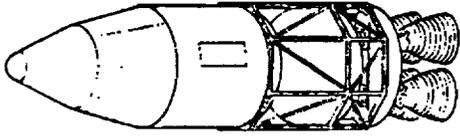
Lunar ascent

Vehicle core, descent stages discarded
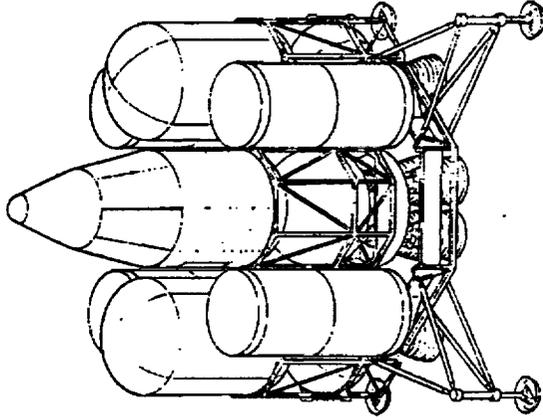
Lunar arrival

TLI tanks discarded
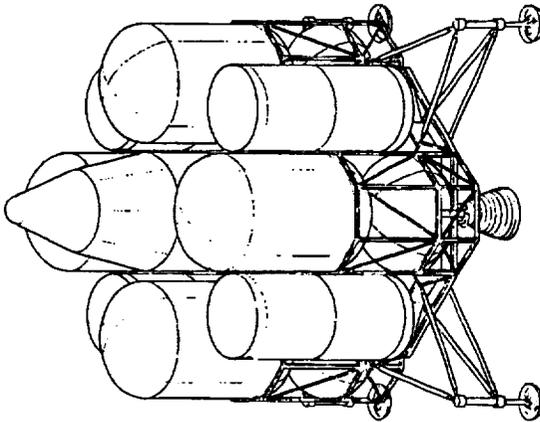
Mission start

Fully assembled configuration

Figure 1-2.3.1-4. Ground-Based Vehicle Configuration Sequence
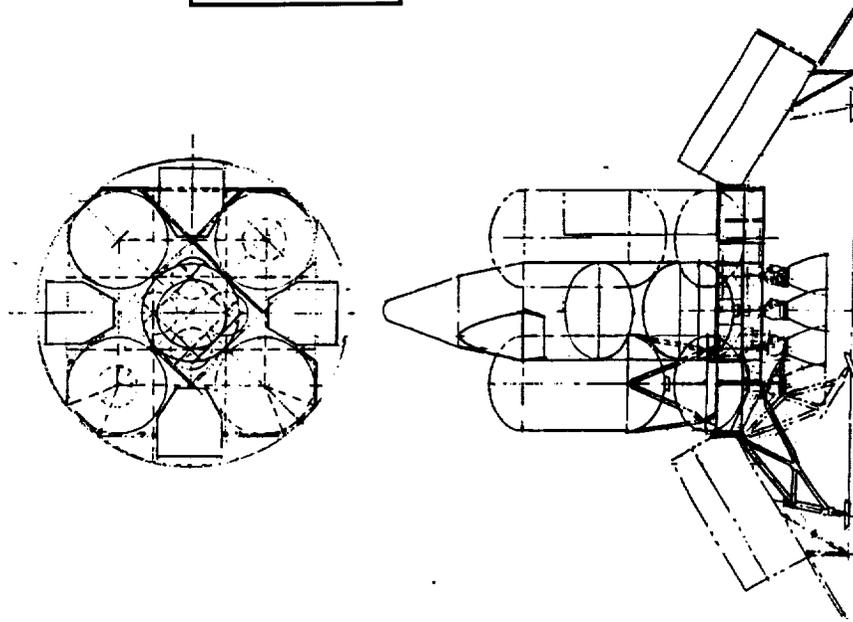
- Self unloading
- Cargo split into four pieces
- Ramps integral to cargo pallet
- Relatively low c.g. for landing stability

Figure 1-2.3.1-5. Ground-Based Payload Unloading
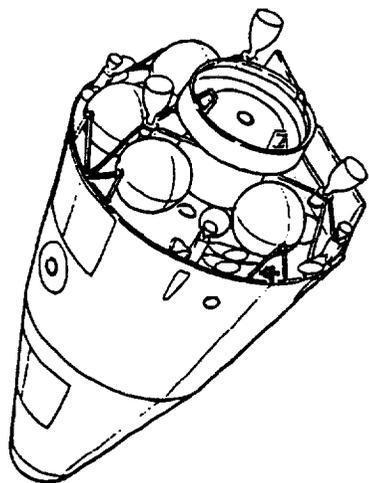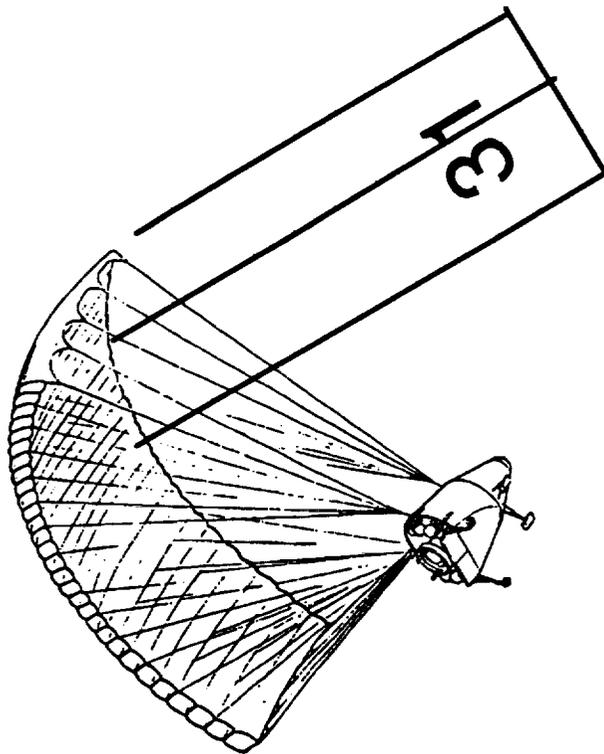
Scarfed biconic shape

L/D > .8

Parafoil recovery

Refurbishable

Independent RCS

Capable of surviving water ditch

Figure 1-2.3.1-6. Ground Recovery Crew Module

BOEING

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 1-2.3.2-1. Ground-Based Vehicle Subsystem Overview

203

D180-32040-2

2. Main Tankage - The core tankset and each droptank set has a single LO2 tank and a single LH2 tank with associated slosh baffles and propellant acquisition devices. 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 1-2.3.2-2 and a mass statement is given in Figure 1-2.3.2-3.

3. Protection - Includes thermal control and damage protection of the main cryogenic tanks, thermal control of avionics and power equipment, thermal and radiation protection of the crew during long-duration exposure in space, and thermal protection of the crew module for the reentry maneuver.

4. Main Propulsion - Consists of a total of six advanced expander-cycle engines with electromechanical actuation and propellant delivery, pressurization, fill, and vent systems.

5. Reaction Control - Includes four GO2/GH2 thruster modules on the delivery stages and four on the crew module, with associated accumulators, pressurization, and control.

6. Electrical Power - 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.

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

8. Communication and Data Handling - Provisions for communication, vehicle health maintenance, and data handling, with audio and video interfaces for piloted operations and instrumentation for droptank monitoring and control.

9. Displays and Controls - Provisions on the crew module for limited crew control and status monitoring of the vehicle during critical phases of the mission.

10. Environmental Control - Provisions on the crew module for atmosphere supply and control, internal equipment cooling, and metabolic and equipment heat rejection.

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

Structure
Al-Li domes
and barrel
sections

Avionics
Redundant gyros
Horizon sensor
VHMS sensors
Processor

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

Reaction Control Options
Hydrazine thrusters
Cold gas thrusters
Resistojets
Torque rods
Momentum wheels

Figure 1-2.3.2-2. LOX Tanker Description

ITEM	QTY	MASS (KG)	REMARKS (ENGLISH UNITS)
STRUCTURES AND MECHANISMS		232	
AFT INTERFACE STRUCTURE		162	
SPACECRAFT INTERFACE RING	1	54	L=44 ft, A=2.0 in2 + 10% ALUMINUM
LAUNCH VEHICLE INTERFACE RING	1	54	L=44 ft, A=2.0 in2 + 10% ALUMINUM
EQUIPMENT SUPPORT TRUSS	2	20	ESTIMATE
SOLAR PANEL SUPPORT STRUCTURE	84	20	INSULATED STAND-OFF PADS
UMBILICAL PLATE	2	14	
DOCKING / SEPARATION SYSTEMS		70	
SPACECRAFT BERTHING MECHANISM	1	50	ESTIMATE
RETENTION FITTINGS	4	20	
TANKAGE - MAIN		1439	
LO2 TANK		1439	
FWD DOME	1	129	S=233 SF, t=0.07 in +30% weldlands 2090 Alum-Lithium
FWD RING	1	54	L=44 ft, A=2.0 in2 +10% 2219 Aluminum
CYLINDER	1	933	S=1019 SF, t=0.116 in +30% weldlands 2090 Alum-Lithium
AFT RINGS	3	161	L=44 ft, A=2.0 in2 +10% 2219 Aluminum
AFT DOME	1	129	S=233 SF, t=0.07 in +30% weldlands 2090 Alum-Lithium
SLOSH BAFFLES / SCREENS	1	24	FOR SLOSH DAMPING 2090 Alum-Lithium
VORTEX BAFFLE, SCREEN	1	9	START BASKET, SUMP 2090 Alum-Lithium
PROPELLANT GAUGING	1	0	INCLUDED IN INSTRUMENTATION
PROTECTION		402	
METEOROID / DEBRIS PROT (END CAP ONLY)		25	S=250 SF, t=0.018 in, Gr/Ep with fiberglas supports (40% of tot)
EXTERNAL THERMAL CONTROL		330	
LO2 TANK INSULATION, MLI		258	S=1514 SF @ 0.375 psf (90 layer, incl instl) Double-Alum Kapton
LO2 TANK INSULATION, FOAM		72	S=1433 SF @ 0.104 psf (t=0.5 in) CPR 488
MLI PURGE AND VENT SYSTEM		15	PURGE DUCT AND AUTOMATIC VENTS - ESTIMATE
THERMAL REJECTION PALLETS	2	32	FOR AVIONICS, BATTERY HEAT REJECTION
PROPULSION - MAIN PROPULSION		143	
LO2 FILL, DRAIN (6 IN)		32	
ISOLATION VALVE	1	10	
DISCONNECT / CONTROL VALVE	1	14	
DUCT ASSEMBLY - 6 IN	1	18	
LO2 TANK SPACE VENT, RELIEF		30	
THERMODYNAMIC VENT SYS	1	20	5 FT @ 8 LB/FT, VACUUM-JACKETED ALUMINUM
DUCT ASSEMBLY, NON-PROP VENT	1	10	
LO2 TANK GROUND VENT, RELIEF		23	
DISCONNECT / VENT VALVE	1	3	SELF CLOSING, RISE-OFF DISCONNECT ALUMINUM
DUCT ASSEMBLY	1	20	
LO2 TANK PRESSURIZATION		45	
DISCONNECTS	1	2	2.0 IN DIAMETER ALUMINUM
VALVES	3	4	SELF CLOSING, RISE-OFF DISCONNECT ELECTROMECHANICAL VALVE
DUCT ASSEMBLY	1	39	
SUPPORT / INSTALLATION	1	13	10 % OF HARDWARE ALUMINUM

Figure 1-2.3.2-3. LOX Tanker Elements (Sheet 1 of 2)

BOEING

ITEM	QTY	MASS (KG)	REMARKS (ENGLISH UNITS)
PROPULSION - AUXILIARY			
ATTITUDE CONTROL SYSTEM		**99** / 54	BLOWDOWN HYDRAZINE SYSTEM
THRUSTERS	12	15	
PROPELLANT SUPPLY TANKS	2	27	
SUPPLY MANIFOLD	1	7	
SUPT / INSTL		5	10 % OF HARDWARE
SOLID ROCKET THRUSTERS	2	45	FOR TANK DEORBIT
POWER SOURCE		**199**	
BATTERY	4	33	1-KW BATTERY / NI-H2 Rechargeable
SOLAR ARRAY PANELS		160	240 SF EACH - FIXED PANELS
SUPPORT / INSTALLATION		6	
WIRING & ELECTRICAL INTERFACE		35	
POWER DISTRIBUTION ELECTRONICS		10	
WIRING		20	
WIRING SUPPORT / INSTL		5	
GUIDANCE, NAVIGATION, AND CONTROL		**100**	
GUIDANCE AND NAVIGATION		59	
IMU	2	44	
HORIZON SENSOR	2	5	
DOCKING SENSOR	2	10	
VALVE CONTROL	2	18	
LASER INITIATORS	2	14	
SUPPORT / INSTALLATION		9	10 % OF HARDWARE
COMM AND DATA HANDLING		**98**	
COMMUNICATION AND TRACKING		47	
TRANSPONDER	2	24	
CMD DESTRUCT EQUIPMENT	2	6	
DIPLEXERS	1	2	
POWER AMP	2	12	
RF SWITCH	3	1	
ANTENNAS	2	2	
DATA HANDLING			
FAULT TOLERANT PROCESSOR	1	18	
HEALTH MANAGEMENT			
MAINTENANCE SENSORS	5	5	
INSTRUMENTATION		19	
PROPELLANT GAUGES / ELEX	1	10	
SENSOR INTERFACE UNIT (SIU)	4	2	
NETWORK INTERFACE UNIT (NIU)	2	2	
SENSORS	25	5	
SUPPORT / INSTALLATION		9	10 % OF HARDWARE

Figure 1-2.3.2-3. *LOX Tanker Elements (Sheet 2 of 2)*

1-2.3.3   Mass Properties

Mass summaries for the ground-based STV concept are given in Figures 1-2.3.3-1, 1-2.3.3-2, and 1-2.3.3-3 for the piloted lunar, unpiloted lunar, and unpiloted GEO delivery missions, respectively. A weight growth margin of 15% 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.

The current ground-based vehicle concept can either deliver 11,630 kg of 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 tons of cargo is delivered to the lunar surface over 21 piloted and 4 cargo-only missions. 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.

Summary and sequential mass properties for the ground-based lunar piloted mission are shown in Figure 1-2.3.3-4. The longitudinal center of mass ranges from 2.94m from the engine gimbal plane at startburn to 8.0m from the engine gimbal plane prior to trans-Earth injection. At lunar landing, the cg is about 8.44m from the landing pad plane.

1-2.3.4   Performance

The selected ground-based STV main propulsion system is a LO2/LH2 system and uses advanced engines with a vacuum thrust of 15,000 lb per engine and an assumed specific impulse of 481 seconds. The reaction control system is a gaseous O2/gaseous H2 system with an assumed specific impulse of 410 seconds. The selected vehicle concept is designed to satisfy the piloted and unpiloted lunar missions, with flight elements capable of capturing other non-lunar missions given in the previous section.

Lunar Mission Performance. A mission timeline for the piloted lunar mission is given in Figure 1-2.3.4-1 and sequential mass and fluid inventories for the lunar piloted and lunar cargo-only missions are given in Figures 1-2.3.4-2 and 1-2.3.4-3, respectively. Included are main and auxiliary propulsion fluids,

Top View / Side View

STV Mass Summary
Ground-based Vehicle — Lunar Piloted Mission

All mass in kg

| | Core Vehicle | | | | | | | Drop-tanks | | Delivered |
| | TEI Segment | | | Prop Mod | Delivery Segment | | | TLI Tnkst #1 | TLI Tnkst #2 | Cargo |
	Crew Mod	Av. Pallet	Tankset		Lander	D Stg #1	D Stg #2			
Structure and Mechanisms	3341	155	505	483	1206	502	502	433	433	
Structures & Mechs - Landing gear	281	-	-	-	741	-	-	-	-	
Tankage - Main	-	116	385	-	-	659	659	659	659	
Protection	1315	-	378	917	82	570	570	570	570	
Propulsion - Main	-	-	376	-	828	545	545	380	380	
Propulsion - Reaction Control	162	-	-	56	-	190	190	-	-	
Power Source	-	374	-	-	-	-	-	-	-	
Wiring & Electrical Interface	272	265	28	-	78	39	39	28	28	
Guidance, Navigation & Control	130	464	-	21	19	-	-	-	-	
Communication & Data Handling	189	391	37	-	-	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
Segment subtotal			34980			101637			96780	
Total						245026				

Figure 1-2.3.3-1. Ground-Based STV Mass Summary - Lunar Piloted

Top View

Side View

STV Mass Summary Ground-based Vehicle	Lunar Cargo Mission - Unmanned							All mass in kg
	Core Vehicle					Drop-tanks		Delivered
	TEI Segment		Delivery Segment					
	Av. Pallet	Prop Mod	Lander	D Stg #1	D Stg #2	TLI Tnkst #1	TLI Tnkst #2	Cargo
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	21	19	37	37	-	-	
Communication & Data Handling	391	-	-	-	-	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	43443
	3820		101637			96780		
			245680					
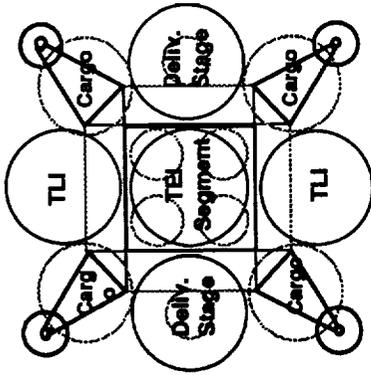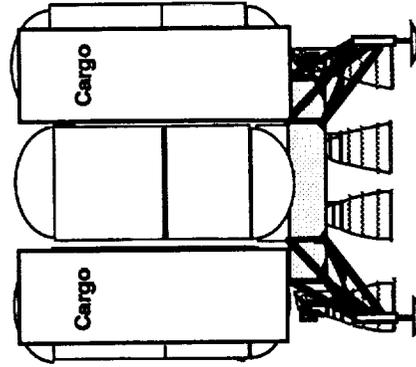
Figure 1-2.3.3-2. Ground-Based STV Mass Summary - Lunar Cargo

BOEING

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

	Core Vehicle Delivery Segment		Delivered
	Av. Pallet	D Stg #1	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	
	1438	48902	24000
		50340	
		74340	
Total LEO-Assembled Mass			24000
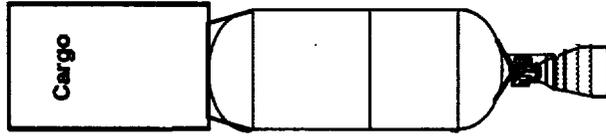
Side View

Figure 1-2.3.3-3. Ground-Based STV Mass Summary - GEO Delivery

	ITEM	Mass - kg		CENTER OF MASS - m			MOMENTS (KG-M^2)		
		W3	W2	Xcg	Ycg	Zcg	IXX	IYY	IZZ
Mass Properties Summary									
CARGO			11640	2.29	0.00	0.00	2.324E+05	1.838E+05	8.304E+04
	CARGO 1	5820		2.29	2.21	3.68	8.949E+03	1.310E+04	1.310E+04
	CARGO 2	5820		2.29	-2.21	-3.68	8.949E+03	1.310E+04	1.310E+04
CREW MODULE			11533	11.68	0.00	0.00	2.008E+04	2.008E+04	1.592E+04
	BALLISTIC RETURN CAB	8394		11.68	0.00	0.00	1.789E+04	1.789E+04	1.149E+04
	EQUIPMENT PALLET	2031		11.68	0.00	0.00	1.911E+03	1.911E+03	4.153E+03
	CREW	400		11.68	0.00	0.00	1.000E+02	1.000E+02	1.000E+02
	EVA SUITS	400		11.68	0.00	0.00	1.000E+02	1.000E+02	1.000E+02
	CREW CONSUMABLES - total	308		11.68	0.00	0.00	7.700E+01	7.700E+01	7.700E+01
TANKSET- ASCENT			19834	3.11	0.00	0.00	6.257E+03	4.598E+04	4.598E+04
	STAGE INERT	2217		3.66	0.00	0.00	6.257E+03	1.409E+04	1.409E+04
	PROPELLANT	17617		3.04	0.00	0.00	0.000E+00	3.115E+04	3.115E+04
ASCENT P/A MODULE			1791	-0.01	0.00	0.00	3.721E+03	3.412E+03	3.412E+03
	STAGE INERT	1791		-0.01	0.00	0.00	3.721E+03	3.412E+03	3.412E+03
LANDER			3686	-1.10	0.00	0.00	7.612E+04	4.966E+04	4.966E+04
	LANDER STRUCTURE	2926		-1.00	0.00	0.00	4.225E+04	3.186E+04	3.186E+04
	LANDING GEAR (DEPLOYED)	760		-1.50	0.00	0.00	3.387E+04	1.765E+04	1.765E+04
DROP TANK SET-TLI (2 SETS)			98394	2.58	0.00	0.00	1.759E+06	1.779E+06	8.528E+05
	TANKSET INERT	3150		3.70	2.03	-3.68	9.901E+03	2.894E+04	2.894E+04
	PROPELLANT	46047		2.50	2.03	-3.68	-1.164E-10	1.900E+05	1.900E+05
	TANKSET INERT	3150		3.70	-2.03	3.68	9.901E+03	2.894E+04	2.894E+04
	PROPELLANT	46047		2.50	-2.03	3.68	-1.164E-10	1.900E+05	1.900E+05
STAGE - Descent (2 SETS)			99766	2.52	0.00	0.00	2.105E+06	4.661E+05	2.550E+06
	TANKSET INERT	3877		2.82	4.57	0.00	1.063E+04	4.285E+04	4.285E+04
	PROPELLANT	46006		2.50	4.57	0.00	0.000E+00	1.899E+05	1.899E+05
	TANKSET INERT	3877		2.82	-4.57	0.00	1.063E+04	4.285E+04	4.285E+04
	PROPELLANT	46006		2.50	-4.57	0.00	0.000E+00	1.899E+05	1.899E+05
TOTAL AT LEO ASSEMBLY			246644	2.94	0.00	0.00	4.202E+06	3.541E+06	4.593E+06

SEQUENCED MASS DATA									
LEO ASSEMBLY			246644	2.94	0.00	0.00	4.202E+06	3.541E+06	4.593E+06
START TRANS-LUNAR INJECTION			247405	3.01	0.00	0.00	4.183E+06	3.674E+06	4.738E+06
PRIOR TO LUNAR ORBIT INSERTION			112122	3.54	0.00	0.00	1.644E+06	1.673E+06	2.857E+06
LLO OPERATIONS			89252	3.80	0.00	0.00	1.162E+06	1.536E+06	2.242E+06
LUNAR LANDING			59313	4.44	0.00	0.00	5.402E+05	1.325E+06	1.416E+06
BEGIN LUNAR ASCENT			33989	6.42	0.00	0.00	3.350E+04	7.268E+05	7.218E+05
START TRANS EARTH INJECTION			23006	7.99	0.00	0.00	3.004E+04	5.240E+05	5.191E+05
START REENTRY			11854	11.68	0.00	0.00	2.064E+04	2.067E+04	1.639E+04

Figure 1-2.3.3-4. Summary Ground-Based Mass Properties

Event	Event Duration (Hours)	Mission E.T. (Hours)	Mission E.T. (Days)	Sequenced Mass (kg)	ΔV (m/s)
Launch #1 - Core, crew mod., and aerobrake	0.0	0.0	0.0	16791	
Storage at SSF (21 days between launches)	504.0	504.0	21.0	16791	
Launch #2 - LD droptanks, and PSS cargo	0.0	504.0	21.0	88264	
Storage at SSF	504.0	1008.0	42.0	87921	
Launch #3 - First TLI tankset	0.0	1008.0	42.0	158955	
Storage at SSF	504.0	1512.0	63.0	156500	
Launch #4 - Second TLI tankset	0.0	1512.0	63.0	227534	
Final vehicle integration	72.0	1584.0	66.0	227534	
Add protective water	0.0	1584.0	66.0	229334	
Fill core vehicle propellant tanks	4.0	1588.0	66.2	251119	
Flight readiness verification	96.0	1684.0	70.2	251119	
Vehicle closeout	192.0	1876.0	78.2	251119	
Launch #5 - Crew, crew consum. (STS)	0.0	1876.0	78.2	246788	
Crew ingress	1.0	1877.0	78.2	247879	
Final vehicle checkout	4.0	1881.0	78.4	247879	
Separate from LEO node	0.0	1881.0	78.4	247879	
Departure prox ops	4.3	1885.3	78.6	247879	
Wait for TLI node (up to one revolution)	1.5	1886.8	78.6	247879	
Trans Lunar injection (TLI) burn	0.2	1887.0	78.6	120761	3300
Stage TLI droptanks	0.0	1887.0	78.6	110585	
TCM to "free return" trajectory	0.1	1887.1	78.6	110585	
Lunar transit	72.0	1959.1	81.6	110585	
TCM (target to landing site)	0.0	1959.1	81.6	110585	95
Lunar approach	12.0	1971.1	82.1	110585	
First lunar descent burn	0.2	1971.3	82.1	87085	1075
Low Lunar orbit coast (up to one revolution)	2.0	1973.3	82.2	87085	
Lunar landing burn	0.2	1973.5	82.2	54467	1920
Crew remains in transit module	48.0	2021.5	84.2	54467	
Crew transfer to habitat module	0.0	2021.5	84.2	53667	
Unload cargo	0.0	2021.5	84.2	43797	
Lander on surface with no surface support	672.0	2693.5	112.2	43497	
Lander activation, crew ingress with return P/L	4.0	2697.5	112.4	44797	
Transfer LD droptanks residuals to core tanks	1.0	2698.5	112.4	44797	
Drop descent tanks	0.0	2698.5	112.4	39967	
Lunar ascent burn	0.2	2698.7	112.4	26585	1822
Low Lunar orbit coast	2.0	2700.7	112.5	26585	
Second Lunar ascent burn	0.2	2700.9	112.5	19894	1075
Earth transit	72.0	2772.9	115.5	19894	16
TCM	0.0	2772.9	115.5	19894	
Earth approach	12.0	2784.9	116.0	19894	
Dump protective water	0.1	2785.0	116.0	18094	
Final TCM	0.1	2785.1	116.0	18094	
Aeroassist maneuver	0.1	2785.2	116.0	18094	
Coast	0.8	2786.0	116.1	18094	
Earth orbit circularization burn	0.1	2786.1	116.1	16821	310
LEO node rendezvous orbital maneuvers	48.0	2834.1	118.1	16821	
LEO node arrival prox ops	4.3	2838.4	118.3	16821	
Vehicle closeout	1.0	2839.4	118.3	16821	
Crew egress	1.0	2840.4	118.3	15730	
Remove Lunar payload	1	2841.4	118.4	15230	

E.T. - Elapsed Time
TCM - Trajectory Correction Maneuver

Figure 1-2.3.4-1. Ground-Based Nominal Timeline

Figure 1-2.3.4-2. Sequential Mass and Fluid Inventory Lunar-Piloted Mission

LUNAR TRANSFER SEQUENCE	Delta V (m/sec) MPS	RCS	Time Hrs	Pwr KW	KW-hr	Fluids (kg) MPS nom	MPS Losses	RCS	Pwr Fluids	ECLSS Consumables Food	Water	O2	N2	Stored Waste Metabolic	Fluids	Inert Mass (kg)	Delta Mass (kg)	Total Mass (kg)
LEO Mass (when applicable)						0			0	0	0	0	0	0	0	0	0	0
Launch #1			0.0	1.0	0.0	+90883	0	0	0	0	0	0	0	0	0	+5907	+96790	96790
LEO stay prior to Launch #2			0.0	1.0	0.0	+90900	0	0	0	0	0	0	0	0	0	+58037	+148937	245727
Departure Preparations			180.0	2.0	360.0	0	-897	0	-131								-1027	244700
Start Trans-lunar Injection						180756		0	0	0	0	0	0	0	0	63944		244700
Coast		3	4.5	2.0	9.0	0	-11	-182	-3								-197	244503
Trans-lunar Inject (TLI tanks)	2111		0.2	2.0	0.4	-88210	-21	0	0								-88232	156271
Trans-lunar Inject (Des tanks)	1189		0.0	2.0	0.0	-34816		0	0								-34816	121455
Drop TLI Stage, Tankset						-1389		-18	-27							-5907	-7341	114114
Trans-lunar Coast		3	72.0	2.0	144.0	0	-55	-85	-52								-193	113922
Mid-Course Correction	95		0.0	2.0	0.0	-2271	-21	0	0								-2292	111629
Coast		3	15.0	2.0	30.0	0	-11	-83	-11								-105	111524
Prior to Lunar Orbit Insertion																		111524
Lunar orbit insertion	1075		0.2	2.0	0.4	-22727	-21	0	0	0	0	0	0	0	0		-22748	88776
Orbital Rendezvous	0	0	0.0	2.0	0.0	0	0	0	0	0	0	0	0	0	0		0	88776
Drop Stage 2		0				0		0	0	0	0	0	0	0	0	0	0	88776
Lunar Orbit Operations																		88776
Corrections		5	0.0	2.0	0.0	0	0	-110	0	0	0	0	0	0	0		-110	88666
Lunar Descent	1920		0.2	2.0	0.4	-29646	-21	0	0								-29668	58998
Lunar Surface Operations																		58998
Lunar Ops			0.0	2.0	0.0	-927		-28	-6							-43443	-44404	14594
EOM Mass	6390	14	272		544	0		0	-6	0	0	0	0	0	0	14595		14594

Figure 1-2.3.4-3. Sequential Mass and Fluid Inventory Lunar Cargo Mission

non-propulsive consumables, waste fluids, and sequential time and power levels.

Evolutionary Mission Capture. For capture of non-lunar unpiloted missions, the delivery stage portion of the lunar vehicle can be used as an independent vehicle, as shown in Figure 1-2.3.4-4. Prior to advanced engine development, the delivery stage with an avionics/power pallet and RL10 engine can be used. The CNDB missions analyzed, including required cargo and ΔV, are given in Figure 1-2.3.4-5. A configuration and mass summary of the current ground-based lunar vehicle flight elements used for capture of the CNDB missions is shown in Figure 1-2.3.4-6. For non-lunar piloted missions, the ascent stage with crew module and the avionics/power pallet can be used. For greater capability, an ascent stage with the avionics/power pallet and two delivery stages can be integrated onto a lander platform.

Cargo delivery capabilities of various configurations of the ground-based STV concept are given in Figure 1-2.3.4-7. Also shown are the CNDB mission payloads and ΔV's. Unpiloted delivery missions are shown as single points on the chart and are captured by a single delivery stage with RL10 except for the lunar cargo delivery mission, which requires the lunar vehicle with advanced engines. 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 the lander platform is required. To deliver the lunar return stage and lunar cargo, the full lunar vehicle is required.

1-2.3.5   Launch and Recovery

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

- 25 mt P/L capability to GEO
- 45 mt propellant mass
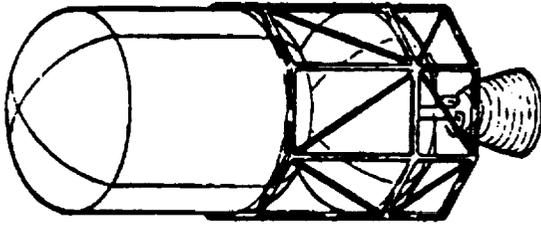
Single engine, single tankset vehicle: add avionics, P/L interface
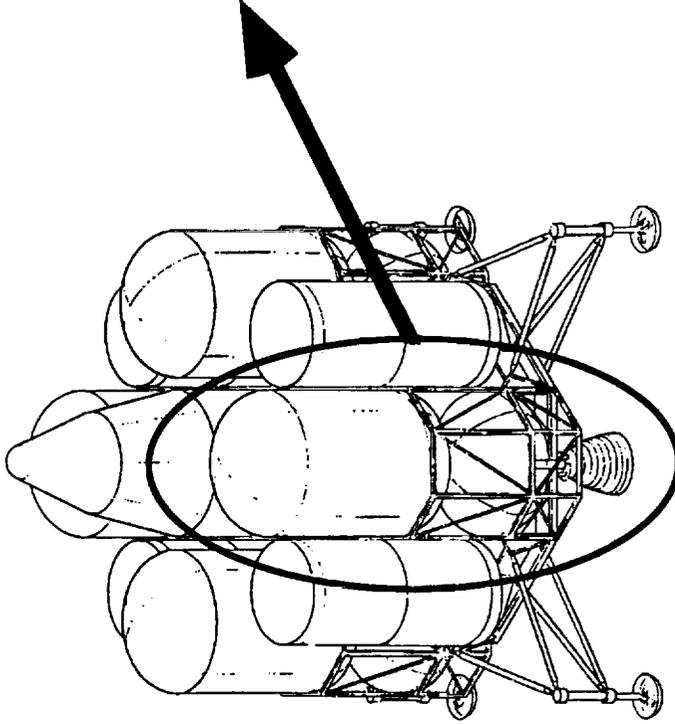
Descent "strap-on" (2)
(tank, plumbing, engine)

CNDB Missions are captured with little modification to descent strap-on

Figure 1-2.3.4-4.   Use of Lunar Elements for CNDB Missions

BOEING

Payloads
- ⊗ CNDB Delivered Payload
- ● Return mass, incl return payload and GB Crew Module

Mission Types
- ▦ Unpiloted Delivery
- ⊞ Piloted or Servicing with Return
- ▨ Lunar Outpost Support

Mission	Delta V (m/s) Delivery	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 1-2.3.4-5. CNDB Mission Requirements

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D180-32040-2

Flight Element Mass Summary

Flight Element	Inert Mass (kg)	Propellant Mass (kg)
Crew Module / Avionics	11474	50
- with Protection	13274	50
Core Tankset	2240	17640
Core P/A Module	1791	0
Delivery Stage (1)	3465	45440
- with Avionics	4908	45440
Lander	3832	0
TLI Tankset (1)	2950	45440

Figure 1-2.3.4-6. Ground-Based Concept Description
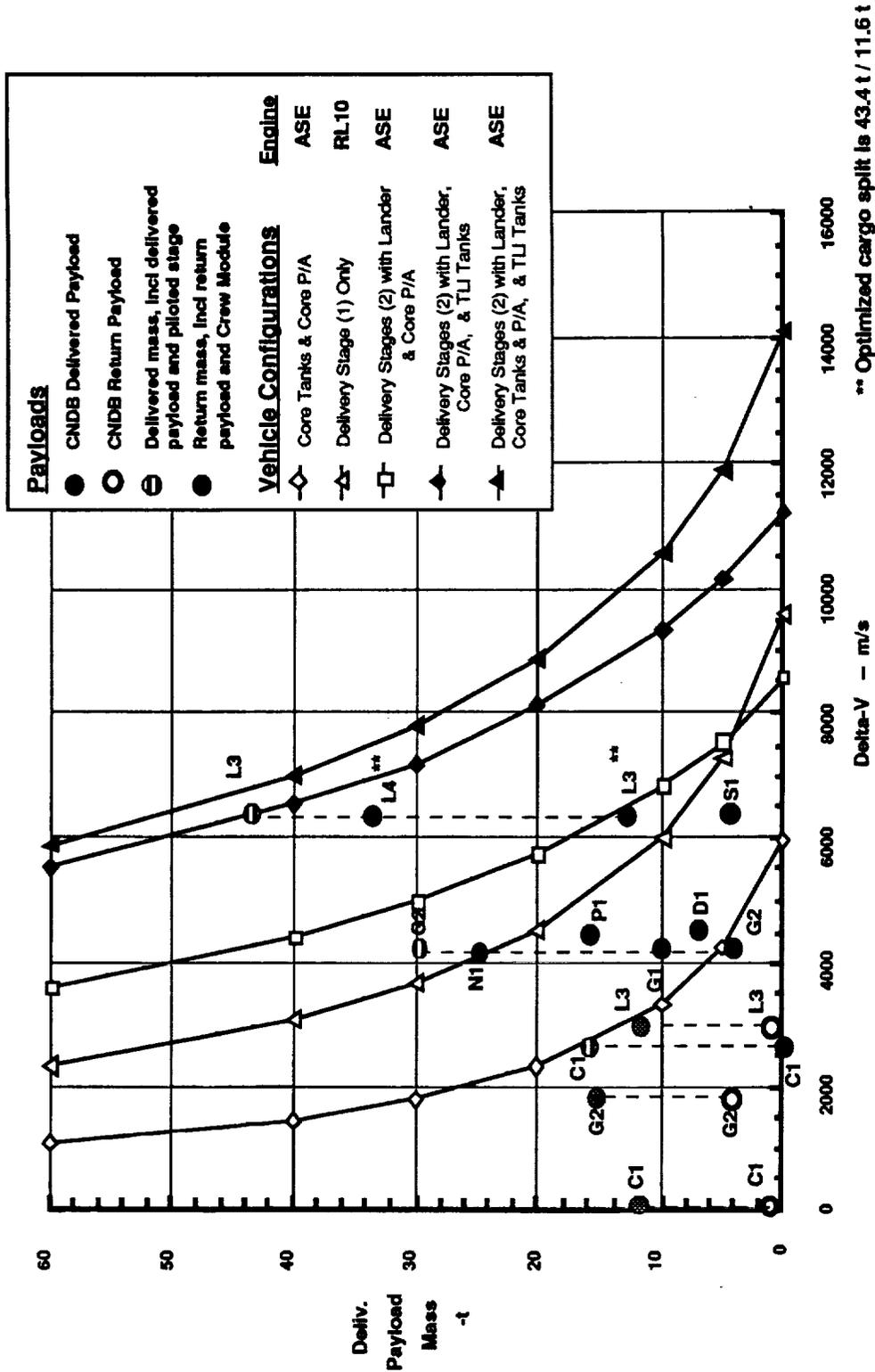
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Figure 1-2.3.4-7.   Ground-Based Vehicle Capabilities

rendezvous). For the latter case, the first launch would include the vehicle with offloaded LO2 tanks and the second launch would include a tanker to fill the vehicle LO2 tanks. In both cases, the only reusable element is the crew module with the avionics pallet, which reenters the Earth's atmosphere and returns to the ground where it is refurbished and reused.

1-2.4   SUBSYSTEM DESCRIPTION

Each subsystem of the space-based and ground-based vehicles is discussed in terms of its key requirements and a description of its hardware and function.

1-2.4.1   Structure and Mechanisms

Requirements. Top-level requirements for the structure subsystem are given in Figure 1-2.4.1-1.

Hardware Description. The major structural elements of the space-based and ground-based STV are as follows:

1. Body structure.
2. Thrust structure.
3. Lunar landing gear.
4. Tank module structures.
5. Aerobrake and aeroshell structure.
6. Crew module structure.

Detail weights of the space-based vehicle structures are given in Figures 1-2.4.1-2, 1-2.4.1-3, and 1-2.4.1-4 for the core vehicle, droptank modules, and crew module, respectively. Weights of the ground-based vehicle structures are given in Figure 1-2.4.1-5 for the ascent segment, Figure 1-2.4.1-6 for the delivery segment, and Figure 1-2.4.1-7 for the crew module.

Body Structure. The space-based core stage has an external load-bearing body structure with structural interfaces to the crew module, aerobrake, cargo, and droptanks. The body structure includes 12 major longerons, a series of interior stabilizing struts, forward and aft stabilizing struts, and exterior closeout panels. The major longerons transfer primary loads and form the backbone of the structure on which the rest of the structure is supported. They are 15.0-foot-long graphite/epoxy beams with beef-up and pads for structural attachments. Eight of the longerons include interface fittings for the landing gear attachment, cargo attachment, and descent droptank attachment. The stabilizing struts are graphite/epoxy struts of varying lengths and sizes with titanium end fittings. The

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		Basing		Requirement Source		
		Space	Ground	Given	Derived	Provis.
Structure and Mechanisms						
Metallic factors of safety		x	x	x		
1.1 on Yield						
1.5 on Ultimate						
Non-metallic Factor of safety		x	x	x		
1.5 on ultimate for continous structure						
2.0 on ultimate for discontinuities						
Windows, doors, hatches		x	x	x		
2.00 x MEOP Proof pressure						
3.00 x MEOP Ultimate pressure						
Personnel compartment		x	x	x		
1.50 x MEOP Proof pressure						
1.65 x MEOP Yield pressure						
2.00 x MEOP Ultimate pressure						
Major structure to withstand liftoff loads		x	x		x	
Aerobrake shall be removable for expendable cargo missions		x		x		

Figure 1-2.4.1-1.   Structures and Mechanisms Design Requirements

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Item	Qty	Mass (kg)	Remarks
CORE STAGE STRUCTURES		1935	
FWD INTERFACE STRUCTURE		74	
AEROBRAKE INTERFACE FTGS	6	12	GR/EP STRUCT
TANK MODULE UMBILICAL SUPT TRUSS	2	30	FOR TLI TANK UMBILICALS, L=7 ft; ALUMINUM
TANK MODULE UMBILICAL PLATE	2	14	ALUMINUM
CARGO/CREW MODULE UMBILICAL PLATE	3	18	ALUMINUM
MAIN BODY STRUCTURE		592	
FWD BULKHEAD PRIMARY STRUTS	18	68	L,ave = 9.7 FT, A= 1.0 IN2 + 20% FTGS
AFT BULKHEAD PRIMARY STRUTS	18	68	L,ave = 9.7 FT, A= 1.0 IN2 + 20% FTGS
PRIMARY LONGERONS	6	53	L= 15 FT, A=1.5 IN2 +20% FTGS; GR/EP
SECONDARY LONGERONS	6	35	L= 15 FT, A=1.0 IN2 +20% FTGS; GR/EP
PRIMARY INTERIOR STABILIZING STRUTS	5	56	L=19 FT, A=1.5 IN2 +20% FTGS
EXTERIOR LATERAL PANELS	8	217	S= 75 SF EA. @ 0.80 PSF; GR/EP panels, struts
LOWER CLOSEOUT PANELS	1	95	S= 420 SF TOTAL @ 0.5 PSF; GR/EP panels, struts
SECONDARY STRUCTURES		301	
EQUIPMENT MOUNTING PANELS	2	45	s=50 sf ea @ 1.0 psf; ALUMINUM
LANDING GEAR FTGS	16	16	TITANIUM
TANK MODULE SUPT TRUSS ASSY	2	80	ALUMINUM
TANK MODULE DEPLOY MECHANISM	2	40	Gas actuators
CARGO MODULE SUPT TRUSS ASSY	2	80	ALUMINUM
CARGO MODULE DEPLOY MECHANISM	2	40	Gas actuators
THRUST STRUCTURE		184	
PRIMARY THRUST RING	1	72	14 FT DIA RING, A=4.0 IN2 + 25% BEEFUP; Gr/Ep
PRIMARY THRUST STRUTS	16	51	L,ave = 7 FT, A=1.0 IN2 + 40% FTGS
SECONDARY STABILIZING STRUTS	3	21	L= 15 FT, A=1.0 IN2 + 40% FTGS; Gr/Ep
TANK MODULE UMBILICAL PLATE	2	28	ALUMINUM
ENGINE INTERFACE FITTINGS	6	12	ENGINE, ACTUATOR INTERFACE PADS; aluminum
LANDING GEAR		688	
PRIMARY ATTENUATOR STRUT ASSY	4	138	L= 12 FT ea. gas-filled shock struts; aluminum tube, ftgs
SECONDARY STRUT ASSEMBLY	16	190	L=14 FT ea, A=2.0 in2 +30% ftgs; Gr/Ep tube, Ti ftgs
DEPLOYMENT / ATTENUATION STRUTS	8	280	L=14 FT ea, A=3.0 in2 +50% ftgs, actuator; aluminum tube, ftgs
LANDING PADS	4	60	4-FT DIA PADS, WITH PROBES
DEPLOY/ RETRACT MECHANISMS	6	20	LATCHES, DRIVES, ETC; Gas-filled shock struts
SEPARATION SYSTEMS		96	
TANK MODULE RETENTION FTGS	6	48	ELECTRO-MECHANICAL LATCHES
CARGO MODULE RETENTION FTGS	6	48	ELECTRO-MECHANICAL LATCHES

Figure 1-2.4.1-2. Space-Based Core Stage Structures

Item	Qty	Mass (kg)	Remarks	
CREW MODULE STRUCTURES		1496		
CREW MODULE SUPPORT STRUCUTRE		163		
CORE VEHICLE INTERFACE STRUCTURE	1	74	S= 137 SF @ 1.2 PSF	COMPOSITE
FTGS, CABIN ATTACHMENT	16	16		
CORE VEHICLE UMBILICAL PLATES	2	15		
EXTERNAL EQUIPMENT SUPPORT STR		34	S=50 SF @ 1.5 PSF	
RADIATOR PANEL SUPPORT	2	24		
PRESSURIZED CABIN PRIMARY STR		610		
BULKHEAD, FWD	1	170	S=250 SF @ 1.5 PSF	ALUMINUM SKIN / STRINGER
BULKHEAD, AFT	1	170	S=250 SF @ 1.5 PSF	ALUMINUM
MINOR FRAMES, CABIN	3	112	L, ave= 45 FT , A= 1.5 IN2	ALUMINUM
COVER PANELS, CYLINDER		158	S=205 SF @ 1.7 PSF	ALUMINUM
CABIN SECONDARY STRUCTURE		364		
PARTITIONS	4	180	SLEEP STATIONS, ETC	ALUMINUM
INTERNAL EQUIPMENT SUPT STRUCTURE		34	S=50 SF @ 1.5 PSF	
FLOORING		150	S= 165 SF @ 2.0 PSF	
WINDOWS		72		
LATERAL WINDOWS	4	48	S= 1.0 SF EA @ 27 PSF	
LATERAL WINDOWS, RETAINER	4	24		
DOCKING MECHANISMS		160		
DOCKING ADAPTER MECHANISM		160		
HATCHES		127		
EVA HATCH, STRUCTURE	1	31	40-IN DIA, SHUTTLE-TYPE	
EVA HATCH, MECHANISM		25		
DOCKING HATCH, STRUCTURE	1	31	40-IN DIA, SHUTTLE-TYPE	
DOCKING HATCH, WINDOW & RETAINER	1	15		
DOCKING HATCH, MECHANISM		25		

Figure 1-2.4.1-3. Space-Based Crew Module Structures

Item	Qty	Mass (kg)		Remarks
DESCENT TANKSET STRUCTURES			474	
INTERTANK		336		
BASIC STRUCTURE	1	198		GR/EP Sandwich S= 485 SF @ 0.9 PSF
LONGERONS	2	11		ALUMINUM L= 120 IN, A= 1.0 IN2
FRAME - tank support interface	2	76		ALUMINUM L= 45 FT, A=1.5 IN2
FRAME - LAUNCH VEH INTERFACE	1	51		ALUMINUM L= 45 FT, A=2.0 IN2
CORE STAGE INTERFACE STRUCTURE			94	ESTIMATE
INTERFACE TRUSS	2	80		Gr/Ep struts, Ti ftgs
UMBILICAL PLATE	2	14		
SEPARATION SYSTEMS			44	
RETENTION FITTINGS	4	20		
PUSH-OFF ACTUATORS, FTGS	2	24		
TLI TANKSET STRUCTURES			884	
FWD INTERFACE STRUCTURE			29	
TANK INTERCONNECT TRUSS	1	29		ESTIMATE
INTERTANK			717	
BASIC STRUCTURE	1	423		GR/EP Sandwich S= 1036 SF @ 0.9 PSF
LONGERONS	4	54		ALUMINUM L= 198 IN, A= 1.5 IN2
FRAME - TANK SUPPORT	2	137		ALUMINUM L= 63 FT, A=2.0 IN2
FRAME - LAUNCH VEH INTERFACE	1	103		ALUMINUM L= 63 FT, A=3.0 IN2
AFT INTERFACE STRUCTURE			94	ESTIMATE
AEROBRAKE INTERFACE TRUSS	1	80		Gr/Ep struts, Ti ftgs
UMBILICAL PLATE	2	14		
SEPARATION SYSTEMS			44	
RETENTION FITTINGS	4	20		
PUSH-OFF ACTUATORS, FTGS	2	24		

Figure 1-2.4.1-4.   Space-Based Tankset Structures

ITEM	QTY	Mass (kg)		REMARKS (ENGLISH UNITS)
STRUCT AND MECHS - PALLET			155	
EQUIPMENT SUPPORT STRUCTURE		155		
EQUIPMENT SUPPORT FRAMEWORK	1	87		ALUMINUM
CREW MODULE INTERFACE	6	6		S=320 SF @ 0.6 PSF
CORE VEHICLE INTERFACE	4	32		BOLTED INTERFACE
CREW MODULE UMBILICAL PLATES	2	15		ELECTROMECHANICAL LATCHES
CORE VEHICLE UMBILICAL PLATES	2	15		
STRUCT AND MECHS - ASCENT TANKSET			505	
FWD INTERFACE STRUCTURE		77		
CREW MODULE INTERFACE RING	1	34		ALUMINUM DIA = 13.5 FT, A = 1.5 IN2
SUPPORT STRUTS, FTGS	16	37		GR/EP, TI FTGS L=5.0 FT, A=1.0 IN2 + 40% FTGS
CREW MODULE UMBILICAL PLATE	1	6		ALUMINUM
INTERTANK		205		
LO2 TANK INTERFACE RING	1	34		ALUMINUM DIA = 13.5 FT, A= 1.5 IN2
SUPPORT STRUTS, FTGS	16	126		GR/EP, TI FTGS L=11.5 FT, A=1.5 IN2 + 40% FTGS
LH2 TANK INTERFACE RING	1	45		ALUMINUM DIA = 13.5 FT, A= 2.0 IN2
AFT INTERFACE STRUCTURE		137		
SUPPORT STRUTS, FTGS	16	74		GR/EP, TI FTGS L=5.0 FT, A=2.0 IN2 + 40% FTGS
INTERFACE RING	1	45		ALUMINUM DIA = 13.5 FT, A= 2.0 IN2
P/A MODULE UMBILICAL PLATE	2	18		ALUMINUM
SEPARATION SYSTEMS		86		
CREW MODULE SEPARATION	4	54		ELECTROMECHANICAL LATCHES
DESCENT,TLI TANK MOD SEPARATION	4	32		EXPLOSIVE BOLTS
STRUCT AND MECHS - ASCENT PROP MODULE			483	
INTERFACE STRUCTURE		153		
FWD INTERFACE RING	1	45		ALUMINUM DIA = 13.5 FT, A= 2.0 IN2
TANK MODULE UMBILICAL PLATE	2	14		ALUMINUM
LANDER INTERFACE FTGS	4	80		
LANDER UMBILICAL PLATE	2	14		ALUMINUM
THRUST STRUCTURE		298		
PRIMARY THRUST RING	1	136		ALUMINUM DIA = 13.5 FT, A= 6.0 IN2
AFT RING	1	45		ALUMINUM DIA = 13.5 FT, A= 2.0 IN2
PRIMARY THRUST FTG	4	80		ALUMINUM
SECONDARY STABILIZING STRUTS	8	29		aluminum Lave = 4 FT, A=2.0 IN2 + 40% FTGS
ENGINE INTERFACE FITTINGS	4	8		ENGINE, ACTUATOR INTERFACE PADS
SEPARATION SYSTEMS				
LANDER SEPARATION	4	32		EXPLOSIVE BOLTS

Figure 1-2.4.1-5. Ground-Based Ascent Stage Structures

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ITEM	QTY	Mass (kg)	REMARKS (ENGLISH UNITS)
STRUCT AND MECHANISMS - LANDER		1947	
FWD INTERFACE STRUCTURE		82	
TANK MODULE UMBILICAL PLATE	10	70	ALUMINUM
CARGO MODULE UMBILICAL PLATE	2	12	ALUMINUM
MAIN BODY STRUCTURE		680	
DEEP TRUSS FRAMES	4	326	Gr/Ep; S= 150 sf ea. @ 1.2 psf
UPPER CLOSEOUT PANELS	4	120	GR/EP panels, struts; S= 133 SF EA @ 0.5 PSF
LATERAL CLOSEOUT PANELS	12	114	S= 42 SF EA @ 0.5 PSF
LOWER CLOSEOUT PANELS	4	120	GR/EP panels, struts; S= 133 SF EA @ 0.5 PSF
SECONDARY STRUCTURES		372	
ASCENT STAGE INTERFACE FTGS	4	60	ALUMINUM
LANDING GEAR INTERFACE FTGS	16	32	TITANIUM
TANK MODULE SUPT TRUSS ASSY	4	160	ALUMINUM
CARGO MODULE SUPT TRUSS ASSY	2	80	ALUMINUM
CARGO MODULE DEPLOY MECHANISM	2	40	Gas actuators
LANDING GEAR / support structure		741	
INTERFACE FTGS	16	48	TITANIUM
PRIMARY ATTENUATOR STRUT ASSY	4	110	aluminum tube, ftgs; L= 12 FT ea. with crush-core attenuation
SECONDARY STRUT ASSEMBLY	16	217	Gr/Ep tube, Ti ftgs; L=16 FT ea, A=2.0 in2 +30% ftgs
DEPLOYMENT STRUTS	8	266	aluminum tube, ftgs; L=16 FT ea, A=2.5 in2 +50% ftgs, spring actuator
LANDING PADS	4	60	4-FT DIA PADS, WITH PROBES
DEPLOY MECHANISMS	4	40	Gas-filled shock struts; LATCHES, DRIVES, ETC
SEPARATION SYSTEMS		72	
ASCENT STAGE SEPARATION	4	24	EXPLOSIVE BOLTS
CARGO MODULE SEPARATION	6	48	EXPLOSIVE BOLTS
STRUCT AND MECHS - DELIV STG, TLI TANKS		502	
INTERTANK		269	
LO2 TANK INTERFACE RING	1	57	ALUMINUM; DIA = 13.5 FT, A= 2.5 IN2
SUPPORT STRUTS, FTGS	16	168	GR/EP, TI FTGS; L=11.5 FT, A=2.0 IN2 + 40% FTGS
LH2 TANK INTERFACE RING	1	45	ALUMINUM; DIA = 13.5 FT, A= 2.0 IN2
CORE STAGE INTERFACE STRUCTURE		124	
SUPPORT STRUTS, FTGS	8	37	GR/EP, TI FTGS; L=5.0 FT, A=2.0 IN2 + 40% FTGS
INTERFACE RING	1	23	ALUMINUM; DIA = 13.5 FT, A= 2.0 IN2
LANDER INTERFACE FTGS	3	30	Gr/Ep struts, Ti ftgs; ESTIMATE
ASCENT TANK MODULE FWD I/F ASSY	1	20	ALUMINUM; FOR DESCENT TANK STABILIZATION
UMBILICAL PLATE	2	14	
SEPARATION SYSTEMS		40	
ASCENT TANK MOD SEPARATION	2	16	EXPLOSIVE BOLTS
TILT ACTUATORS, FTGS	2	24	
THRUST STRUCTURE - Delivery Stage only		70	
PRIMARY THRUST RING	1	42	ALUMINUM; DIA=6 FT, A=4.0 IN2
PRIMARY THRUST STRUTS	8	26	Gr/Ep struts, Ti ftgs; L ave = 3.5 FT, A=2.0 IN2 + 40% FTGS
ENGINE INTERFACE FITTINGS	1	2	aluminum; ENGINE, ACTUATOR INTERFACE PADS

Figure 1-2.4.1-6. Ground-Based Lander and Tankset Structures

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ITEM	Qty	Mass, kg	REMARKS (ENGLISH UNITS)
STRUCTURE - CREW MODULE		2485	
NOSE EXTERNAL BODY STRUCTURE		383	
BULKHEADS, FRAMES	6	94	ALUMINUM
SKIN PANELS		169	ALUMINUM
LONGERONS	6	29	ALUMINUM L=8.8 FT, A=1.0 IN2
LANDING GEAR WELL, FRAMES		64	ALUMINUM
LANDING GEAR DOOR	1	10	ALUMINUM
LANDING GEAR MECHANISMS		17	ALUMINUM
MID-BODY EXTERNAL STRUCTURE		550	
BULKHEADS, FRAMES	4	185	ALUMINUM
SKIN PANELS		248	ALUMINUM
LONGERONS	6	31	ALUMINUM L=9.6 FT, A=1.0 IN2
FTGS, CABIN ATTACHMENT	22	15	TITANIUM S=12 SF EA @3.0 PSF + 16 LB SEP
CHUTE COVER PANELS	2	40	ALUMINUM S=7.5 SF @ 9 PSF
BODY FLAP CLOSEOUT, HINGE SUPT		31	RCC / INSTL
CREW MODULE SECONDARY STRUCTURE		79	
EQUIPMENT PALLET INTERFACE FTGS	6	6	BOLTED INTERFACE
EQUIPMENT PALLET UMBILICAL PLATES	2	15	
UNPRESS EQUIP SUPPORT STRUCT		34	
RADIATOR PANEL SUPPORT	2	24	S=50 SF @ 1.5 PSF
PRESSURIZED CABIN PRIMARY STR		612	ALUMINUM SKIN / STRINGER
BULKHEAD, FWD	1	27	ALUMINUM S=30 SF @ 2.0 PSF
BULKHEAD, AFT	1	177	ALUMINUM S=110 SF @ 3.0 PSF + 60 LB gusset
MINOR FRAMES - CABIN, TUNNEL	5	100	ALUMINUM L, ave= 45 FT , A= 1.5 IN2
COVER PANELS, CYLINDER		308	ALUMINUM S=400 SF @ 1.7 PSF
CABIN SECONDARY STRUCTURE		392	
PARTITIONS	4	180	SLEEP STATIONS, ETC
FLOORING		83	S= 92 SF @ 2.0 PSF
PRESSURIZED EQUIP SUPPORT STRUCT		34	S=50 SF @ 1.5 PSF
LATERAL WINDOWS - PRESSURE WINDOW	2	24	S= 1.0 SF EA @ 27 PSF
LATERAL WINDOWS - THERMAL WINDOW	2	8	S= 1.0 SF EA @ 9 PSF
LATERAL WINDOWS - RETAINER	2	12	
AFT WINDOWS - PRESSURE WINDOW	2	24	S= 1.0 SF EA @ 27 PSF
AFT WINDOWS - RETAINER	2	12	
FTGS, CABIN ATTACHMENT	22	15	ALUMINUM
DOCKING MECHANISMS		160	
HATCHES		127	
EVA HATCH, STRUCTURE	1	31	TITANIUM 40-IN DIA, SHUTTLE-TYPE
EVA HATCH, MECHANISM		25	
DOCKING HATCH, STRUCTURE	1	31	40-IN DIA, SHUTTLE-TYPE
DOCKING HATCH, WINDOW & RETAINER		15	
DOCKING HATCH, MECHANISM		25	
BODY FLAP		182	
BODY FLAP STRUCTURE	2	127	RCC / INSTL S=16 SF EA @ 9.0 PSF
BODY FLAP ACTUATORS	2	55	

Figure 1-2.4.1-7.   Ground-Based Crew Module Structures

exterior and lower closeout panels are sandwich panels with honeycomb core and graphite/epoxy facesheets. These panels provide structural stiffness and shielding for the core tanks and crew module.

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 droptanks 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.

Thrust Structure. The thrust structure distributes thrust loads from the main engines to the vehicle and resists lateral engine gimbal loads. The space-based vehicle has an octagonal thrust ring attached to the core stage with 16 thrust struts and associated fittings. The thrust structure is of graphite/epoxy design, with six engine mounting pads and associated thrust vector actuator supports. The thrust structure also includes struts for lateral load stabilization. The thrust ring is estimated to have an average 4.0-in^2 cross-sectional area, excluding local beef-up and pads, and the thrust struts are 7-foot-long graphite/epoxy struts with titanium end fittings.

On the ground-based vehicle, both the core stage and delivery stages have thrust structures. The core stage is made up of a core tankset and core propulsion module, and the thrust structure is part of the propulsion module. It consists of an aluminum double-cruciform beam structure and circular thrust ring with an average 4.0-in^2 cross-sectional area 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/thrust vector control (TVC) actuator pads and lateral load stabilization struts. Each single-engine delivery stage has a thrust structure that consists of a cruciform thrust beam for load distribution into the tankset structure and an engine interface/TVC actuator support pad.

Lunar Landing Gear. Lunar landing gear includes those provisions required for landing on the lunar surface. On the space-based vehicle, that includes four sets of landing gear on the core stage that are deployed during lunar transit, stowed during Earth return, and then reused for the next mission. 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. Both types of landing gear include primary attenuator struts with 4-foot-diameter landing pads, major support struts, and deployment struts with secondary impact attenuation. The space-based attenuator struts are reusable gas-filled struts, but the ground-based struts include an expendable crushable core. The space-based deployment struts also have provisions for retracting the landing gear following lunar ascent.

To have the capability to land on a 15-degree slope for the initial unpiloted missions (assuming that the vehicle descends with a level attitude with up to 2.0-m/s vertical velocity and 0.5-m/s horizontal velocity) the space-based landing pads must be located at a 9.38m radius from the vehicle centerline and the ground-based vehicle landing pads must be at a 9.68m radius from the vehicle centerline.

Tank Module Structures. Both space- and ground-based tanksets that are launched full have external load-bearing tank support structures designed to support the filled tanks during launch and to provide a structural core vehicle interface. The current space-based tanksets include 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 droptanks 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 droptanks 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.

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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 in 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 and the inner tube for smaller on-orbit loads. Once in orbit the outer tube pulls away from the inner tube, reducing the on-orbit heat leak through the struts.

Aerobrake and Aeroshell Structure. The space-based aerobrake is a rigid space-assembled shell structure of 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. These longerons feed the loads from the honeycomb panels into the truss members. The truss structure that carries the load into the vehicle core structure consists of two primary beams that 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 assembly. On these side panels, three secondary trusses spread radially from the core structure attachment points as shown in Figure 1-2.4.1-8.

The primary and secondary trusses are graphite polyimide members in a open truss configuration. The circular longerons are solid graphite/polyimide sections with attachment points for the honeycomb panels. These panels are composed of graphite polyimide facesheets with high-temperature honeycomb core. The graphite polyimide allows the aerobrake structure to run hotter than would be allowable with an aluminum structure (650°F for GR/PI versus 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 thermal protection system (TPS). The impact of this coefficient of thermal expansion (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. Detail weights of the space-based aerobrake structure and thermal protection are given in Figure 1-2.4.1-9.

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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

Structural Definition

Graphite/polyimide honeycomb
panels

Graphite/polyimide support beams

High temperature seals
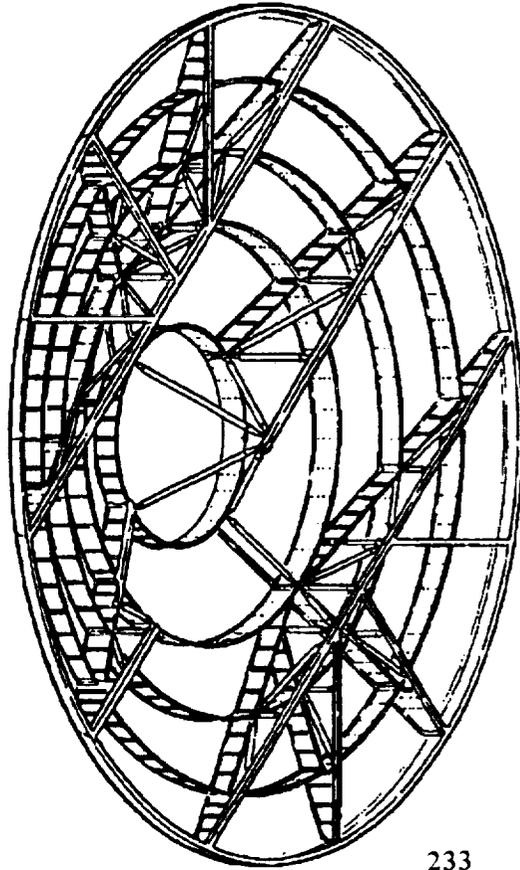
TPS Definition

Zirconia fiberous ceramic
mechanically attachable to
GR/PI

BlackGlas™ overlayment

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Figure 1-2.4.1-8. Aerobrake Structure and TPS Concept

Item	Qty	Mass (kg)	Total (kg)	Remarks (English Units)	Material
STRUCTURES AND MECHANISMS			2082		
PRIMARY STRUCT - CENTER ASSEMBLY	1		722		
MAJOR LONGERONS	2	140		L=48 ft ea, A ave=4.5 IN2	Gr/Pi
EDGE FRAME	2	20		L=17 ft ea, A=1.75 in2	Gr/Pi
RIBS / BULKHEADS	8	63		L=16 ft ea, A=1.5 in2	Gr/Pi
NOSE-CAP INTERFACE RING	1	55		L=31 ft, A=2.0 in2	Titanium
SANDWICH COVER PANELS	9	438		S=965 SF total @ 1.0 PSF	Gr/Pi H/C Sandwich
UMBILICAL DOORS	2	6		S= 3 SF ea @ 2.0 PSF	Gr/Pi H/C Sandwich
PRIMARY STRUCT - LATERAL SECTION			934		
MAJOR LONGERONS	4	188		L=48 ft ea, A ave=3.0 IN2	Gr/Pi
EDGE FRAME	2	70		L=62 ft ea, A=1.75 in2	Gr/Pi
RIBS / BULKHEADS	16	63		L ave=8 ft ea, A=1.5 in2	Gr/Pi
SANDWICH COVER PANELS	18	613		S=1350 SF total @ 1.0 PSF	Gr/Pi H/C Sandwich
SECONDARY STR - Brake assembly	1		351		
FRAME ASSEMBLY PROVISIONS		121		20% of primary structure Wt	
PANEL INSTALLATION PROVISIONS		158		15% of panel Wt	
BRAKE SUPPORT INTERFACE FTGS	8	32		Estimate	Titanium
UMBILICAL DOOR MECHANISMS	2	40		Estimate	Titanium
SECONDARY STR - BRAKE SUPPORT	1		75		
SUPPORT STRUTS	8	59		L=15 ft ea, a=1.5 in2	Gr/Ep
END FITTINGS, MISC	16	16		Estimate	Titanium
PROTECTION			1477		
EXTERNAL TPS			1372		
CENTER SECTION TPS PANELS		606		S= 971 SF @ 1.1 psf + 25% attach	Zirconia H/C panels
LATERAL SECTIONS TPS PANELS		766		S=1350 SF tot @ 1.0 psf + 25% attach	Zirconia H/C panels
INTERNAL INSULATION / TCS			105		
AEROBRAKE BACKWALL MLI		105		S=2321 SF @ 0.10 PSF	Backwall insul, MLI
COMMUNICATION & DATA HANDLING			15		
VEHICLE HEALTH MONITORING			5		
SENSORS	4	5			
INSTRUMENTATION			10		
SENSOR INTERFACE UNIT	2	2			
NETWORK INTERFACE UNIT	3	3			
SENSORS	25	5			

Figure 1-2.4.1-9. Aerobrake Structure and Thermal Protection

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Crew Module Structure. 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 extravehicular activity (EVA) and crew transfer.

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, as shown in Figure 1-2.4.1-10. The crew module has windows for landing and docking maneuvers and two hatches for EVA and crew transfer.
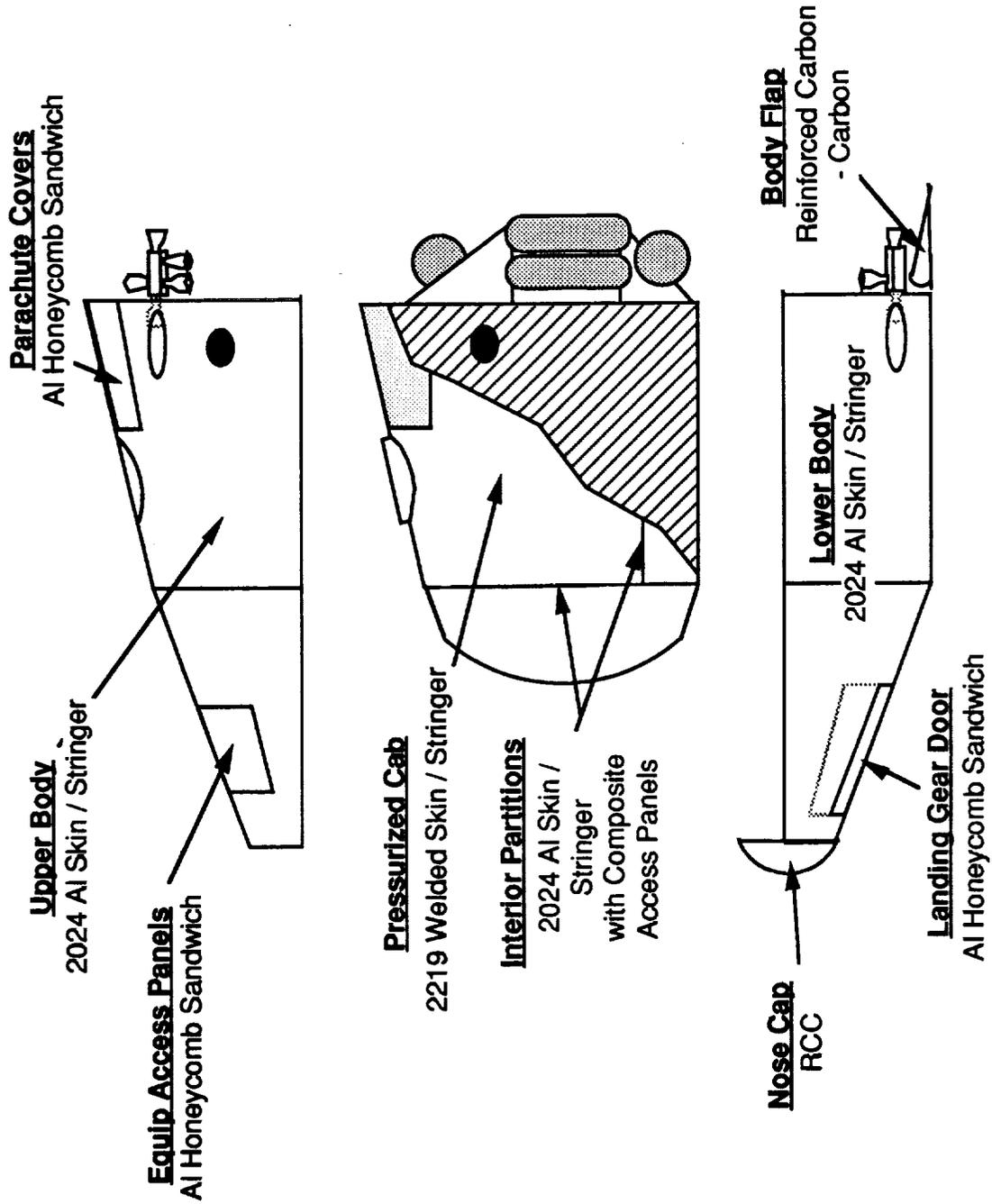
Figure 1-2.4.1-10. Ground-Based Crew Module Structures

Parachute Covers
Al Honeycomb Sandwich

Upper Body
2024 Al Skin / Stringer

Equip Access Panels
Al Honeycomb Sandwich

Pressurized Cab
2219 Welded Skin / Stringer

Interior Partitions
2024 Al Skin /
Stringer
with Composite
Access Panels

Body Flap
Reinforced Carbon
- Carbon

Lower Body
2024 Al Skin / Stringer

Landing Gear Door
Al Honeycomb Sandwich

Nose Cap
RCC

1-2.4.2 Ground Recovery

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, as shown in Figure 1-2.4.2-1.

Parachutes. The parachute system includes a primary and backup drogue chute for initial deceleration and a primary and backup high-glide parafoil chute for final deceleration to touchdown (Figure 1-2.4.2-2). Also included are the parafoil control mechanisms for final descent steering and installation provisions for the chutes. The drogue chutes are 53-foot-diameter mortar-deployed conical ribbon chutes for deceleration to a terminal velocity of 160 ft/sec. The main chutes are two-stage controllable parafoils; the initial reefed condition slows the module to about 22-ft/sec vertical velocity to minimize drift, then the parachute is opened fully to slow the vertical velocity to about 10 ft/sec for final touchdown.

Ground Return Landing Gear. 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.

Water Splashdown Provisions. 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 nosedown, so provisions must be included in the parachute system for achieving this attitude. Other provisions for a water

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ITEM	Qty	Mass, kg	REMARKS (ENGLISH UNITS)
RECOVERY SYSTEM - CREW MODULE		1137	
PARACHUTE SYSTEM		798	
DROQUE CHUTES - PRIMARY, BACKUP	2	268	
MAIN CHUTES - PRIMARY, BACKUP	2	412	
PARACHUTE CONTROL SPINDLES, MOTOR	2	45	
PARACHUTE INSTALLATION		73	10% OF PARACHUTES MASS
LANDING GEAR		281	
NOSE LANDING GEAR	1	50	
AFT LANDING GEAR	2	200	
FLOTATION COLLARS	4	5	
LANDING GEAR SUPPORT / INSTL		26	10% OF LANDING GEAR MASS
SEPARATION		59	
PARACHUTE COVER SEPARATION	2	18	
CORE VEHICLE SEPARATION BOLTS	6	41	EXPLOSIVE BOLTS

Figure 1-2.4.2-1. Ground-Based Crew Module Recovery
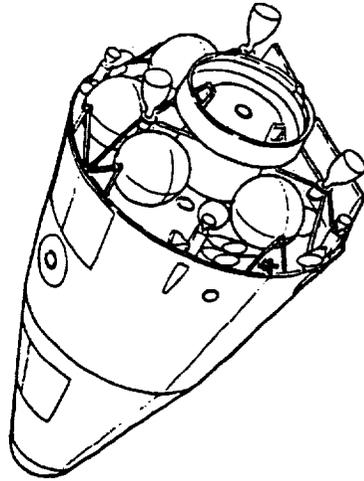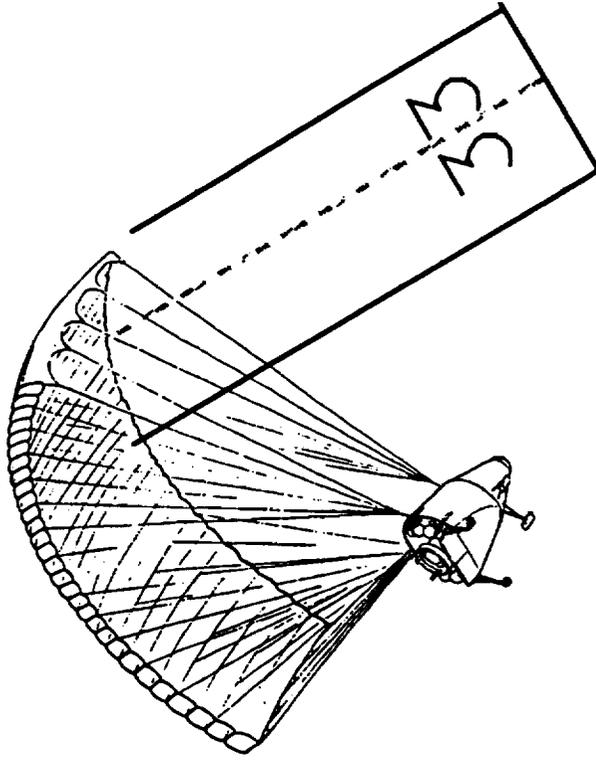
High L/D (<.8) reentry capsule
Large crossrange capability
Hypersonic maneuvering
Drogue chute deploys at M≈1

Parafoil landing
Recovery to prepared landing site
Low speed landing
Conventional landing gear

Figure 1-2.4.2-2.   Ground Recovery Concept

spashdown include flotation bags and associated inflation device of sufficient size to right the module and keep escape hatches well above the water level.

1-2.4.3 Propellant Tankage

The STV propellant tankage contains cryogenic LO2 and LH2 for the STV main engine propellant, reaction control propellant, and fuel cell reactants. The space-based core stage has two LO2 tanks and two LH2 tanks. The TLI and descent tanksets each have one LO2 and one LH2 tank, all of varying diameters because of vehicle integration size limitations caused by the aerobrake. The ground-based tanks (including core tanks, delivery stage tanks, translunar droptanks, and LO2 tanker) all have a common diameter to simplify manufacturing and installation. The core tankset, TLI tanksets, and delivery stages each have a single LO2 tank and single LH2 tank.

Requirements. Top-level requirements for the main propellant tankage are given in Figure 1-2.4.3-1.

Space-Based Core Tanks. The space-based vehicle tanks are made of all-welded 2090-T81 aluminum-lithium and are supported by struts in an external load-bearing structure. All tanks have zero-g start baskets for propellant acquisition and are sized for usable propellant capacity as follows:

1. Core Tanks - 1,537-kg liquid hydrogen each and 9,223-kg liquid oxygen each.
2. Descent Tanksets - 4,057-kg liquid hydrogen each and 24,344-kg liquid oxygen each.
3. TLI Tanksets - 9,667-kg liquid hydrogen each and 57,999-kg liquid oxygen each.

Detail weights of the space-based tanks are given in Figures 1-2.4.3-2 and 1-2.4.3-3.

All hydrogen tanks shells are sized to permit room-temperature pneumostatic proof testing to ensure mission life requirements. The core tanks are designed for reuse and are tested to 1.35 times the maximum expected operating pressure (MEOP) of 23.2 psi. The TLI and descent droptanks are designed for single use and tested to 1.10 times the MEOP of 24.5 and 23.8 psi, respectively.

BOEING

Propellant Tankage	Basing		Requirement Source		
	Space	Ground	Given	Derived	Provis.
Pressure vessel design	X	X	X		
Leak before rupture design					
1.10 x MEOP Proof Pressure					
1.50 x MEOP Ultimate Pressure					
Partial vent to refill	X				X
Design cycles - 40 (reusable), 5 (expendable)	X	X			X

Figure 1-2.4.3-1. Tankage Design Requirements

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D180-32040-2

Item	Qty	Mass (kg)	Remarks	
CORE STAGE TANKAGE		583		
LH2 TANK		349		
FWD DOME	4	36	S= 54 SF, t=0.020 in + 30 % weldlands	2090 Alum-Lithium
FWD RING	4	24	L= 20 ft, A= 0.5 in2 + 10%	2219 Aluminum
CYLINDER	4	119	S= 180 SF, t= .020 in + 30% weldlands	2090 Alum-Lithium
AFT RING	4	34	L= 25 ft, A= 0.5 in2 + 10% + 5 lb pads	2219 Aluminum
AFT DOME	4	36	S= 54 SF, t=0.020 in + 30 % weldlands	2090 Alum-Lithium
SLOSH RING BAFFLE	4	16	FOR SLOSH DAMPING	2090 Alum-Lithium
VORTEX BAFFLE, SCREEN	4	20	START BASKET, SUMP	2090 Alum-Lithium
PROPELLANT GAUGING		0	INCLUDED IN INSTRUMENTATION	
TANK SUPPORT STRUTS, FTGS	32	64	Passive Disconnect Struts	Gr/Ep, Ti ftgs
LO2 TANK		234		
FWD DOME	2	21	S= 62 SF, t=0.021 in + 30 % weldlands	2090 Alum-Lithium
FWD RING	2	27	L= 22 ft, A= 1.0 in2 + 10%	2219 Aluminum
CYLINDER	2	47	S= 95 SF, t=0.030 in + 30 % weldlands	
AFT RING	2	36	L= 22 ft, A= 1.0 in2 + 10% + 10 lb pads	2219 Aluminum
AFT DOME	2	21	S= 62 SF, t=0.021 in + 30 % weldlands	2090 Alum-Lithium
SLOSH BAFFLES / SCREENS	2	8	FOR SLOSH DAMPING	2090 Alum-Lithium
VORTEX BAFFLE, SCREEN	2	10	START BASKET, SUMP	2090 Alum-Lithium
PROPELLANT GAUGING		0	INCLUDED IN INSTRUMENTATION	
TANK SUPPORT STRUTS, FTGS	32	64	Passive Disconnect Struts	Gr/Ep, Ti ftgs

Figure 1-2.4.3-2. Space-Based Core Stage Tankage

Item	Qty	Mass (kg)	Remarks	
DESCENT TANKSET TANKAGE		**597**		
LH2 TANK		325		
FWD DOME	1	52	S= 227 SF, t=0.029 in + 30 % weldlands	2090 Alum-Lithium
FWD RING	1	27	L= 45 ft, A= 1.0 in2 + 10%	2219 Aluminum
CYLINDER	1	107	S= 355 SF, t= .036 in + 30% weldlands	2090 Alum-Lithium
AFT RING	1	36	L= 453 ft, A= 1.0 in2 + 10% + 20 lb pads	2219 Aluminum
AFT DOME	1	52	S= 227 SF, t=0.029 in + 30 % weldlands	2090 Alum-Lithium
SLOSH RING BAFFLE	1	10	FOR SLOSH DAMPING	2090 Alum-Lithium
VORTEX BAFFLE, SCREEN		9	START BASKET	
PROPELLANT GAUGING		0	INCLUDED IN INSTRUMENTATION	
TANK SUPPORT STRUTS, FTGS	16	32	Passive Disconnect Struts	Gr/Ep, Ti ftgs
LO2 TANK		272		
FWD DOME	1	79	S= 224 SF, t=0.043 in + 30 % weldlands	2090 Alum-Lithium
FWD RING	1	27	L= 45 ft, A= 1.0 in2 + 10%	2219 Aluminum
CYLINDER	0	0		
AFT RING	1	36	L= 45 ft, A= 1.0 in2 + 10% + 20 lb pads	2219 Aluminum
AFT DOME	1	79	S= 224 SF, t=0.043 in + 30 % weldlands	2090 Alum-Lithium
SLOSH RING BAFFLE	1	10	FOR SLOSH DAMPING	2090 Alum-Lithium
VORTEX BAFFLE, SCREEN	1	9	START BASKET, SUMP	
PROPELLANT GAUGING		0	INCLUDED IN INSTRUMENTATION	
TANK SUPPORT STRUTS, FTGS	16	32	Passive Disconnect Struts	Gr/Ep, Ti ftgs
TLI TANKSET TANKAGE		**1089**		
LH2 TANK		599		
FWD DOME	1	137	S= 509 SF, t=0.034 in + 30 % weldlands	2090 Alum-Lithium
FWD RING	1	39	L= 63 ft, A= 1.0 in2 + 10%	2219 Aluminum
CYLINDER	1	185	S= 401 SF, t= .056 in + 30% weldlands	2090 Alum-Lithium
AFT RING	1	48	L= 63 ft, A= 1.0 in2 + 10% + 20 lb pads	2219 Aluminum
AFT DOME	1	137	S= 509 SF, t=0.034 in + 30 % weldlands	2090 Alum-Lithium
SLOSH RING BAFFLE	1	12	FOR SLOSH DAMPING	2090 Alum-Lithium
VORTEX BAFFLE, SCREEN		9	START BASKET, SUMP	
PROPELLANT GAUGING		0	INCLUDED IN INSTRUMENTATION	
TANK SUPPORT STRUTS, FTGS	16	32	Passive Disconnect Struts	Gr/Ep, Ti ftgs
LO2 TANK		490		
FWD DOME	1	165	S= 372 SF, t=0.054 in + 30 % weldlands	2090 Alum-Lithium
FWD RING	1	33	L= 54 ft, A= 1.0 in2 + 10%	2219 Aluminum
CYLINDER	0	0		
AFT RING	1	42	L= 54 ft, A= 1.0 in2 + 10% + 20 lb pads	2219 Aluminum
AFT DOME	1	165	S= 372 SF, t=0.054 in + 30 % weldlands	2090 Alum-Lithium
SLOSH BAFFLES / SCREENS	1	12	FOR SLOSH DAMPING	2090 Alum-Lithium
VORTEX BAFFLE, SCREEN		9	START BASKET, SUMP	
PROPELLANT GAUGING		0	INCLUDED IN INSTRUMENTATION	
TANK SUPPORT STRUTS, FTGS	32	64	Passive Disconnect Struts	Gr/Ep, Ti ftgs

Figure 1-2.4.3-3. Space-Based Droptank Set Tankage

The oxygen tank shells are sized to permit room temperature proof testing using water as the proof-test fluid to ensure life requirements, with the core tanks tested to 1.35 times the MEOP of 35.7 psi. The TLI and descent tanks are tested to 1.10 times the MEOP of 45.8 and 36.1 psi, respectively.

In all expendable tanks launched from the ground full, the MEOP occurs during launch. In the reusable core tanks that are launched from the ground empty, the MEOP occurs during trans-Earth injection. The assumed ullage maximum vent pressure is 22 psi.

Ground-Based Tanks. The ground-based vehicle tanks are also made of all-welded 2090-T81 aluminum-lithium and are supported by struts in an external load-bearing structure. The LO2 tanker that is launched separate from the vehicle is supported on a launch adapter. All tanks have zero-g start baskets for propellant acquisition and are sized for usable propellant capacity as follows:

1. Core Tanks - 2,520-kg LH2 and 34,183-kg LO2 (the LO2 tank is oversized to keep a common tank diameter).
2. Delivery Stage Tanks - 6,490-kg liquid hydrogen each and 34,183-kg liquid oxygen each.
3. TLI Droptanks - 6,490-kg liquid hydrogen each and 34,183-kg liquid oxygen each.
4. LO2 Tanker - 136,800-kg liquid oxygen.

Detail weights of the ground-based tanks are given in Figure 1-2.4.3-4.

The hydrogen tanks shells are all expendable and sized to permit room temperature pneumostatic proof testing to 1.10 times the MEOP of 23.5 psi for the core tank and 25.2 psi for the TLI and delivery tanks. The oxygen tanks shells are sized to permit room temperature proof testing, using water as proof-test fluid, to 1.10 times the MEOP of 43.2 psi for the core tank, TLI tanks, and delivery stage tanks and 87.9 psi for the LO2 tanker.

In all ground-based vehicle tanks, the MEOP occurs at launch maximum acceleration, assuming an ullage maximum vent pressure of 22 psi. During

ITEM	QTY	Mass (kg)	REMARKS (ENGLISH UNITS)	
TANKAGE - ASCENT TANKSET			**385**	
LH2 TANK		186		
FWD DOME	1	40	S= 233 SF, t=0.022 in + 30 % weldlands, stiff	2090 Alum-Lithium
FWD RING	1	26	L= 42 ft, A= 1.0 in2 + 10%	2219 Aluminum
CYLINDER	1	28	S= 114 SF, t= .031 in + 30% weldlands, stiff	2090 Alum-Lithium
AFT RING	1	34	L= 42 ft, A= 1.0 in2 + 10% + 5 lb pads	2219 Aluminum
AFT DOME	1	40	S= 233 SF, t=0.022 in + 30 % weldlands, stiff	2090 Alum-Lithium
SLOSH RING BAFFLE	1	8	FOR SLOSH DAMPING	2090 Alum-Lithium
VORTEX BAFFLE, SCREEN	1	10	START BASKET, SUMP	2090 Alum-Lithium
PROPELLANT GAUGING	0	0	INCLUDED IN INSTRUMENTATION	
TANK SUPPORT STRUTS, FTGS	0	0	Passive Disconnect Struts	Gr/Ep, Ti ftgs
LO2 TANK		199		
FWD DOME	1	65	S= 233 SF, t=0.036 in + 30 % weldlands, stiff	2090 Alum-Lithium
FWD RING	1	51	L= 42 ft, A= 2.0 in2 + 10% + 5 lb pads	2219 Aluminum
CYLINDER	0	0		
AFT RING	0	0		
AFT DOME	1	65	S= 233 SF, t=0.036 in + 30 % weldlands, stiff	2090 Alum-Lithium
SLOSH BAFFLES / SCREENS	1	8	FOR SLOSH DAMPING	2090 Alum-Lithium
VORTEX BAFFLE, SCREEN	1	10	START BASKET, SUMP	2090 Alum-Lithium
PROPELLANT GAUGING	0	0	INCLUDED IN INSTRUMENTATION	
TANK SUPPORT STRUTS, FTGS	0	0	Passive Disconnect Struts	Gr/Ep, Ti ftgs
TANKAGE - DELIV STAGE & TLI TANKSET			**659**	
LH2 TANK		386		
FWD DOME	1	40	S= 233 SF, t=0.022 in + 30 % weldlands, stiff	2090 Alum-Lithium
INTERMEDIATE RING	2	51	L= 42 ft, A= 1.0 in2 + 10%	2219 Aluminum
CYLINDER	1	202	S= 718 SF, t= .033 in + 40% weldlands, stiff	2090 Alum-Lithium
AFT RING	1	28	L= 42 ft, A= 1.0 in2 + 10% + 5 lb pads	2219 Aluminum
AFT DOME	1	40	S= 233 SF, t=0.022 in + 30 % weldlands, stiff	2090 Alum-Lithium
SLOSH RING BAFFLE	2	16	FOR SLOSH DAMPING	2090 Alum-Lithium
VORTEX BAFFLE, SCREEN	1	10	START BASKET	2090 Alum-Lithium
PROPELLANT GAUGING	0	0	INCLUDED IN INSTRUMENTATION	
TANK SUPPORT STRUTS, FTGS	0	0	Passive Disconnect Struts	Gr/Ep, Ti ftgs
LO2 TANK		273		
FWD DOME	1	65	S= 233 SF, t=0.036 in + 30 % weldlands, stiff	2090 Alum-Lithium
FWD RING	1	39	L= 42 ft, A= 1.5 in2 + 10%	2219 Aluminum
CYLINDER	1	46	S= 97 SF, t= .060 in + 30% weldlands, stiff	2090 Alum-Lithium
AFT RING	1	41	L= 42 ft, A= 1.5 in2 + 10% + 5 lb pads	2219 Aluminum
AFT DOME	1	65	S= 233 SF, t=0.036 in + 30 % weldlands, stiff	2090 Alum-Lithium
SLOSH RING BAFFLE	1	8	FOR SLOSH DAMPING	2090 Alum-Lithium
VORTEX BAFFLE, SCREEN	1	10	START BASKET, SUMP	2090 Alum-Lithium
PROPELLANT GAUGING	0	0	INCLUDED IN INSTRUMENTATION	
TANK SUPPORT STRUTS, FTGS	0	0	Passive Disconnect Struts	Gr/Ep, Ti ftgs

Figure 1-2.4.3-4. Ground-Based Vehicle Tankage

piloted flight, axial accelerations are significantly less than during launch, reducing tank bottom pressures and adding to inflight margins of safety.

For ease of design, all vehicle oxygen tanks are the same size and design as the core LO2 tank. If launched full, these tanks have the same membrane thickness as the core tank, but if launched empty when used in conjunction with a LO2 tanker the membranes could be thinner.

The vehicle propellant tanks are supported in the body structure by pin-ended struts. The hydrogen tanks are supported by fiberglass struts with aluminum end fittings, and the oxygen tanks are supported by graphite/epoxy struts with titanium end fittings.

Propellant Acquisition and Management. Antivortex baffles over tank outlets are required for all tanks (TLI, drop, and core) to minimize propellant residuals by suppressing vapor pull-through. Slosh baffling will likely be required and is included in the tank designs, although often structural members alone can provide adequate slosh damping. The number of and locations for slosh damping baffles will depend on the structural dynamics of the complete vehicle, and therefore will require analysis with NASTRAN (structural dynamics code) and FLOW-3D (computational fluid dynamics code) relatively late in the design process.

1-2.4.4   Protection

The function of the protection subsystem is to protect primary structure, equipment, and crew from the effects of the space environment and the effects of aerobraking and reentry.

Requirements. Top-level requirements for the protection subsystem are given in Figure 1-2.4.4-1.

Hardware Description. The major elements of the protection subsystem include:

1. Radiation protection.
2. Micrometeoroid and space debris protection.
3. External thermal protection (TPS).
4. Passive thermal control.
5. Purge and vent systems.
6. Window and hatch conditioning.

Detail weights of the space-based vehicle protection systems are given in Figures 1-2.4.4-2 and 1-2.4.4-3. Detailed weights for the ground-based vehicle are given in Figures 1-2.4.4-4 and 1-2.4.4-5 for the vehicle and crew module, respectively. The space-based aerobrake protection weights were shown in section 1-2.4.1.

Radiation Protection. One possible method of protecting the crew from dangerous levels of radiation during transit to and from the lunar surface is to surround them with a layer of water, which is then used for reentry or aeromaneuver cooling. Both ground- and space-based crew modules have a series of conformal water jackets surrounding the pressure shell, which are launched full in the case of the ground-based system or filled on orbit in the case of the space-based system. The current designs are sized to hold 1,800 kg of water for radiation protection.

Micrometeoroid and Debris Protection. All tanksets include provisions for meteoroid and debris protection to minimize the risk of a tank wall penetration

BOEING

Protection	Basing		Requirement Source		
	Space	Ground	Given	Derived	Provis.
Protection					
Radiation health	X	X	X		
NCRP Report 98, July 31, 1989 (dosage levels) adhered to					
Protection provided per NASA STD-3000, sec. 5.7.2.2					
Radiation protection not required for LEV					
Meteoroid / debris protection	X	X	X		
0.9955 probab. of no crit. element failure from meteoroid impact					
NASA TM 100471 - Orbital Debris Environment, 1989					
SP 8013 - Meteoroid environment near Earth to Lunar surface					
Provide protection for equipment and crew during aeromaneuver	X	X		X	
On-orbit assembly capability	X			X	
Reusability or simple replacement / refurbishment	X	X		X	

Figure 1-2.4.4-1. Protection System Design Requirements

BOEING

Item	Qty	Mass (kg)	Remarks
CORE STAGE PROTECTION		**437**	
EXTERNAL TPS		0	
EXTERNAL TCS		309	
LH2 TANK INSULATION, MLI	4	196	S= 288 sf ea. @0.375 psf (90 layer, incl instl) Double-Alum Kapton
LO2 TANK INSULATION, MLI	2	75	S= 219 SF ea. @ 0.375 psf (90 layer, incl instl) Double-Alum Kapton
MISC INSULATION		38	S= 1200 SF @ 0.07 psf FOSR
EQUIPMENT THERMAL PALLET	8	128	For equipment cooling
CREW MODULE PROTECTION		**509**	
RADIATION PROTECTION		350	Water system
STRUCTURE - tanks	4	300	Welded tanks around crew module — s. steel
WATER PUMPS	3	30	for water placement control - reversible
WATER PLUMBING, FTGS, ETC	2	20	
METEOROID / DEBRIS PROTECTION		0	NOT REQUIRED
EXTERNAL TPS		21	
MLI - REENTRY PROTECTION	21	21	S= 450 SF @ 0.1 PSF — MLI
INTERNAL INSULATION / TCS		112	
BULK INSULATION	112	112	S= 705 SF @ 0.35 PSF SCALED FROM SHUTTLE — BULK INSUL
PURGE AND VENT SYSTEM		19	SCALED FROM SHUTTLE
DUCTING	10		
VALVES	5		
SUPPORT, INSTALLATION	4	8	
WINDOW / HATCH CONDITIONING			SCALED FROM SHUTTLE
PLUMBING	3		
DESSICANT, VALVES, DISCONNECTS	4		
SUPPORT, INSTALLATION	1		
THERMAL REJECTION PALLETS	0	0	

Figure 1-2.4.4-2. Space-Based Core Protection Systems

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BOEING

Item	Qty	Mass (kg)	Remarks
DESCENT TANKSET PROTECTION		**385**	
METEOROID / DEBRIS PROTECTION		89	S= 900 SF, t=0.018 in, Gr/Ep with fiberglas supports (40% of tot)
EXTERNAL THERMAL CONTROL		273	
LH2 TANK INSULATION, MLI	138		S= 809 SF @0.375 psf (90 layer, incl instl) Double-Alum Kapton
LH2 TANK INSULATION, FOAM	38		S= 809 SF @ 0.104 psf (t=0.5 in) CPR 488
LO2 TANK INSULATION, MLI	76		S= 448 SF @ 0.375 psf (90 layer, incl instl) Double-Alum Kapton
LO2 TANK INSULATION, FOAM	21		S= 448 SF @ 0.104 psf (t=0.5 in) CPR 488
PURGE AND VENT SYSTEM		23	PURGE DUCT AND AUTOMATIC VENTS - ESTIMATE
TLI TANKSET PROTECTION		**642**	
METEOROID / DEBRIS PROTECTION		148	S= 1500 SF, t=0.018 in, Gr/Ep with fiberglas supports (40% of tot)
EXTERNAL THERMAL CONTROL		471	
LH2 TANK INSULATION, MLI	242		S= 1419 SF @0.375 psf (90 layer, incl instl) Double-Alum Kapton
LH2 TANK INSULATION, FOAM	67		S= 1419 SF @ 0.104 psf (t=0.5 in) CPR 488
LO2 TANK INSULATION, MLI	127		S= 744 SF @ 0.375 psf (90 layer, incl instl) Double-Alum Kapton
LO2 TANK INSULATION, FOAM	35		S= 744 SF @ 0.104 psf (t=0.5 in) CPR 488
PURGE AND VENT SYSTEM		23	PURGE DUCT AND AUTOMATIC VENTS - ESTIMATE

Figure 1-2.4.4-3. Space-Based Droptank Set Protection Systems

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BOEING

ITEM	QTY	Mass (kg)		REMARKS (ENGLISH UNITS)
PROTECTION - PALLET				
EXTERNAL TPS			116	FOSR
REENTRY / IN-SPACE PROTECTION		10		S= 315 SF @ 0.07 PSF SCALED FROM SHUTTLE
PURGE AND VENT SYSTEM		10		
DUCTING		5		
VALVES		3		
SUPPORT, INSTALLATION		2		
THERMAL REJECTION PALLETS	6		96	
PROTECTION - ASCENT TANKAGE			378	
METEOROID / DEBRIS PROTECTION	1	110		S= 1010 SF, t=0.02 in, Gr/Ep with fiberglas supports (40% of tot)
EXTERNAL TCS		238		
LH2 TANK INSULATION, MLI	1	103		Double-Alum Kapton S= 608 sf ea. @0.375 psf (90 layer, incl instl)
LH2 TANK INSULATION, FOAM	1	28		CPR 488 S= 608 sf ea. @0.104 psf (t=0.5 in)
LO2 TANK INSULATION, MLI	1	83		Double-Alum Kapton S= 489 SF ea. @ 0.375 psf (90 layer, incl instl)
LO2 TANK INSULATION, FOAM	1	24		CPR 488 S= 489 SF ea. @ 0.104 psf (t=0.5 in)
PURGE AND VENT SYSTEM		30		PURGE DUCT AND AUTOMATIC VENTS - ESTIMATE
PROTECTION - LANDER			82	
EXTERNAL TPS		82		
IN-SPACE/LUNAR SURFACE PROTECTION		50		FOSR S= 1570 SF @ 0.07 PSF
LANDING GEAR SURFACE PROTECTION		32		FOSR S= 700 SF @ 0.10 PSF
PROTECTION - DELIV STAGE & TLI TANKSET			570	
METEOROID / DEBRIS PROTECTION	1	144		S= 1319 SF ea, t=0.02 in, Gr/Ep with fiberglas supports (40% of tot)
EXTERNAL THERMAL CONTROL		397		
LH2 TANK INSULATION, MLI	1	211		Double-Alum Kapton S= 1237 SF ea@0.375 psf (90 layer, incl instl)
LH2 TANK INSULATION, FOAM	1	59		CPR 488 S= 1237 SF ea@ 0.104 psf (t=0.5 in)
LO2 TANK INSULATION, MLI	1	100		Double-Alum Kapton S= 589 SF ea@ 0.375 psf (90 layer, incl instl)
LO2 TANK INSULATION, FOAM	1	28		CPR 488 S= 589 SF ea@ 0.104 psf (t=0.5 in)
PURGE AND VENT SYSTEM		30		PURGE DUCT AND AUTOMATIC VENTS - ESTIMATE

Figure 1-2.4.4-4. Ground-Based Vehicle Protection Systems

BOEING

ITEM	Qty	Mass, kg	REMARKS (ENGLISH UNITS)	
PROTECTION - CREW MODULE		1315		
RADIATION PROTECTION		350		
STRUCTURE - tanks	4	300	Water system	s. steel
WATER PUMPS	3	30	Welded tanks around crew module	
WATER PLUMBING, FTGS, ETC	2	20	for water placement control - reversible	
METEOROID / DEBRIS PROTECTION		0	NOT REQUIRED	
EXTERNAL TPS		801		
NOSE CAP PANELS, INSTL		47	S= 13.0 SF @ 5.0 PSF + 60% INSTL	
BODY TPS		560	S = 561 SF @ 2.20 PSF (t=0.85 IN)	
TPS INSTALLATION SUBSTRATE		160	S= 561 SF @ 0.63 PSF	
AFT BULKHEAD TPS		34	S= 184 SF @ 0.4 PSF	
INTERNAL INSULATION / TCS		124		
NOSE CAP BACKWALL INSULATION		24	S= 9.0 SF @ 6.0 PSF	ABLATOR
BODY MLI INSULATION		17	S = 525 SF @ 0.07 PSF	COMPOSITE
BODY BULK INSULATION		83	S = 525 SF @ 0.35 PSF	RIGIDIZED TABI
PURGE AND VENT SYSTEM		31	SCALED FROM SHUTTLE	MLI
DUCTING		15		BULK INSUL
VALVES		10		
SUPPORT, INSTALLATION		6		
WINDOW / HATCH CONDITIONING		8	SCALED FROM SHUTTLE	
PLUMBING		4		
DESSICANT, VALVES, DISCONNECTS		4		
SUPPORT, INSTALLATION		1		
THERMAL REJECTION PALLETS	0	0		

*Figure 1-2.4.4-5. Ground-Based Crew Module Protection Systems*

and leak. Crew module pressurized capsules are also vulnerable but are shielded by other vehicle structure. The ground-based crew module has an exterior aerodynamic shell, and the space-based crew module is nested in the core stage and shielded by the aerobrake.

The expendable droptank sets are shielded in part by intertank and support structures and by a combination of an exterior bumper shield and tank thermal insulation. The bumper shields are 0.018-inch-thick graphite/epoxy shells supported 3 inches off the tank wall by a fiberglass standoff structure. These shields protect the tanks from meteoroid impact and ground-handling damage. Theoretically, in the event of a debris impact, the impacting object vaporizes at the bumper and the smaller particles then spread and impact the next layers of material, in this case tank insulation. With this design, the probability of impact on the tank wall itself is small.

External Thermal Protection. Thermal protection systems on the current vehicle designs include a reusable tile system on the space-based aerobrake and a combination ablator and reusable tile system on the ground-based crew module.

The space-based aerobrake is a reusable system that must be space assembled and inspected at the Space Station between flights. The thermal protection system consists of fibrous zirconia panels that can be installed at the Space Station. The panels are made of a Boeing-developed fiberous ceramic molded into a honeycomb substrate and covered with a tough ceramic coating. The zirconia panels are an average 1.0 inch thick, similar in density to shuttle tiles, and mechanically attached to the aerobrake structure. Each panel covers a much larger area than a shuttle tile, minimizing joint closure and inspection problems.

The ground-based crew module has a combination of bolt-on ablator panels on the nose cap, a reusable carbon-carbon body flap, and bolt-on reusable fibrous zirconia panels over the rest of the body, as shown in Figure 1-2.4.4-6. The ablator panels include an expendable ablator on a mechanically attached substrate that can be replaced after each flight. The zirconia panels are similar to those mentioned earlier and are reusable.

Reinforced Carbon - Carbon

Ablator or Transpiration cooled Zirconia

Fiberous Zirconia

Rigidized TABI

TABI on aft Bulkhead

Upper Surface

Lower Surface

Stagnation point TPS options

Transpiration cooling

Advantages	Disadvantages
Reusable	Complex
Fault tolerant	Untried
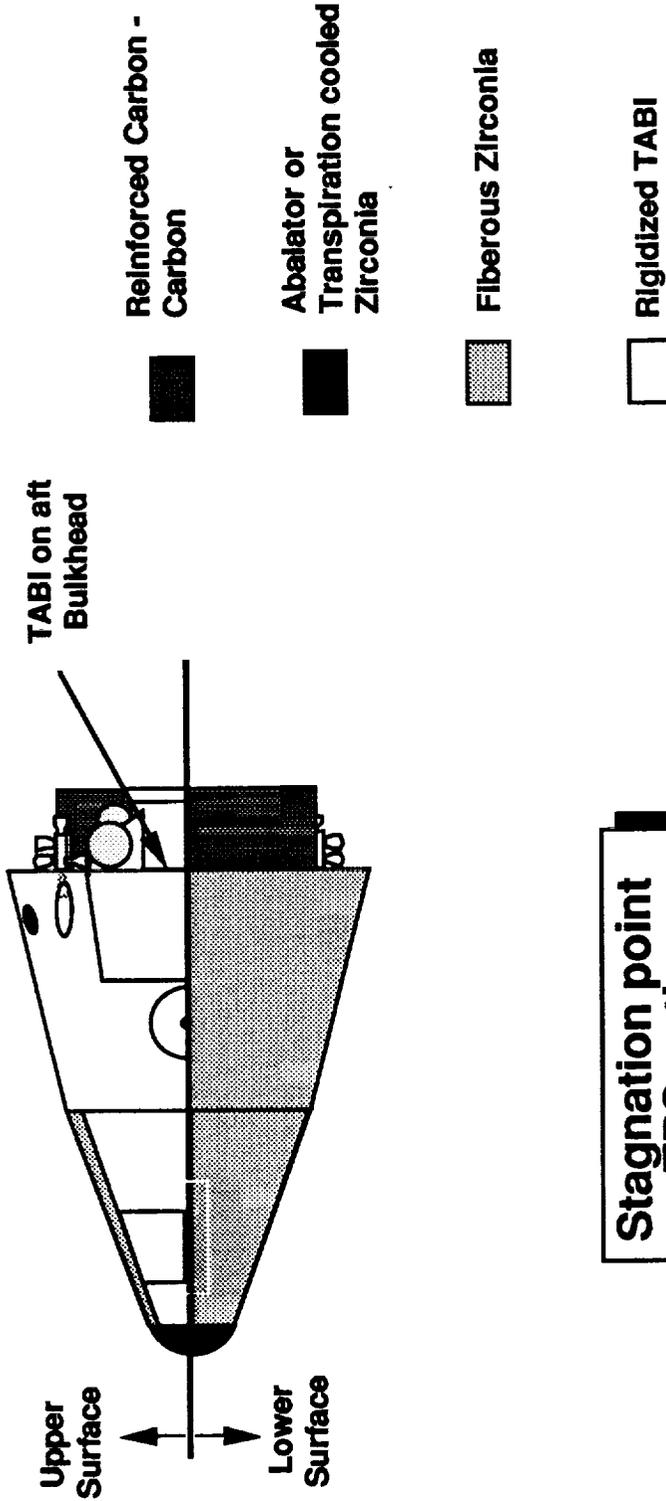
Bolt on ablator

Advantages	Disadvantages
Simple	Heavy
Easily replaced	Must be replaced
Well tested	

Figure 1-2.4.4-6. Ground-Based Crew Module Aeroshell TPS Concept

Passive Thermal Control. Passive thermal control systems include insulation blankets, rigid foam insulation, thermal control coatings, and selected radiative surfaces.

All cryogenic tanks that are launched full are insulated with a combination of low-density foam and multilayer insulation (MLI) to minimize heat leak into the tank during pre-launch, launch, and lunar transit. Space-based tanks are insulated only with MLI. The foam insulation is a sprayable isocyanurate foam with low thermal conductance such as CPR-488 (400.4 kg/m^3 density), which is effective during pre-launch and launch conditions and is applied to the outside of the tanks. A 0.5-inch layer of foam is applied to all cryogenic tanks launched full. The MLI consists of layers of doubly-aluminized Kapton with Dacron net spacers and an outside purge barrier of beta-cloth for damage resistance. The MLI is effective thermal control for the space environment and is applied outside the foam insulation. For the STV designs, 90 layers of MLI are applied to all cryogenic tanks, with a unit mass of 1.83 kg/m^2.

MLI, bulk insulation, and thermal coatings are used around the crew modules and on selected structural elements for inflight thermal control and to reduce thermal distortions. Figure 1-2.4.4-7 shows the internal thermal insulation areas of the ground-based crew module.

Both the ground-based and space-based avionics and power equipment are mounted on thermal pallets that radiate excess heat to space during transit. These pallets make use of heat-pipe technology and are considered passive, requiring no mechanical circulation devices or external controls or monitoring.

Purge and Vent. All MLI and enclosed volumes require gas purge prior to launch to prevent buildup of ice and venting during launch for pressure equalization. The purge systems distribute dry gas (helium for the hydrogen tanks and nitrogen for the LO2 tanks) from ground support equipment into the MLI areas and enclosed body volumes. During launch, provision is made to vent trapped gases through pressure-sensitive closeouts and doors to prevent pressure damage to multilayer materials.

Forward Body-shell bulk insulation areas

MLI installed in gear well and over all bulkheads and frames

Crew cabin bulk insulation areas (between outer shell and pressure wall)

MLI installed over all shell frames and bulkheads, as well as over aft bulkhead and equipment
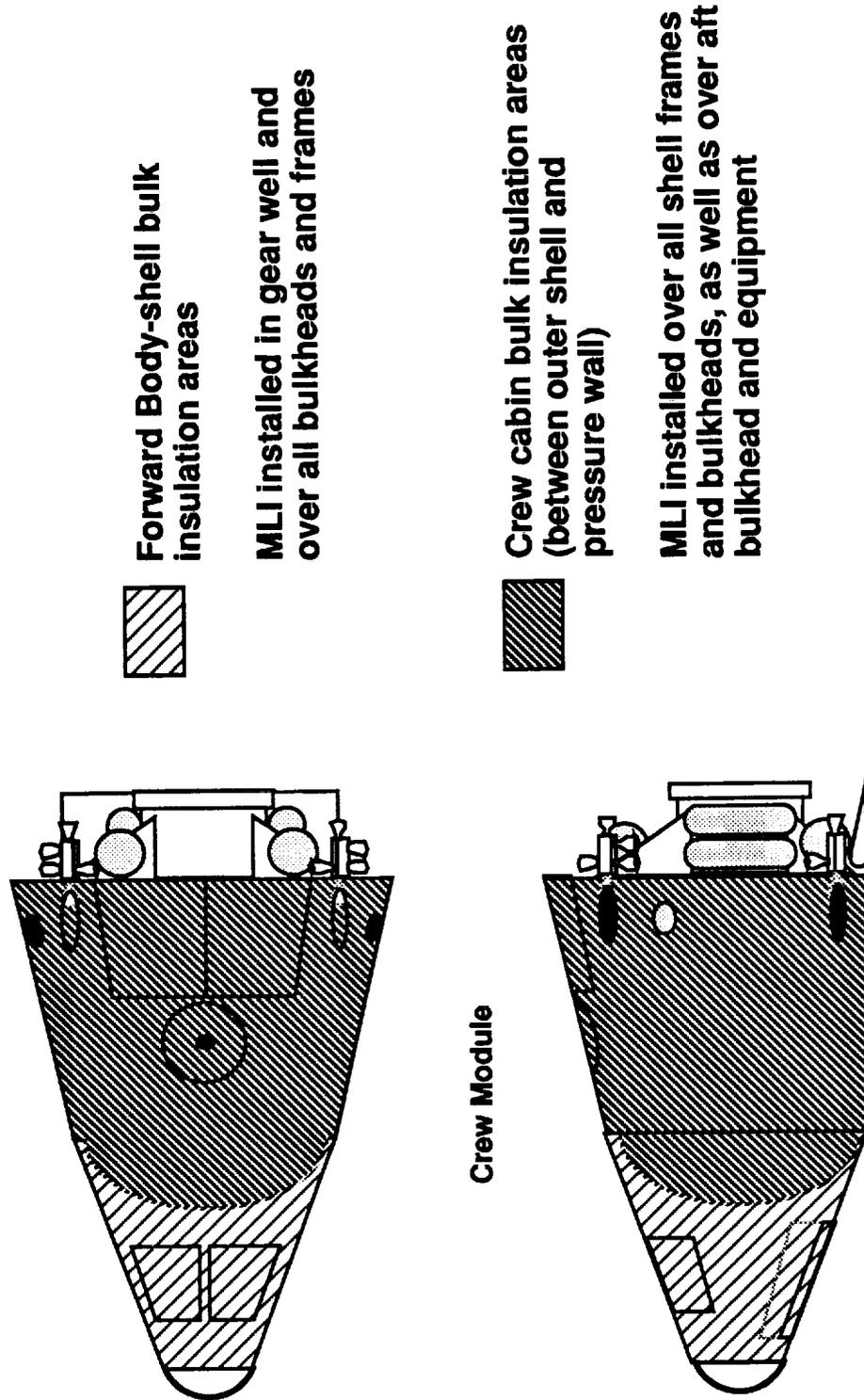
Crew Module

Figure 1-2.4.4-7. Ground-Based Crew Module Internal Insulation

Window and Hatch Conditioning. The window and hatch conditioning systems thermally condition hatches and windows to prevent damage caused by thermal distortion and to maintain ease of operation. The system consists of dessicants and active redundant thermal cooling loops to minimize condensation and maintain constant temperatures.

1-2.4.5   Main Propulsion

The main propulsion system provides propulsion for all burns requiring high thrust and/or large ΔV changes.

Requirements. Top-level requirements for the main propulsion subsystem are given in Figure 1-2.4.5-1. Derived STV engine parameters are shown in Figure 1-2.4.5-2.

Hardware Description. The space-based vehicle includes six advanced expander cycle engines on the core stage with propellant delivery, fill and drain, vent, and pressurization systems. A schematic of the space-based main propulsion system is given in Figure 1-2.4.5-3, and main propulsion part lists and weights are given in Figures 1-2.4.5-4, 1-2.4.5-5, and 1-2.4.5-6 for the space-based core stage, TLI droptank sets, and descent droptank sets, respectively.

The ground-based vehicle has a total of six advanced expander cycle engines with four of the engines on the core stage and two located on the delivery stages. The ground-based vehicle also includes provisions for propellant fill and drain, delivery, pressurization, and vent systems. Detail main propulsion weights are shown in Figures 1-2.4.5-6 through 1-2.4.5-11 for the ground-based vehicle flight elements.

Main Engines. The main engines used for the lunar missions are advanced LO2/LH2 expander cycle space engines rated at a maximum vacuum thrust of 66,723N (15,000 lbf) each, with a specific impulse of 481 seconds at an oxidizer-to-fuel mixture ratio of 6:1. These engines require a throttling capability of 5:1 for the lunar landing. For early unmanned non-lunar missions, RL10 engines are used, with a similar vacuum thrust and an Isp of 470 seconds.

The advanced 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 include provisions for supplying autogenous pressurization gases for tank pressurization.

Main Propulsion	Basing		Requirement Source		
	Space	Ground	Given	Derived	Provis.
Main engines shall require no servicing between missions	x		x		
Lines and fitting design	x	x	x		
2.00 x MEOP Proof Pressure (<1.5 in dia)					
4.00 x MEOP Ultimate Pressure (<1.5 in dia)					
1.20 x MEOP Proof Pressure (>1.5 in dia)					
1.50 x MEOP Ultimate Pressure (>1.5 in dia)					
LO2 / LH2 propellants	x	x	x		
Engine throttle capability for lunar landing	x	x	x		
7-15 year life with 20 restarts	x		x		
Tank head idle for propellant settling	x	x			x
Autogenous O2/H2 pressurization	x	x			x

Figure 1-2.4.5-1. Main Propulsion Design Requirements

Parameter	Value	Rationale
No. Engines	6	Provide engine out operation
Thrust	66270 n	Adequate thrust to performTLI maneuver
Throttling	1/5	Minimun impact to turbopump design; lower risk
Specific Impulse	481sec	Saves 25000 kg launched from LEO
Expansion Ratio	640	Necessary to achieve Isp
Chamber Press.	10,342 kpa	Reduce engine envelope and increase performance
Mixture Ratio	6	Minimize inert (tank) weight
Gimbal Angle	±10°(2 axes)	Control authority during lunar landing
No. of Starts	300	⎫ 5 missions minimum reuse
Tot. Burn Time	36000sec	⎬
Start	Tank head	Eliminate propellant tank pressurization

Figure 1-2.4.5-2. Main Engine Requirements

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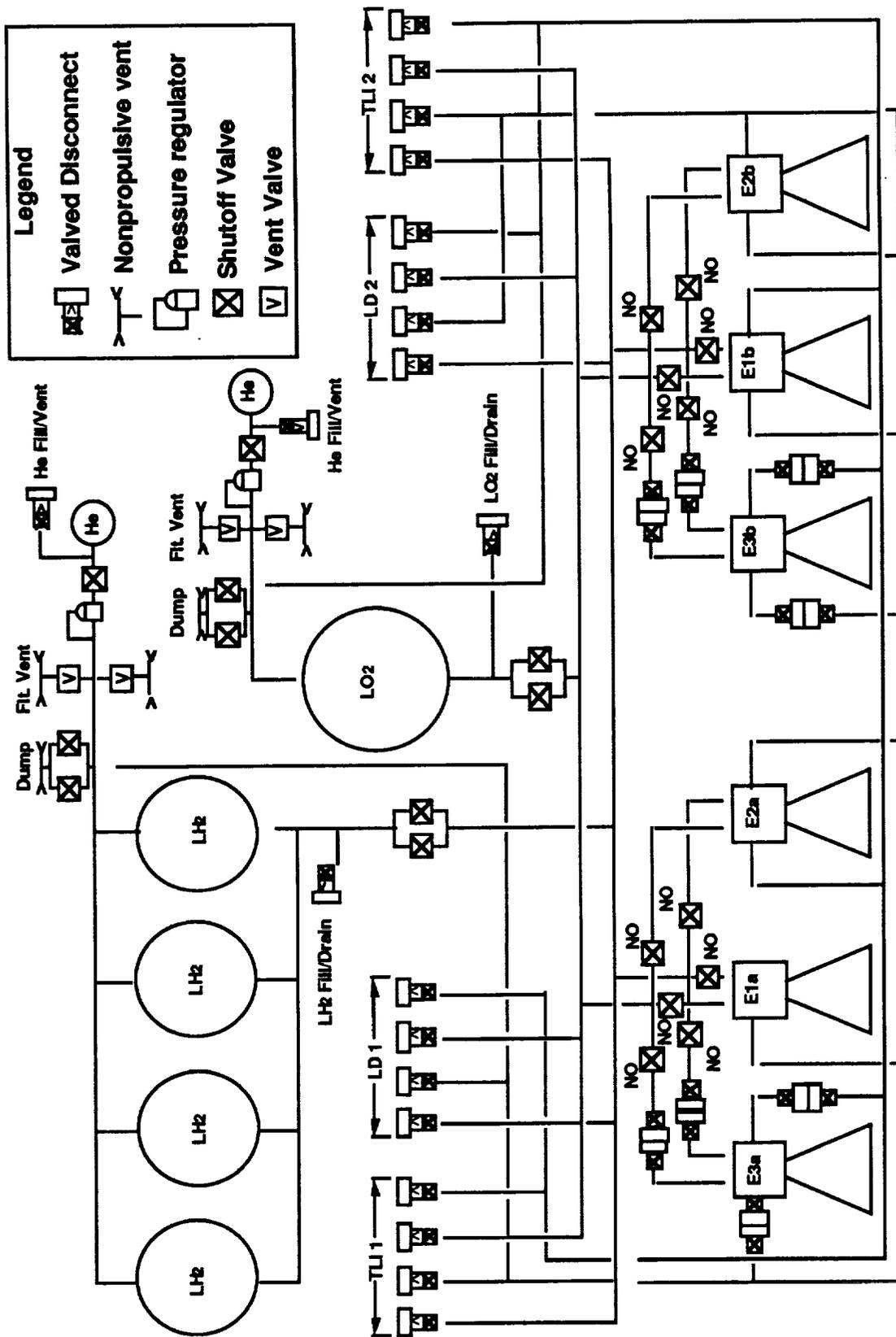
Figure 1-2.4.5-3. Space-Based Core Main Propulsion Schematic

Item	Qty	Mass (kg)	Remarks
CORE STAGE MAIN PROPULSION		2126	
MAIN ENGINES	6	854	Advanced engines, 15 klbt ea., (481 sec Isp) — no nozzle retract
ENGINE ANCILLARY EQUIP		73	
TVC ACTUATORS	12	51	
TVC ACTUATOR SUPT/ INSTL	12	21	
LH2 FEED		393	
DISCONNECT - 6 in	3	41	SELF CLOSING, RISE-OFF DISCONNECT ELECTROMECHANICAL VALVE
PREVALVE - 6 in	6	41	
DUCT ASSEMBLY - 6 in.	1	244	L~90 ft total @ 6 lb/ft, incl tank transfer lines — ALUMINUM
DUCT ASSEMBLY - 2.5 in	6	38	L~7 ft each @ 2 lb/ft, engine feed assemblies — ALUMINUM
INSULATION	1	7	2.5% of manifold mass
SUPPORT / INSTALLATION	1	22	6% of hardware
LH2 FILL, DRAIN		33	
DISCONNECT - 4 in	1	7	SELF CLOSING, RISE-OFF DISCONNECT
VALVES - 4 in	1	4	5.0 IN DIA ELECTROMECHANICAL VALVE — ALUMINUM
DUCT ASSEMBLY - 4 in	1	18	L~10 ft total @ 4 lb/ft
INSULATION	1	1	5% of manifold mass
SUPPORT / INSTALLATION	1	3	10% of hardware
LH2 TANK SPACE VENT, RELIEF		110	
THERMODYNAMIC VENT SYS	4	60	
PARA / ORTHO CATALYST BED	1	10	ALUMINUM
DUCT ASSEMBLY	1	30	
SUPPORT / INSTALLATION	1	10	10% of hardware
LH2 TANK PRESSURIZATION		105	
DISCONNECTS	4	7	SELF CLOSING, RISE-OFF DISCONNECT ELECTROMECHANICAL VALVE
FLOW CONTROL VALVES	18	24	incl tank module press lines
DUCT ASSEMBLY	1	64	10% of hardware — 5.0 IN DIAMETER — ALUMINUM
SUPPORT / INSTALLATION	1	10	
LO2 FEED		334	
DISCONNECT - 6 in	3	41	SELF CLOSING, RISE-OFF DISCONNECT ELECTROMECHANICAL VALVE
PREVALVE - 6 in	6	41	
DUCT ASSEMBLY - 6 in.	1	191	L~70 ft total @ 6 lb/ft, incl tank transfer lines — ALUMINUM
DUCT ASSEMBLY - 2.5 in	6	38	L~7 ft each @ 2 lb/ft, engine feed assemblies — ALUMINUM
INSULATION	1	4	2.0% of manifold mass
SUPPORT / INSTALLATION	1	19	6% of hardware — 5.0 IN DIAMETER
LO2 FILL, DRAIN		33	
DISCONNECT - 4 in	1	7	SELF CLOSING, RISE-OFF DISCONNECT
VALVES - 4 in	1	4	5.0 IN DIA ELECTROMECHANICAL VALVE — ALUMINUM
DUCT ASSEMBLY - 4 in	1	18	L~10 ft total @ 4 lb/ft
INSULATION	1	1	5% of manifold mass
SUPPORT / INSTALLATION	1	3	10% of hardware
LO2 TANK SPACE VENT, RELIEF		88	
THERMODYNAMIC VENT SYS	4	60	
LINES, FTGS, ETC	1	20	10% of hardware — 2.0 IN DIAMETER — ALUMINUM
SUPPORT / INSTALLATION	1	8	
LO2 TANK PRESSURIZATION		105	
DISCONNECTS	4	7	SELF CLOSING, RISE-OFF DISCONNECT ELECTROMECHANICAL VALVE
FLOW CONTROL VALVES	18	24	incl tank module press lines
DUCT ASSEMBLY	1	64	10% of hardware — ALUMINUM
SUPPORT / INSTALLATION	1	10	

Figure 1-2.4.5-4. Space-Based Core Stage Main Propulsion

Item	Qty	Mass (kg)	Remarks
DESCENT TANKSET MAIN PROPULSION		257	
LH2 FEED, FILL, DRAIN (6-IN)	1	14	
DISCONNECT / CONTROL VALVE	1	44	
DUCT ASSEMBLY - 6 IN		58	16 FT @ 6 LB/FT, VACUUM-JACKETED — ALUMINUM
LH2 TANK SPACE VENT, RELIEF	1	20	
THERMODYNAMIC VENT SYS	1	0	
PARA / ORTHO CATALYST BED	1	10	
DUCT ASSEMBLY, NON-PROP VENT		30	INCLUDED IN CORE TANK ONLY — ALUMINUM
LH2 TANK GROUND VENT, RELIEF	1	3	SELF CLOSING, RISE-OFF DISCONNECT
DISCONNECT / VENT VALVE	1	20	
DUCT ASSEMBLY		23	SELF CLOSING, RISE-OFF DISCONNECT — ALUMINUM
LH2 TANK PRESSURIZATION	1	2	2.0 IN DIAMETER
DISCONNECTS	3	4	SELF CLOSING, RISE-OFF DISCONNECT
VALVES	1	10	ELECTROMECHANICAL VALVE
DUCT ASSEMBLY		16	ALUMINUM
LO2 FEED, FILL, DRAIN (6 IN)	1	14	
DISCONNECT / CONTROL VALVE	1	24	
DUCT ASSEMBLY - 6 IN		38	9 FT @ 6 LB/FT, VACUUM-JACKETED — ALUMINUM
LO2 TANK SPACE VENT, RELIEF	1	20	
THERMODYNAMIC VENT SYS	1	10	
DUCT ASSEMBLY, NON-PROP VENT		30	ALUMINUM
LO2 TANK GROUND VENT, RELIEF	1	3	SELF CLOSING, RISE-OFF DISCONNECT
DISCONNECT / VENT VALVE	1	20	
DUCT ASSEMBLY		23	ALUMINUM
LO2 TANK PRESSURIZATION	1	2	2.0 IN DIAMETER
DISCONNECTS	3	4	SELF CLOSING, RISE-OFF DISCONNECT
VALVES	1	10	ELECTROMECHANICAL VALVE
DUCT ASSEMBLY			ALUMINUM
SUPPORT / INSTALLATION		23	10 % OF HARDWARE

Figure 1-2.4.5-5. Space-Based Descent Tankset Main Propulsion

Item	Qty	Mass (kg)	Remarks
TLI TANKSET MAIN PROPULSION		393	
LH2 FEED, FILL, DRAIN (6-IN)			
DISCONNECT / CONTROL VALVE	1	14	ALUMINUM
DUCT ASSEMBLY - 6 IN	1	80	22 FT @ 8 LB/FT, VACUUM-JACKETED / ALUMINUM
LH2 TANK SPACE VENT, RELIEF			
THERMODYNAMIC VENT SYS	1	20	
PARA / ORTHO CATALYST BED	1	0	INCLUDED IN CORE TANK ONLY
DUCT ASSEMBLY, NON-PROP VENT	1	10	ALUMINUM
LH2 TANK GROUND VENT, RELIEF			
DISCONNECT / VENT VALVE	1	3	SELF CLOSING, RISE-OFF DISCONNECT
DUCT ASSEMBLY	1	20	ALUMINUM
LH2 TANK PRESSURIZATION			
DISCONNECTS	1	2	2.0 IN DIAMETER
VALVES	3	4	SELF CLOSING, RISE-OFF DISCONNECT ELECTROMECHANICAL VALVE
DUCT ASSEMBLY	1	48	ALUMINUM
LO2 FEED, FILL, DRAIN (6 IN)			
DISCONNECT / CONTROL VALVE	1	14	
DUCT ASSEMBLY - 6 IN	1	44	12 FT @ 8 LB/FT, VACUUM-JACKETED / ALUMINUM
LO2 TANK SPACE VENT, RELIEF			
THERMODYNAMIC VENT SYS	1	20	
DUCT ASSEMBLY, NON-PROP VENT	1	10	ALUMINUM
LO2 TANK GROUND VENT, RELIEF			
DISCONNECT / VENT VALVE	1	3	SELF CLOSING, RISE-OFF DISCONNECT
DUCT ASSEMBLY	1	20	ALUMINUM
LO2 TANK PRESSURIZATION			
DISCONNECTS	1	2	2.0 IN DIAMETER
VALVES	3	4	SELF CLOSING, RISE-OFF DISCONNECT ELECTROMECHANICAL VALVE
DUCT ASSEMBLY	1	39	ALUMINUM
SUPPORT / INSTALLATION		36	10 % OF HARDWARE

Figure 1-2.4.5-6. Space-Based TLI Tankset Main Propulsion

ITEM	QTY	Mass (kg)	REMARKS (ENGLISH UNITS)	
MAIN PROPULSION - ASCENT TANKAGE		**376**		
LH2 FEED		39		
DISCONNECT - 6 in	1	14	SELF CLOSING, RISE-OFF DISCONNECT	
VALVE - 6 in	1	7	ELECTROMECHANICAL VALVE	
DUCT ASSEMBLY - 6 in.	1	15	L=5 ft total @ 6 lb/ft, incl tank transfer lines	ALUMINUM
INSULATION	1	0	2.5% of manifold mass	
SUPPORT / INSTALLATION	1	2	6% of hardware	
LH2 TANK VENT, RELIEF		50		
THERMODYNAMIC VENT SYS	1	15		
PARA / ORTHO CATALYST BED	1	10		ALUMINUM
DUCT ASSEMBLY	1	20		
SUPPORT / INSTALLATION	1	5	10% of hardware	
LH2 TANK PRESSURIZATION		73		
DISCONNECTS	2	2	SELF CLOSING, RISE-OFF DISCONNECT	
FLOW CONTROL VALVES	18	24	ELECTROMECHANICAL VALVE	ALUMINUM
DUCT ASSEMBLY	1	40	10% of hardware	
SUPPORT / INSTALLATION	1	7	5.0 IN DIAMETER	
LO2 FEED		104		
DISCONNECT - 6 in	1	14	SELF CLOSING, RISE-OFF DISCONNECT	
VALVE - 6 in	1	7	ELECTROMECHANICAL VALVE	
DUCT ASSEMBLY - 6 in.	1	76	L=28 ft total @ 6 lb/ft, incl tank transfer lines	ALUMINUM
INSULATION	1	2	2.0% of manifold mass	
SUPPORT / INSTALLATION	1	6	6% of hardware	
LO2 TANK VENT, RELIEF		39		
THERMODYNAMIC VENT SYS	1	15		
LINES, FTGS, ETC	1	20		ALUMINUM
SUPPORT / INSTALLATION	1	4	10% of hardware	
LO2 TANK PRESSURIZATION		73		
DISCONNECTS	2	2	SELF CLOSING, RISE-OFF DISCONNECT	
FLOW CONTROL VALVES	18	24	ELECTROMECHANICAL VALVE	ALUMINUM
DUCT ASSEMBLY	1	40	incl tank module press lines	
SUPPORT / INSTALLATION	1	7	10% of hardware	

Figure 1-2.4.5-7. Ground-Based Ascent Tankset Main Propulsion

BOEING

ITEM	QTY	Mass (kg)	REMARKS (ENGLISH UNITS)
MAIN PROPULS - ASCENT PROP MODULE		**917**	
MAIN ENGINES	4	570	Advanced engines, 15 klbt ea., (481 sec Isp) — no nozzle retract
ENGINE ANCILLARY EQUIP		48	
TVC ACTUATORS	8	34	
TVC ACTUATOR SUPT/INSTL	8	14	
LH2 FEED		107	
DISCONNECT - 6 in	1	41	SELF CLOSING, RISE-OFF DISCONNECT
PREVALVE - 6 in	4	28	ELECTROMECHANICAL VALVE
DUCT ASSEMBLY - 6 in.	1	15	L=5 ft total @ 6 lb/ft, incl tank transfer lines — ALUMINUM
DUCT ASSEMBLY - 2.5 in	4	16	L=4 ft each @ 2 lb/ft, engine feed assemblies — ALUMINUM
INSULATION	1	1	2.5% of manifold mass
SUPPORT / INSTALLATION	1	6	6% of hardware
LH2 TANK PRESSURIZATION		13	
DISCONNECTS	1	2	SELF CLOSING, RISE-OFF DISCONNECT
DUCT ASSEMBLY	1	10	10% of hardware, 5.0 IN DIAMETER — ALUMINUM
SUPPORT / INSTALLATION	1	1	
LO2 FEED		106	
DISCONNECT - 6 in	1	41	SELF CLOSING, RISE-OFF DISCONNECT
PREVALVE - 6 in	4	28	ELECTROMECHANICAL VALVE
DUCT ASSEMBLY - 6 in.	1	15	L=5 ft total @ 6 lb/ft, incl tank transfer lines — ALUMINUM
DUCT ASSEMBLY - 2.5 in	4	16	L=4 ft each @ 2 lb/ft, engine feed assemblies — ALUMINUM
INSULATION	1	0	2.0% of manifold mass
SUPPORT / INSTALLATION	1	6	6% of hardware
LO2 TANK PRESSURIZATION		73	
DISCONNECTS	1	2	SELF CLOSING, RISE-OFF DISCONNECT, 2.0 IN DIAMETER
DUCT ASSEMBLY	1	64	incl tank module press lines — ALUMINUM
SUPPORT / INSTALLATION	1	7	10% of hardware

Figure 1-2.4.5-8. Ground-Based Propulsion Module Main Propulsion

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ITEM	QTY	Mass (kg)		REMARKS (ENGLISH UNITS)	
MAIN PROPULSION - LANDER					
LH2 FEED			828		
DISCONNECT - 6 in	5	68	300	SELF CLOSING, RISE-OFF DISCONNECT	ALUMINUM
TANK ISOLATION VALVE - 6 in	5	35		ELECTROMECHANICAL VALVE	
DUCT ASSEMBLY - 6 in.	1	176		L=65 ft total @ 6 lb/ft, incl tank transfer lines	
INSULATION	1	4		2.5% of manifold mass	
SUPPORT / INSTALLATION	1	17		6% of hardware	
LH2 FILL DRAIN			33		
DISCONNECT - 4 in	1	7		SELF CLOSING, RISE-OFF DISCONNECT	ALUMINUM
VALVES - 4 in	1	4		5.0 IN DIA ELECTROMECHANICAL VALVE	
DUCT ASSEMBLY - 4 in	1	18		L=10 ft total @ 4 lb/ft	
INSULATION	1	1		5% of manifold mass	
SUPPORT / INSTALLATION	1	3		10% of hardware	
LH2 TANK PRESSURIZATION			81		
DISCONNECTS	5	10		SELF CLOSING, RISE-OFF DISCONNECT	ALUMINUM
DUCT ASSEMBLY	1	64		incl tank module press lines	
SUPPORT / INSTALLATION	1	7		10% of hardware	
LO2 FEED			299	5.0 IN DIAMETER	
DISCONNECT - 6 in	5	68		SELF CLOSING, RISE-OFF DISCONNECT	ALUMINUM
TANK ISOLATION VALVE - 6 in	5	35		ELECTROMECHANICAL VALVE	
DUCT ASSEMBLY - 6 in.	1	176		L=65 ft total @ 6 lb/ft, incl tank transfer lines	
INSULATION	1	4		2.0% of manifold mass	
SUPPORT / INSTALLATION	1	17		6% of hardware	
LO2 FILL DRAIN			33	5.0 IN DIAMETER	
DISCONNECT - 4 in	1	7		SELF CLOSING, RISE-OFF DISCONNECT	ALUMINUM
VALVES - 4 in	1	4		5.0 IN DIA ELECTROMECHANICAL VALVE	
DUCT ASSEMBLY - 4 in	1	18		L=10 ft total @ 4 lb/ft	
INSULATION	1	1		5% of manifold mass	
SUPPORT / INSTALLATION	1	3		10% of hardware	
LO2 TANK PRESSURIZATION			81	2.0 IN DIAMETER	
DISCONNECTS	5	10		SELF CLOSING, RISE-OFF DISCONNECT	ALUMINUM
DUCT ASSEMBLY	1	64		incl tank module press lines	
SUPPORT / INSTALLATION	1	7		10% of hardware	

Figure 1-2.4.5-9. Ground-Based Lander Main Propulsion

ITEM	QTY	Mass (kg)	REMARKS (ENGLISH UNITS)
MAIN PROPULSION - DELIVERY STAGE (1)		545	
MAIN ENGINES	1	143	Advanced engines, 15 klbt ea., (481 sec Isp)
ENGINE ANCILLARY EQUIP		12	
TVC ACTUATORS	2		
TVC ACTUATOR SUPT/ INSTL	2		
LH2 FEED		110	
DISCONNECT - 6 in	1	14	SELF CLOSING, RISE-OFF DISCONNECT ELECTROMECHANICAL VALVE
VALVE - 6 in	1	7	
DUCT ASSEMBLY - 6 in.	1	76	L=28 ft total @ 6 lb/ft, incl tank transfer lines
DUCT ASSEMBLY - 2.5 in.	1	5	L=5 ft total @ 2 lb/ft, engine feed assemblies
INSULATION	1	2	2.5% of manifold mass
SUPPORT / INSTALLATION	1	6	6% of hardware
LH2 TANK VENT, RELIEF		50	
THERMODYNAMIC VENT SYS	1	15	
PARA / ORTHO CATALYST BED	1	10	
DUCT ASSEMBLY	1	20	ALUMINUM
SUPPORT / INSTALLATION	1	5	10% of hardware
LH2 TANK PRESSURIZATION		73	
DISCONNECTS	1	2	SELF CLOSING, RISE-OFF DISCONNECT
FLOW CONTROL VALVES	18	24	ELECTROMECHANICAL VALVE
DUCT ASSEMBLY	1	40	10% of hardware ALUMINUM
SUPPORT / INSTALLATION	1	7	5.0 IN DIAMETER
LO2 FEED		48	
DISCONNECT - 6 in	1	14	SELF CLOSING, RISE-OFF DISCONNECT ELECTROMECHANICAL VALVE
VALVE - 6 in	1	7	
DUCT ASSEMBLY - 6 in.	1	19	L=7 ft total @ 6 lb/ft, incl tank transfer lines
DUCT ASSEMBLY - 2.5 in.	1	5	L=5 ft total @ 2 lb/ft, engine feed assemblies
INSULATION	1	0	2.0% of manifold mass
SUPPORT / INSTALLATION	1	3	6% of hardware
LO2 TANK VENT, RELIEF		39	
THERMODYNAMIC VENT SYS	1	15	
LINES, FTGS, ETC	1	20	10% of hardware
SUPPORT / INSTALLATION	1	4	2.0 IN DIAMETER
LO2 TANK PRESSURIZATION		73	SELF CLOSING, RISE-OFF DISCONNECT
DISCONNECTS	1	2	ELECTROMECHANICAL VALVE
FLOW CONTROL VALVES	18	24	incl tank module press lines
DUCT ASSEMBLY	1	40	ALUMINUM
SUPPORT / INSTALLATION	1	7	10% of hardware

Additional REMARKS column entries (aligned right side):
no nozzle retract; ALUMINUM; ALUMINUM; ALUMINUM; ALUMINUM; ALUMINUM; ALUMINUM; ALUMINUM

Figure 1-2.4.5-10. Ground-Based Delivery Stage Main Propulsion

ITEM	QTY	Mass (kg)	REMARKS (ENGLISH UNITS)
MAIN PROPULSION - TLI TANKSET (1)		380	
LH2 FEED		104	
DISCONNECT - 6 in	1	14	SELF CLOSING, RISE-OFF DISCONNECT
VALVE - 6 in	1	7	ELECTROMECHANICAL VALVE
DUCT ASSEMBLY - 6 in.	1	76	L=28 ft total @ 6 lb/ft, incl tank transfer lines / ALUMINUM
INSULATION	1	1	2.5% of manifold mass
SUPPORT / INSTALLATION	1	6	6% of hardware
LH2 TANK VENT, RELIEF		50	
THERMODYNAMIC VENT SYS	1	15	
PARA / ORTHO CATALYST BED	1	10	
DUCT ASSEMBLY	1	20	ALUMINUM
SUPPORT / INSTALLATION	1	5	10% of hardware
LH2 TANK PRESSURIZATION		73	
DISCONNECTS	2	2	SELF CLOSING, RISE-OFF DISCONNECT
FLOW CONTROL VALVES	18	24	ELECTROMECHANICAL VALVE
DUCT ASSEMBLY	1	40	10% of hardware / 5.0 IN DIAMETER / ALUMINUM
SUPPORT / INSTALLATION	1	7	
LO2 FEED		43	
DISCONNECT - 6 in	1	14	SELF CLOSING, RISE-OFF DISCONNECT
VALVE - 6 in	1	7	ELECTROMECHANICAL VALVE
DUCT ASSEMBLY - 6 in.	1	19	L=7 ft total @ 6 lb/ft, incl tank transfer lines / ALUMINUM
INSULATION	1	0	2.0% of manifold mass
SUPPORT / INSTALLATION	1	2	6% of hardware
LO2 TANK VENT, RELIEF		39	
THERMODYNAMIC VENT SYS	2	15	
LINES, FTGS, ETC	1	20	
SUPPORT / INSTALLATION	1	4	10% of hardware / ALUMINUM
LO2 TANK PRESSURIZATION		73	
DISCONNECTS	1	2	2.0 IN DIAMETER / SELF CLOSING, RISE-OFF DISCONNECT
FLOW CONTROL VALVES	18	24	ELECTROMECHANICAL VALVE / incl tank module press lines
DUCT ASSEMBLY	1	40	ALUMINUM
SUPPORT / INSTALLATION	1	7	10% of hardware

Figure 1-2.4.5-11. Ground-Based TLI Tankset Main Propulsion

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Engine Ancillary Equipment. 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 system.

Propellant Feed. Propellant feedlines transfer propellant from the core tanks, descent tanks, and TLI tanks through tank isolation valves and disconnects to a common manifold, through engine isolation valves, and into the main engines. The feedlines are vacuum-jacketed, insulated stainless steel lines and include restrained bellows joints that articulate to compensate for thermal contraction and engine gimbal motion. Tank feedlines and manifold are 6.0 inches in diameter for both LO2 and LH2 and engine feedlines are 2.5 inches in diameter. Tank isolation valves are electromechanically actuated normally-closed valves; pre-valves are normally-open electromechanically actuated closed; 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.

Propellant Fill and Drain. 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.

Tank Vent and Relief. Two separate tank vent and relief systems are provided: one for use when stowed in the launch vehicle fairing and one for use

in space. 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. Figure 1-2.4.5-12 shows a typical heat exchanger and mixer pump unit. Not shown are the valves and lines that route the vented vapor to space.

Because the core stage tanks store propellants on the lunar surface for up to 6 months, the core stage hydrogen TVS vent line is equipped with a catalyst bed for converting hydrogen from its low-temperature equilibrium state in which all the hydrogen is in the para state, to its higher temperature equilibrium state in which up to 75% of the hydrogen is in the higher energy ortho state. This para-to-ortho conversion extracts more energy from the vented hydrogen, reducing the boiloff rate.

Tank Pressurization. Pressurization for the main propellant tanks is autogenous and consists of plumbing for delivery of pressurization gases (GH2 and GO2) from the engine-mounted bleed ports through a manifold and disconnects to the individual propellant tanks.

Vent inlet
Mixer outlet
Vent outlet
Mixer inlet
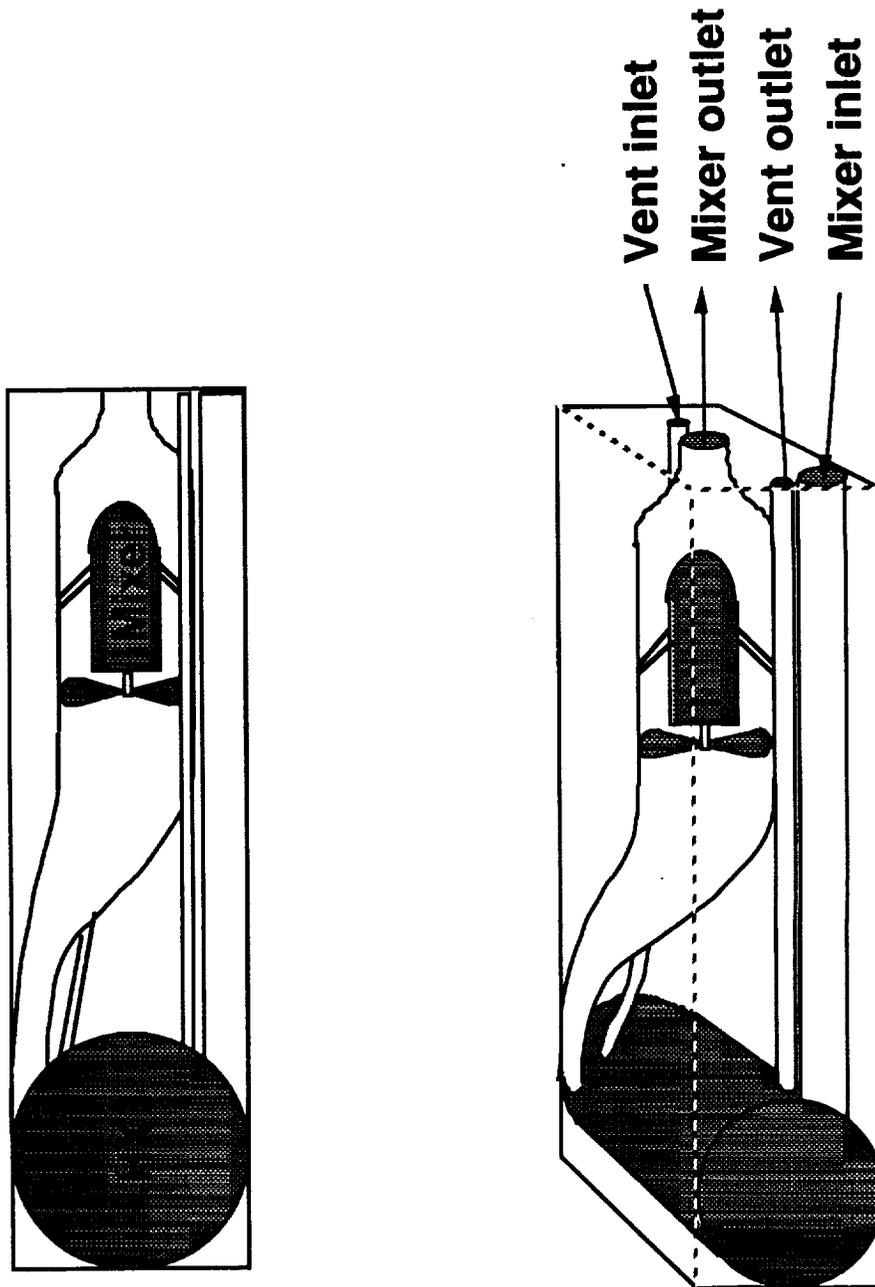
Figure 1-2.4.5-12.   TVS Mixing Vent

1-2.4.6   Reaction Control

The reaction control system provides attitude control during coast periods, rendezvous and docking maneuvers, lunar landing, and atmosphere reentry and provides limited ΔV capability.

Requirements. Top-level requirements for the reaction control subsystem are given in Figure 1-2.4.6-1.

Hardware Description. The system selected for this study is a supercritical GO2/GH2 system supplied from accumulators that are filled with liquid from the main propellant tanks. The space-based STV concept includes a single reaction control system located on the core stage. A schematic is shown in Figure 1-2.4.6-2, and detail weights are given in Figure 1-2.4.6-3.

The ground-based vehicle has two reaction control systems, one on the lunar delivery segment (located on the delivery stages) used for lunar transit and landing and another on the crew module used for ascent, Earth return, and crew module reentry. Detail weights are shown in Figure 1-2.4.6-4 for the delivery stage, and Figure 1-2.4.6-5 for the crew module.

RCS Thrusters. There are a total of sixteen 80-lbf thrusters on the space-based configuration and twenty 75-lbf thrusters on the outbound side of the ground-based vehicle. There are twenty 25-lbf thrusters on the Earth return side of the ground-based vehicle. These LOX/LH2 thrusters are similar in size and configuration as those developed by MSFC during phase I of the Space Station program.

Propellant Supply. The RCS propellant supply includes thruster supply manifolds and valves, two sets of propellant accumulators, and accumulator fill and drain lines and valves. Liquid oxygen and hydrogen are drawn into one set of accumulators from the main propellant feedlines during main engine burns, then isolated and heated to supercritical pressure for use in the reaction control system, electrical power fuel cells, and atmosphere pressurization. During depletion of one set of accumulators, the other set are vented and refilled.

Reaction Control	Basing		Requirement Source		
	Space	Ground	Given	Derived	Provis.
Space station proximity operations capability	X				
Satisfy man-rating requirements	X	X	X	X	
Control orientation, provide initial pointing for main propulsion burn	X	X			X
Control orientation in case of main engine-out event	X	X			X
Provide six degree-of-freedom capability	X	X			X

Figure 1-2.4.6-1. Reaction Control Design Requirements
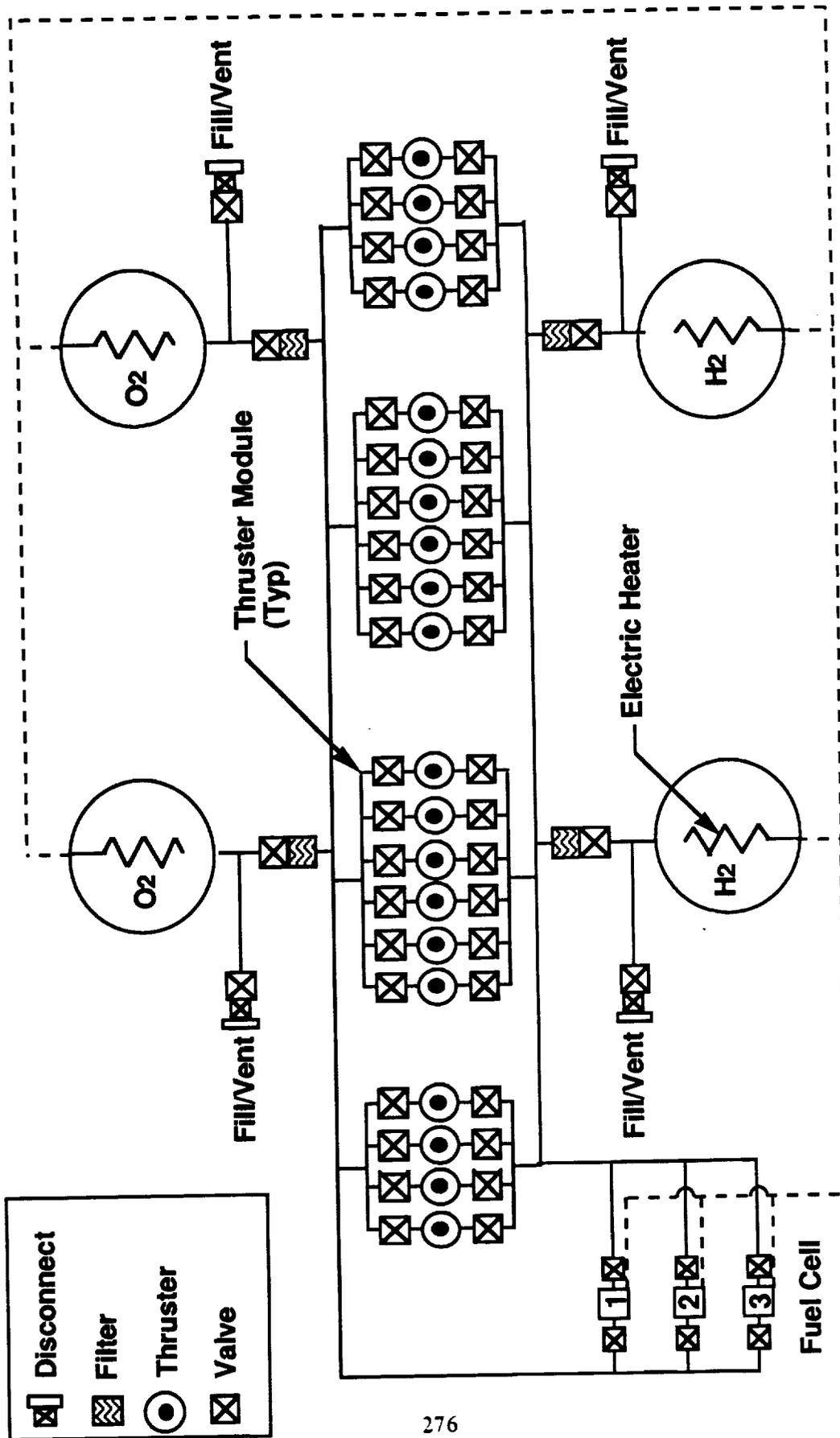
Figure 1-2.4.6-2. Reaction Control System Schematic

D180-32040-2

BOEING

Item	Qty	Mass (kg)	Remarks
CORE STAGE REACTION CONTROL		347	GO2 / GH2 SYSTEM SYSTEM
THRUSTERS - MAIN	24	120	250 LBF THRUSTERS
THRUSTERS - PROX OPS	12	36	25 LBF THRUSTERS
PRESSURIZATION SYSTEM		0	NOT APPLICABLE
TANKAGE	0	0	
VALVES / REGULATORS	0	0	
MANIFOLD/PLUMBING	0	0	
PROPELLANT SUPPLY - MAIN RCS		165	USE FROM MAIN TANKS
TANKAGE	0	0	
ACCUMULATORS - FUEL	2	74	42 in dia supercrit storage , no vacuum jacket aluminum
ACCUMULATORS - OXIDIZER	2	31	29 in dia supercrit storage , no vacuum jacket Inconel 718
SUPPLY VALVES / REGULATORS	8	8	
SUPPLY MANIFOLD/PLUMBING	2	24	SEPARATE O2/H2 MANIFOLDS
FILL / DRAIN VALVES	8	4	
TANK FILL / DRAIN MANIFOLD	2	12	
VENT / RELIEF VALVES	8	4	
TANK VENT / RELIEF MANIFOLD	4	8	
RCS SUPT/INSTL		26	8 % OF RCS

Figure 1-2.4.6-3.   Space-Based Reaction Control System

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ITEM	QTY	Mass (kg)	REMARKS (ENGLISH UNITS)
REACTION CONTROL - DELIVERY STAGE (1)		**190**	**GO2 / GH2 SYSTEM SYSTEM**
THRUSTERS - MAIN	12	60	250 LBF THRUSTERS
THRUSTERS - PROX-OPS	0	0	NOT REQUIRED
PRESSURIZATION SYSTEM	0	0	NOT REQUIRED - SUPERCRITICAL STORAGE
PROPELLANT SUPPLY - MAIN RCS		116	
DISCONNECTS - PROPELLANT SUPPLY	0	0	
ACCUMULATORS - FUEL	2	47	33 in dia supercrit storage, no vacuum jacket · aluminum
ACCUMULATORS - OXIDIZER	2	20	23 in dia supercrit storage, no vacuum jacket · Inconel 718
SUPPLY VALVES / REGULATORS	8	8	
SUPPLY MANIFOLD/PLUMBING	2	18	SEPARATE O2/H2 MANIFOLDS
FILL / DRAIN VALVES	8	4	
TANK FILL / DRAIN MANIFOLD	2	8	
VENT / RELIEF VALVES	8	4	
TANK VENT / RELIEF MANIFOLD	4	8	
RCS SUPT/INSTL		14	8 % OF HARDWARE

Figure 1-2.4.6-4. Ground-Based Delivery Stage Reaction Control System

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ITEM	Qty	Mass, kg	REMARKS (ENGLISH UNITS)
REACTION CONTROL - CREW MODULE		162	GO2 / GH2 SYSTEM SYSTEM
THRUSTERS - MAIN	16	64	100 LBF THRUSTERS
PRESSURIZATION SYSTEM	0	0	NOT APPLICABLE
TANKAGE	0	0	
VALVES / REGULATORS	0	0	
MANIFOLD/PLUMBING	0	0	
PROPELLANT SUPPLY - MAIN RCS		86	
TANKAGE	0	0	USE FROM MAIN TANKS
ACCUMULATORS - FUEL	2	20	22 in dia supercrit storage , no vacuum aluminum
ACCUMULATORS - OXIDIZER	2	10	14 in dia supercrit storage , no vacuum Inconel 718
SUPPLY VALVES / REGULATORS	8	8	
SUPPLY MANIFOLD/PLUMBING	2	20	SEPARATE O2/H2 MANIFOLDS
FILL / DRAIN VALVES	8	4	
TANK FILL / DRAIN MANIFOLD	2	12	
VENT / RELIEF VALVES	8	4	
TANK VENT / RELIEF MANIFOLD	4	8	
RCS SUPT/INSTL		12	8 % OF RCS

Figure 1-2.4.6-5.   Ground-Based Crew Module Reaction Control

BOEING

The sizes of the current RCS accumulators are as follows:

1. Space-Based Core Stage - 42-inch-diameter H2 and 29-inch-diameter O2.
2. Ground-Based Crew Module - 22-inch-diameter H2 and 14-inch-diameter O2.
3. Ground-Based Delivery Stages - 33-inch-diameter H2 and 23-inch-diameter O2.

1-2.4.7   Electrical Power

The electrical power system supplies power for electronics and equipment operation, valve control, thrust vector actuation, environmental control, and mechanism actuation.

Requirements. Top-level requirements for the electrical power subsystem are given in Figure 1-2.4.7-1. A summary of typical vehicle power requirements are shown in Figure 1-2.4.7-2. Mission energy requirements for the lunar missions are given in the concept description sequential mass and fluid inventories, in sections 1-2.2 and 1-2.3, respectively. The maximum power usage, during main engine burns, is 7,900 watts for piloted missions and 5,500 watts for unpiloted missions. The average power usage during coast periods is 4,600 watts for piloted missions and 2,100 watts for unpiloted missions.

Hardware Description. The electrical power subsystem hardware includes a power source, distribution and control components, and associated cables and wire harnesses for power distribution. The space-based vehicle power supply is located on the core stage, with interfaces to the crew module and droptanks for power distribution. Figure 1-2.4.7-3 presents detail weights of the space-based electrical power system on the core stage and crew module.

On the ground-based vehicle, the power supply is located on an equipment pallet that remains with the crew module during Earth return. On unpiloted cargo missions, the pallet is attached to the lander to supply vehicle power. Interfaces between the pallet and other flight elements provide power distribution. Weights of the ground-based electrical power system are given in Figure 1-2.4.7-4 for the vehicle elements and Figure 1-2.4.7-5 for the crew module.

Power Supply. The primary power sources for the ground-based and space-based power systems are three 28V dc, 4.6-kilowatt hydrogen/oxygen fuel cells, as shown in Figure 1-2.4.7-6, fed from accumulators that are filled from the core stage main propellant tanks. The fuel cells are derived from the current STS design but will be able to operate on propellant-grade reactants and are reduced in size from the STS design due to lower power requirements. For

Electrical Power	Basing		Requirement Source		
	Space	Ground	Given	Derived	Provis.
Capability to supply power from external support elements	X	X		X	
Satisfy man-rating requirements	X	X	X		
Provide power to all vehicle subsystems	X	X	X		
Vehicle power - TBD kW nom / TBD kW peak	X	X		X	
Provide Payload power - TBD kW nom / TBD kW peak	X	X		X	

Figure 1-2.4.7-1. Electrical Power Design Requirements

BOEING

	Power Required by Mission Phase (Watts)				
	Station Keeping (unmanned)	Main Eng. Firing	Rend. & Dock	Aero-Maneuver	On-orbit Coast
Avionics	650	2692	2692	2712	1857
Guidance & Navigation	80	162	162	162	-
Valve Driver	70	144	144	144	-
Rendezvous & Dock	-	-	-	-	-
Communication, Tracking	-	726	726	726	726
Rescue Radio	-	-	-	-	-
Controls & Displays	-	465	465	465	200
Instrumentation	100	364	364	364	100
Data Management	400	831	831	831	831
Struct. & Mech. Controllers	-	-	-	20	-
ECLSS	250	1900	1900	1900	1900
Propulsion	100	2000	1000	1000	80
RCS Drivers	100	-	1000	1000	80
TVC Actuators	-	2000	-	-	-
Power Loss (20%)	200	1318	1118	1122	767
Total	1200	7910	6710	6734	4604

Figure 1-2.4.7-2. Power Usage Requirements

Item	Qty	Mass (kg)		Remarks
CORE STAGE POWER SUPPLY			374	
POWER SUPPLY			334	
FUEL CELLS	3	218		Reduced Shuttle Cells - 2 of 3 to supply sustained power
O2 TANKAGE (EPS & ECLSS)	0	0		included in RCS
H2 TANKAGE	0	0		included in RCS
REACTANT FILL & DRAIN PLUMBING	0	0		included in RCS
REACTANT RELIEF, VENT PLUMBING	0	0		included in RCS
REACTANT SUPPLY PLUMBING	2	40		
REACTANT SUPPLY VALVES, DISC	6	6		
COOLANT PLUMBING		7		
COOLANT FLUID		14		IN PERSONNEL PROV -- POTABLE WATER STORAGE
WATER STORAGE TANK		0		Rechargeable, 10 kg/kw-hr @ 5 kw-hr
TVC BATTERY	3	50		Ni-H2
POWER SUPPLY SUPT/INSTL			40	12 % OF ELECT PWR
CORE STG POWER DISTR AND CONTROL			433	
POWER DIST EQUIP			137	
LOAD DISTRIBUTION/CNTRL ASSY	3	81		
INVERTERS	9	34		
REMOTE CONTROLS	10	22		
WIRING			250	
ELECTRICAL POWER SUPT/INSTL		46		12 % OF ELECT PWR
CREW MOD POWER DISTR AND CONTROL			272	
POWER DIST EQUIP			84	
POWER DISTRIBUTION PANELS	3	27		ESTIMATE
INVERTERS	3	17		ESTIMATE
COMMODE POWER SUPPLY	1	5		
EXTERIOR LIGHTS		8		
INTERIOR LIGHTS		10		ESTIMATE
POWER DISTRIBUTION SUPT/INSTL		17		
WIRING			188	
POWER DISTR. WIRE HARNESSES		100		25 % OF SYS
INSTRUMENTATION WIRING		30		
ELECTRICAL CONNECTORS		20		BULKHEAD FEEDTHRU PLATES
HARNESS SUPT/INSTL		38		25 % OF SYS

Figure 1-2.4.7-3. Space-Based Electrical Power System

ITEM	QTY	Mass (kg)		REMARKS (ENGLISH UNITS)
POWER SOURCE - PALLET			374	
POWER SUPPLY		334		
FUEL CELLS	3	218		Reduced Shuttle Cells - 2 of 3 to supply sustained power
O2 TANKAGE (EPS & ECLSS)	0	0		included in RCS
H2 TANKAGE	0	0		included in RCS
REACTANT FILL & DRAIN PLUMBING	0	0		included in RCS
REACTANT RELIEF, VENT PLUMBING	0	0		included in RCS
REACTANT SUPPLY PLUMBING	2	40		
REACTANT SUPPLY VALVES, DISC	6	6		
COOLANT PLUMBING		7		
COOLANT FLUID		14		
WATER STORAGE TANK		0		IN PERSONNEL PROV -- POTABLE WATER STORAGE
TVC BATTERY	3	50		Rechargeable, 10 kg/kw-hr @ 5 kw-hr Ni-H2
POWER SUPPLY SUPT/INSTL	3	40		12 % OF HARDWARE
WIRING & ELECT I/F - PALLET			265	
POWER DIST EQUIP		137		
LOAD DISTRIBUTION/CNTRL ASSY	3	81		
INVERTERS	9	34		
REMOTE CONTROLS	10	22		
WIRING		100		
ELECTRICAL POWER SUPT/INSTL	1	28	28	12 % OF HARDWARE
WIRING & ELECT I/F - ASCENT TANKSET	1		28	
WIRING		25		
ELECTRICAL POWER SUPT/INSTL		3		12 % OF HARDWARE
WIRING & ELECT I/F - ASCENT PROP MODULE	1		56	
WIRING		50		
ELECTRICAL POWER SUPT/INSTL		6		12 % OF HARDWARE
WIRING & ELECT I/F EQUIP - LANDER			78	
WIRING		70		
ELECTRICAL POWER SUPT/INSTL		8		12 % OF HARDWARE
WIRING & ELECT I/F - DELIVERY STG	1		39	
WIRING		35		
ELECTRICAL POWER SUPT/INSTL		4		12 % OF HARDWARE
WIRING & ELECT I/F - TLI TANKSET	1		28	
WIRING		25		
ELECTRICAL POWER SUPT/INSTL		3		12 % OF HARDWARE

Figure 1-2.4.7-4.   Ground-Based Vehicle Electrical Power

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ITEM	Qty	Mass, kg	REMARKS (ENGLISH UNITS)
WIRING & ELECT I/F EQUIP - CREW MOD		272	
POWER DIST EQUIP		84	
POWER DISTRIBUTION PANELS	3	27	
INVERTERS	3	17	
COMMODE POWER SUPPLY	1	5	
EXTERIOR LIGHTS		8	ESTIMATE
INTERIOR LIGHTS		10	ESTIMATE
POWER DISTRIBUTION SUPT/INSTL		17	
WIRING		188	
POWER DISTR. WIRE HARNESSES		100	ESTIMATE 25 % OF SYS
INSTRUMENTATION WIRING		30	
ELECTRICAL CONNECTORS		20	BULKHEAD FEEDTHRU PLATES
HARNESS SUPT/INSTL		38	25 % OF SYS

Figure 1-2.4.7-5. Ground-Based Crew Module Electrical Power

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- **Fuel cells provide prime power**
- **28 vdc power**
- **Dual fault fuel cells and feed system**
- **Cells use propellant grade oxygen and hydrogen**
- **Rechargeable batteries provide load leveling**
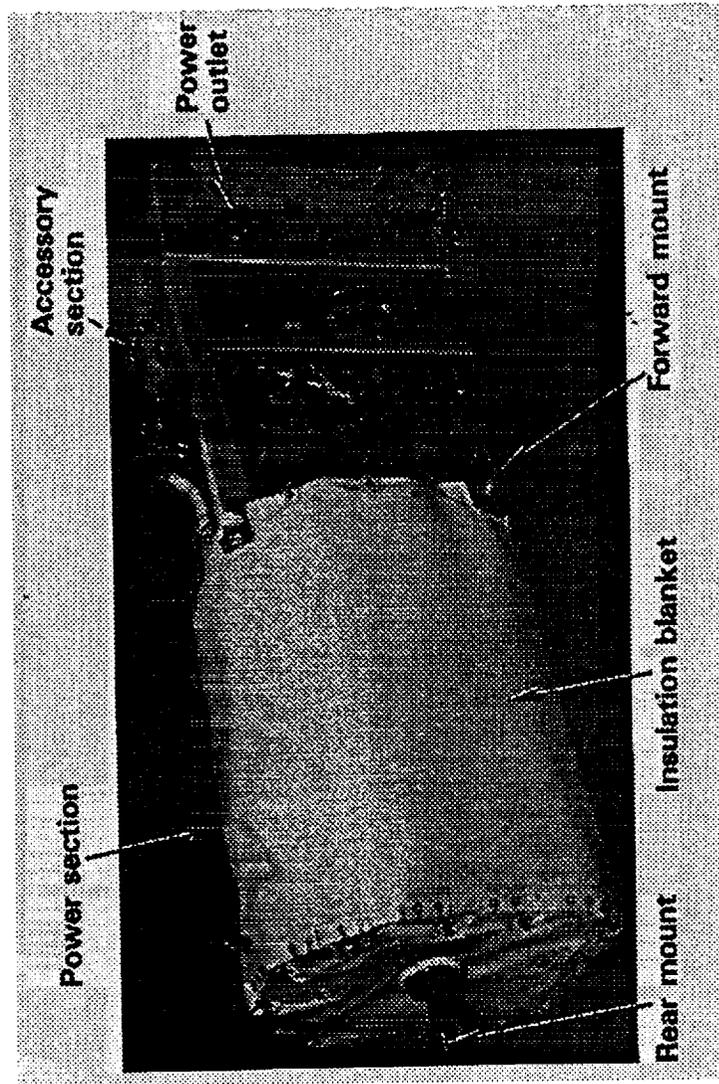
Figure 1-2.4.7-6. Electrical Power Concept

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 current STS orbiter has three fuel cells, each having three stacks of 32 cells with a power output of 7.0-kilowatts nominal and 12-kilowatts peak. All three fuel cells run continuously, providing an power at 21-kilowatts nominal and 36-kilowatts peak. 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 continuously, the total power output is 14.0-kilowatts nominal and 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.

A major issue facing the use of fuel cells is the problem of startup following an extended shutdown period, as would be experienced after a 6-month lunar stay. Currently, efforts are being made to understand this problem and to minimize the impact of making a restartable fuel cell.

Reactant Supply. Fuel cell reactants are drawn from accumulators included in the reaction control subsystem (RCS). The redundant accumulators are sized to provide oxygen and hydrogen reactants for both RCS and electrical power system (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.

Power Distribution and Control. 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.

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Electrical inverters are included to supply 3-phase power to such equipment as main engine actuators and valves, fuel cell controls, and certain environmental control and life support system (ECLSS) components. The inverters are similar to current shuttle inverters.

Wire Harnesses. On the space-based vehicle, the core stage has wire harnesses and interfaces to the crew module and to the droptank sets for power distribution. The crew module wire harnesses distribute power to ECLSS and crew displays and controls, and the droptank wire harnesses distribute power to health monitoring and propellant management equipment.

On the ground-based vehicle, the equipment pallet has wire harnesses and interfaces to the core stage and crew module for power transfer. The crew module wire harnesses distribute power to ECLSS and crew displays and controls, and the droptank and delivery stage wire harnesses distribute power to health monitoring equipment, propellant management equipment, and main engine valves and TVC actuators.

1-2.4.8   Avionics

The avionics subsystem includes provisions for vehicle guidance and navigation, communications, data handling, and piloted controls and displays.

Requirements. Top-level requirements for the combined avionics subsystem are given in Figure 1-2.4.8-1. Additional derived requirements are given in Figure 1-2.4.8-2.

Hardware Description. Figure 1-2.4.8-3 partitions the vehicle avionics into 12 onboard functions: navigation; guidance; attitude control; controls and displays; structures and mechanisms control; telemetry, tracking, and command; mission management; propulsion and critical fluids control; environmental control; power distribution and control; payload services; and vehicle health monitoring. These functions support all phases of ground, flight, on-station, and lunar surface operations, as required for either space-based or ground-based configurations. Figure 1-2.4.8-4 identifies the location of each function for each flight element of the space-based and ground-based configurations. Figure 1-2.4.8-5 shows the function arranged into an architecture.

Detail weights of the space-based avionics are given in Figures 1-2.4.8-6 and 1-2.4.8-7 for the vehicle and crew module, respectively. Weights for the ground-based vehicle and crew module avionics are given in Figures 1-2.4.8-8, 1-2.4.8-9, and 1-2.4.8-10.

Guidance and Navigation. Guidance and navigation equipment provides the means to determine the flight path and attitude of the vehicle throughout the mission, as shown in Figure 1-2.4.8-11. Navigation (NI, NR, NL, ND) computes vehicle position and velocity (six element state vector). Guidance control (GC) provides autonomous trajectory control by adapting to dispersions in thrust, vehicle, and payload cg variations, and unmodeled uncertainties. Attitude control (AC) provides "attitude hold" pointing, attitude rotation from one fixed attitude to another, and fixed rotation rate for mission-unique requirements. Propulsion control (PC) and critical fluids control (FC) accept attitude and velocity commands and provide required valve commands to RCS or OMS engines and valves. Adaptive guidance and control optimizes the trajectory to

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	Requirement Source			Basing	
	Given	Derived	Provis.	Space	Ground
Guidance and Navigation					
Shall be capable of unmanned operations	x			x	x
Provide vehicle attitude, position, and velocity status.					
Communication and Data Handling					
Shall be capable of unmanned operations	x			x	x
Telem., track., command between vehicle and support elements	x			x	x
Provide vehicle health monitoring and status.	x			x	x
Provide built-in test capability to isolate failures.	x			x	x
Provide redundancy management	x			x	x
Controls and Displays					
Notify crew of proper operational / maintenance procedures	x			x	x
Level of crew involvement TBD			x	x	x

Figure 1-2.4.8-1. Avionics Design Requirements

DoD Mission Baseline Will Require Encryption/Decryption And Red Black Communications Architecture

Lunar Surface Support Services For Vehicle Avionics Are Available Within 48 Hours Of Touchdown

Autonomous System Requirements For Insertion Burns, Ascent, Rendezvous, And Landing (Crew Role Limited To Concurrence And Override Functions Not Requiring Piloting Skills)

Man Rating Required For Stv (Two Failure Tolerance For Catastrophic Failures)

Nominal Payload Services
 - 1000 Kilowatt-hour Energy
 - 5 Number Of Monitor And Control Circuits (Jettison, Range Safety)
 - 200 Kbps Telemetry Link

Standard Digital Interfaces To All Major Flight Elements

Provision For Non-destructive On-orbit Mate, De-mate, And Re-mate (Backup And Try Again)

Standardized Interfaces For Orbital Storage While Docked At Space Station Freedom

Avionics From Vehicles Expended On The Lunar Surface Can Be Reused

Gps/Glonass Link Margin Will Be Greater Than 6 Db To Support Mid Course Correction

Technology Freeze Date Of 1996-1997

Figure 1-2.4.8-2. Avionics Groundrules and Assumptions
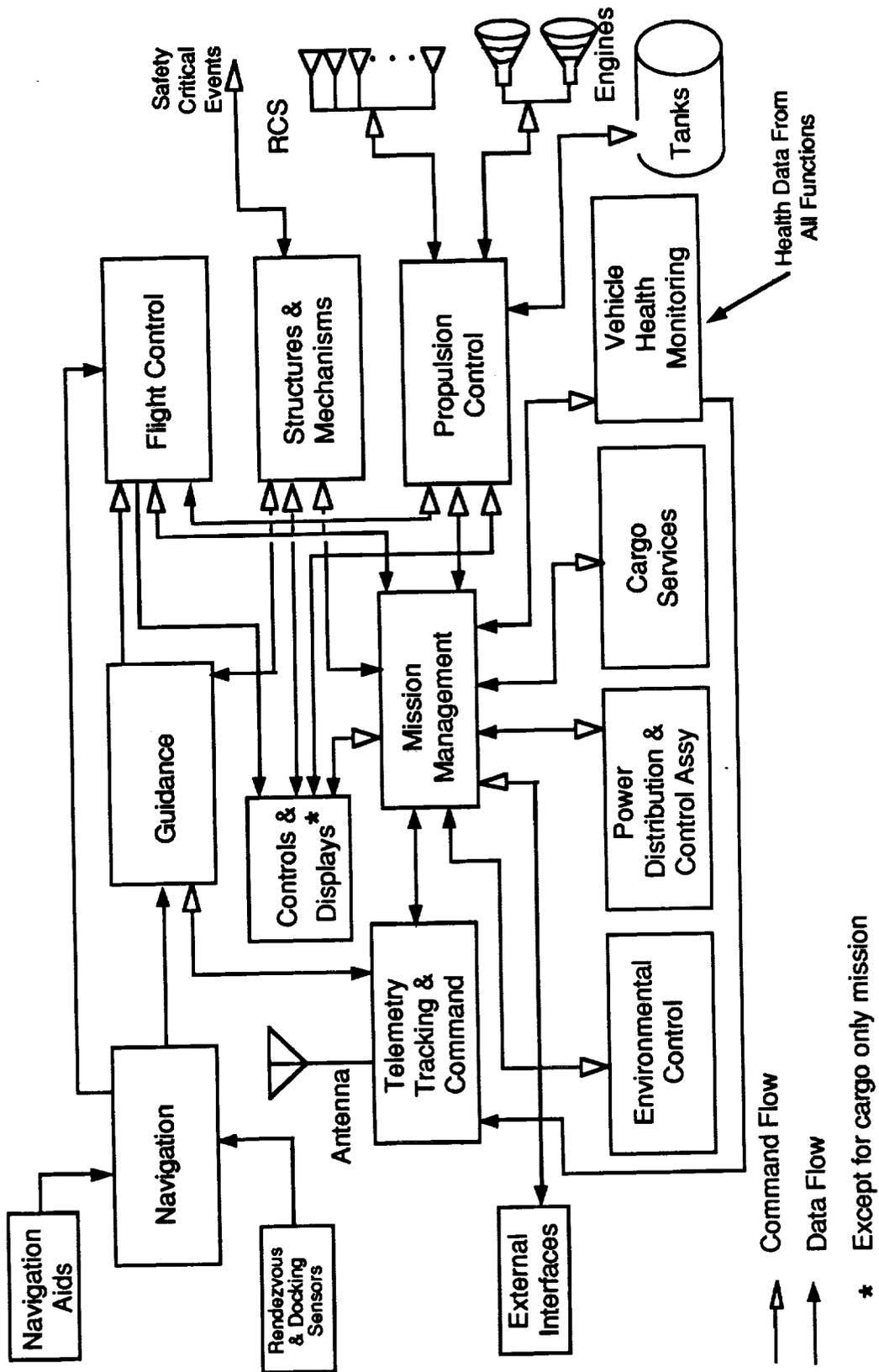
Figure 1-2.4.8-3. Avionics Functional Block Diagram

CONFIGURATION

FLIGHT ELEMENT	Space Based SB1.5S (REFERENCE)		Ground Based		Ground Based, Orbital rendezvous	
	Pilot	Cargo	Pilot	Cargo	Pilot	Cargo
Core Stage TLI, TEI, Descent, Ascent	Vacuum Qual Base Avionics plus: TVC, RCS, Propellant Control	Same as SB Pilot plus Increased Autonomy	TVC, RCS, Propellant Control	Same as GB Pilot	TVC, RCS, Propellant Control	Same as GO Pilot
Crew Module	Controls & Displays Cautions & Warnings	N/A	Base Avionics PALLET; plus RCS, C&D, C&W	Base Avionics PALLET, plus Increased Autonomy	Base Avionics PALLET; plus RCS, C&D, C&W	Base Avionics PALLET, plus Increased Autonomy
Tanks	Tank Sensors and Effectors	Same as SB Pilot	Same as SB Pilot	Same as SB Pilot	Same as SB Pilot	Same as SB Pilot

Figure 1-2.4.8-4. Avionics Function Location Comparison
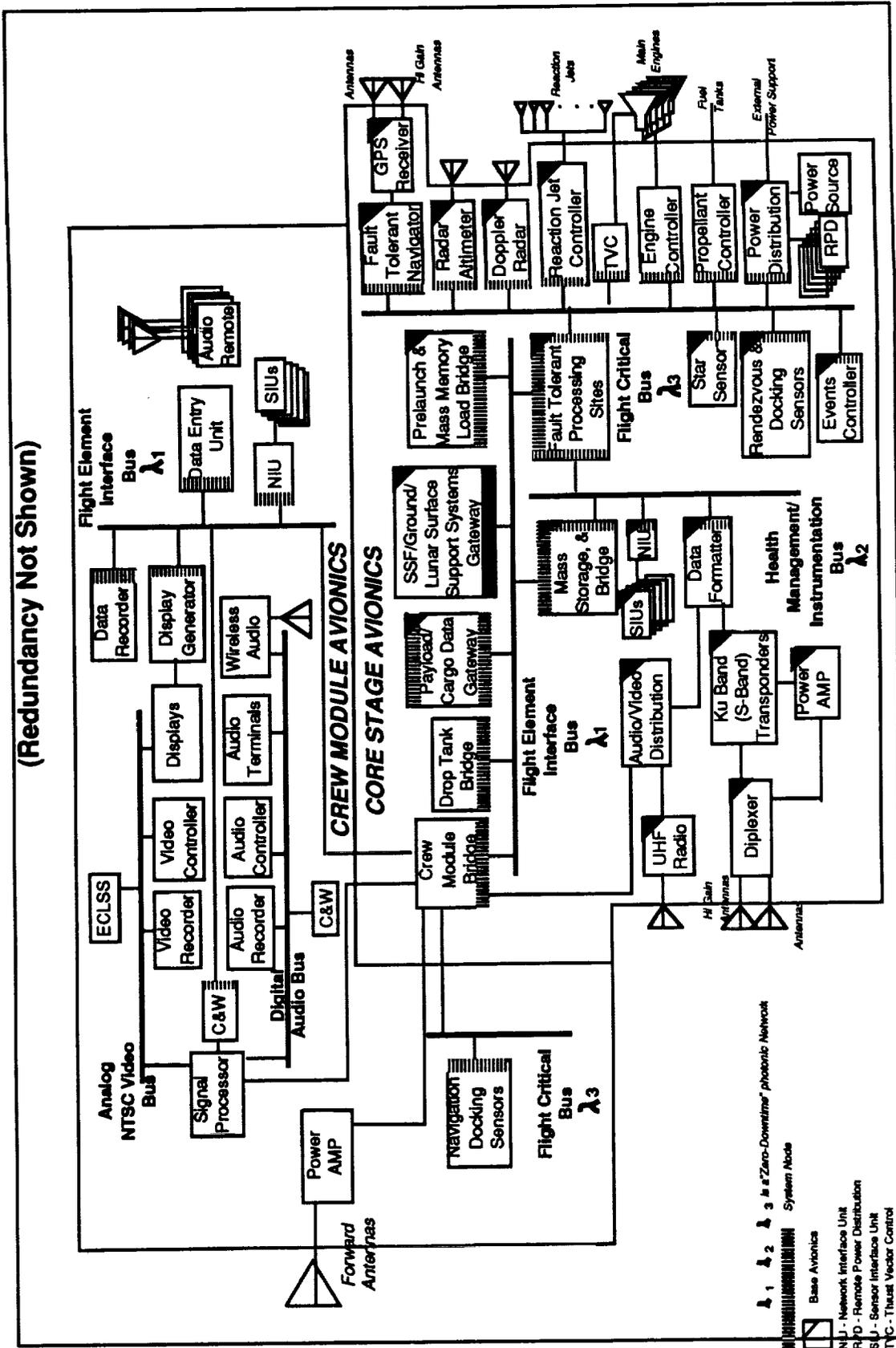
Figure 1-2.4.8-5. Space-Based Avionics Architecture

Item	Qty	Mass (kg)
CORE STAGE G, N, & C		**464**
GUIDANCE AND NAVIGATION		158
FLIGHT IMU	2	44
STAR TRACKER	2	23
SUNSHADE	2	4
OPTICAL BENCH	1	9
GPS RECEIVER, ANTENNAS	1	7
RADAR ALTIMETER	3	4
RANGE, RANGE-RATE RADAR	3	32
DOCKING SENSORS - TELESCOPE, GIMBA	1	5
DOCKING SENSORS - DETECTORS, ETC	2	3
FIBER OPTIC NETWORK	1	27
PROPULSION AND FLUIDS CONTROL		153
TVC CONTROLLER	3	50
RCS CONTROLLER	3	27
VALVE CONTROLLER	3	27
ENGINE CONTROL	6	49
STRUCTURES & MECHS CONTROLLER		111
LASER INITIATORS	5	36
EVENTS CONTROLLER	3	34
VIDEO / LIGHTS - DOCKING	2	41
INSTALLATION / SUPPORT		42

Item	Qty	Mass (kg)
CORE STAGE COMM AND DATA HANDLING		**422**
COMMUNICATIONS AND TRACKING		89
TRANSPONDER	2	24
CMD DESTRUCT EQUIPMENT	2	6
DIPLEXERS	1	2
POWER AMPS	2	12
RF SWITCH	3	1
VIDEO FORMATTER	2	9
ANTENNA - S-BAND	4	3
DATA FORMATTER	2	18
UHF RADIO	2	14
VEHICLE HEALTH MAINTENANCE		129
OPTICAL DISK MEMORY	3	34
SENSOR INTERFACE UNIT	60	31
NETWORK INTERFACE UNIT	4	5
SENSORS	2000	59
DATA HANDLING		166
FAULT TOLERANT PROCESSORS	6	109
MASS MEMORY	3	34
BRIDGE & GATEWAY MISC.	1	23
INSTALLATION / SUPPORT		38

Figure 1-2.4.8-6. Space-Based Core Stage Avionics

Item	Qty	Mass (kg)	
CREW MODULE COMM AND DATA HANDLING			124
COMMUNICATIONS AND TRACKING			65
POWER AMP	2	18	
AUDIO / VIDEO INTERFACE	2	36	
ANTENNAS	3	11	
VEHICLE HEALTH MONITORING			34
MASS MEMORY	3	34	
INSTRUMENTATION			14
SENSOR INTERFACE UNIT (SIU)	30	7	
SENSORS, INSTRUMENTATION	200	7	
SUPT/INSTL			11
CREW MODULE DISPLAYS AND CONTROLS			108
CONTROLS AND DISPLAYS			98
ANNUNCIATOR	2	2	
RECONFIGURABLE DISPLAYS	5	23	
PANEL CONTROL UNITS	2	27	
ELECTRONIC INTERFACES	3	34	
HAND CONTROLLERS	2	14	
SUPT/INSTL			10

Item	Qty	Mass (kg)	
TANKSET G,N,&C			23
FLUIDS CONTROL			23
VALVE CONTROLLERS	2	23	
TANKSET COMM AND DATA HANDLING			39
HEALTH MANAGEMENT			9
MAINTENANCE SENSORS	4	9	
INSTRUMENTATION			30
PROPELLANT GAUGES / ELEX	2	20	
SENSOR INTERFACE UNIT (SIU)	2	2	
NETWORK INTERFACE UNIT (NIU)	25	3	
SENSORS		5	

Figure 1-2.4.8-7. Space-Based Crew Module and Tankset Avionics

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ITEM	QTY	Mass (kg)
G,N,&C - PALLET		**464**
GUIDANCE AND NAVIGATION		158
FLIGHT IMU	2	44
STAR TRACKER	2	23
SUNSHADE	2	4
OPTICAL BENCH	1	9
GPS RECEIVER, ANTENNAS	1	7
RADAR ALTIMETER	3	4
RANGE, RANGE-RATE RADAR	3	32
DOCKING SENSORS - TELESCOPE, GIMBAL	1	5
DOCKING SENSORS - DETECTORS, ETC	2	3
FIBER OPTIC NETWORK	1	27
PROPULSION AND FLUIDS CONTROL		153
TVC CONTROLLER	3	50
RCS CONTROLLER	3	27
VALVE CONTROLLER	3	27
ENGINE CONTROL	6	49
STRUCTURES & MECHS CONTROLLER		111
LASER INITIATORS	5	36
EVENTS CONTROLLER	3	34
VIDEO / LIGHTS - DOCKING	2	41
INSTALLATION / SUPPORT		42
COMM AND DATA - PALLET		**391**
COMMUNICATIONS AND TRACKING		89
TRANSPONDER	2	24
CMD DESTRUCT EQUIPMENT	2	6
DIPLEXERS	1	2
POWER AMPS	2	12
RF SWITCH	3	1
VIDEO FORMATTER	2	9
ANTENNA - S-BAND	4	3
DATA FORMATTER	2	18
UHF RADIO	2	14
VEHICLE HEALTH MAINTENANCE		100
OPTICAL DISK MEMORY	3	34
SENSOR INTERFACE UNIT	60	31
NETWORK INTERFACE UNIT	4	5
SENSORS	1000	30
DATA HANDLING		166
FAULT TOLERANT PROCESSORS	6	109
MASS MEMORY	3	34
BRIDGE & GATEWAY MISC.	1	23
INSTALLATION / SUPPORT		36

Figure 1-2.4.8-8. Ground-Based Pallet Avionics

ITEM	QTY		Mass (kg)	
COMM AND DATA - ASCENT TANKSET				37
INSTRUMENTATION			34	
PROPELLANT GAUGES / ELEX	2	20		
SENSOR INTERFACE UNIT	20	9		
NETWORK INTERFACE UNIT	2	3		
SENSORS	50	2		
INSTALLATION / SUPPORT			3	
COMM AND DATA - ASCENT PROP MODULE				21
HEALTH MANAGEMENT			9	
MAINTENANCE SENSORS		9		
INSTRUMENTATION			11	
SENSOR INTERFACE UNIT (SIU)	4	2		
NETWORK INTERFACE UNIT (NIU)	2	3		
SENSORS	200	6		
INSTALLATION / SUPPORT			1	
COMM AND DATA - LANDER				19
INSTRUMENTATION			17	
SENSOR INTERFACE UNIT	20	9		
NETWORK INTERFACE UNIT	2	3		
SENSORS	200	5		
INSTALLATION / SUPPORT			2	
COMM AND DATA - DELIVERY STG				37
INSTRUMENTATION			34	
PROPELLANT GAUGES / ELEX	2	20		
SENSOR INTERFACE UNIT (SIU)	20	9		
NETWORK INTERFACE UNIT (NIU)	2	3		
SENSORS	50	2		
INSTALLATION / SUPPORT			3	
COMM AND DATA - TLI TANKSET				37
INSTRUMENTATION			34	
PROPELLANT GAUGES / ELEX	2	20		
SENSOR INTERFACE UNIT	20	9		
NETWORK INTERFACE UNIT	2	3		
SENSORS	50	2		
INSTALLATION / SUPPORT			3	

Figure 1-2.4.8-9. Ground-Based Vehicle Avionics

BOEING

ITEM	Qty	Mass, kg	
GUIDANCE, NAV, AND CNTRL - CREW MODULE			130
GUIDANCE AND NAVIGATION			0
PROPULSION AND FLUIDS CONTROL			21
RCS DRIVER	3	21	
STRUCTURES & MECHS CONTROLLER			109
BODY FLAP DRIVER	1	20	
CHUTE, LANDING GEAR CONTROLLER	1	28	
LASER INITIATORS	15	7	
EVENTS CONTROLLER	3	54	
INSTALLATION / SUPPORT			13
COMM AND DATA HANDLING - CREW MODULE			189
COMMUNICATIONS AND TRACKING			83
POWER AMP	2	18	
AUDIO / VIDEO INTERFACE	2	36	
ANTENNAS	3	11	
SEARCH AND RESCUE RADIO	1	18	
VEHICLE HEALTH MONITORING			34
MASS MEMORY	3	34	
INSTRUMENTATION			55
SENSOR INTERFACE UNIT	60	27	
NETWORK INTERFACE UNIT	4	5	
SENSORS	700	23	
DATA HANDLING			0
INSTALLATION / SUPPORT			17
DISPLAYS AND CONTROLS - CREW MODULE			108
CONTROLS AND DISPLAYS			98
ANNUNCIATOR	2	2	
RECONFIGURABLE DISPLAYS	5	23	
PANEL CONTROL UNITS	2	27	
ELECTRONIC INTERFACES	3	34	
HAND CONTROLLERS	2	14	
SUPT/INSTL			10

Figure 1-2.4.8-10. Ground-Based Crew Module Avionics

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BOEING
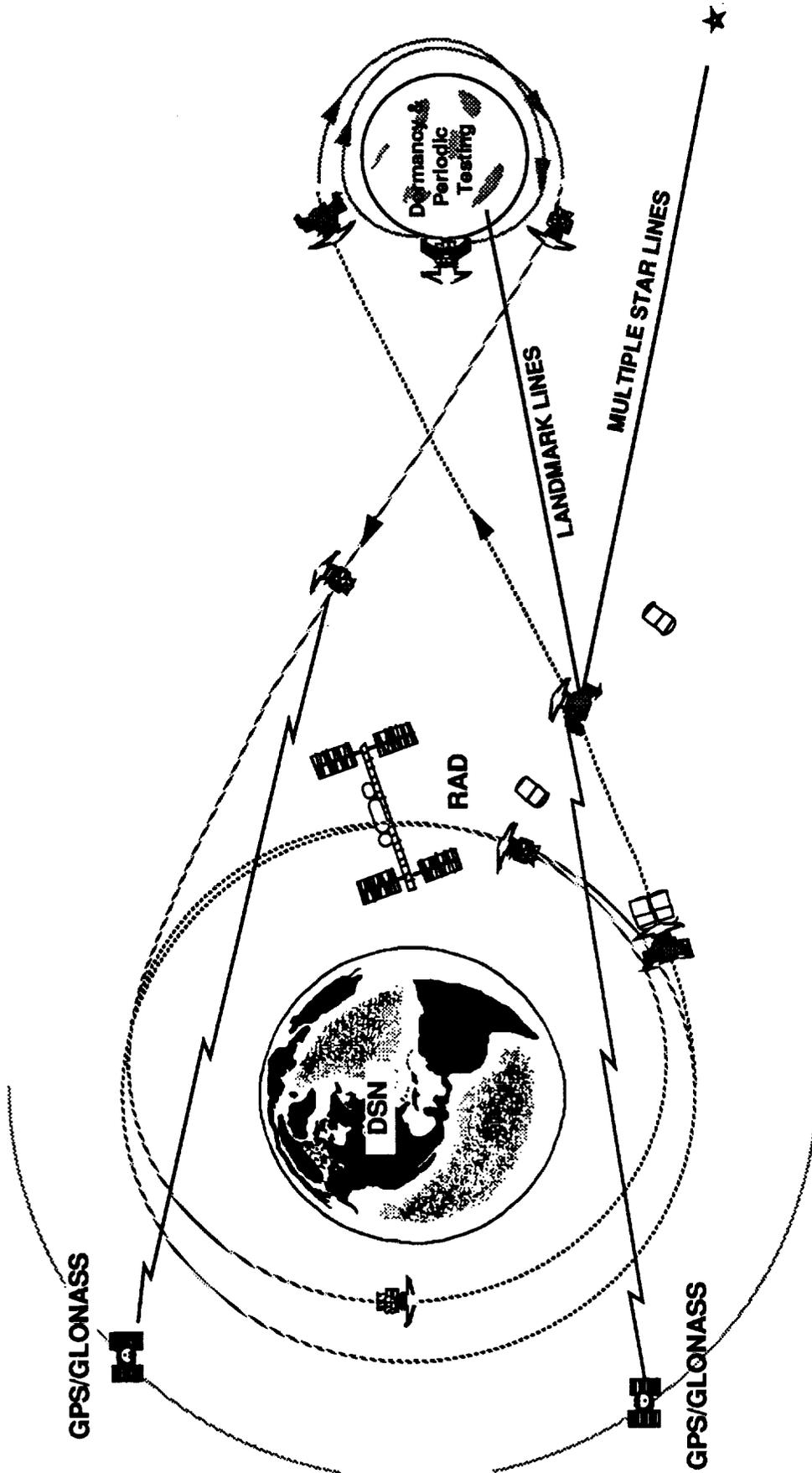
- Transfer Time - 3 Days Plus 48 Hours After landing
- Dormancy: On Lunar Surface and at SSF

Figure 1-2.4.8-11.   Space-Based Guidance and Navigation

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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 ΔV and a set of six accelerometers to measure magnitude of ΔV 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 rendezvous and docking operations 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 nmi. For a cooperative target (transponder), maximum tracking distance is about 200 nmi. Antennas will be stowed prior to deorbit. A laser tracker could provide autonomous docking capability with a reflector target located on the target vehicle.

On the space-based vehicle, provisions for guidance and navigation (including rendezvous and docking) and lunar landing are contained in the core stage and controlled from the crew module (piloted missions). On the ground-based vehicle, avionics equipment are contained in an avionics pallet that returns with the crew module and is controlled from the crew module (piloted missions). All systems include built-in redundancy for piloted operations.

Communication and Tracking. Communication capability is provided between the vehicle and all Earth and orbital support elements. On the space-based vehicle, the core stage carries communication equipment with interfaces to the crew module for piloted operations, and on the ground-based vehicle, this equipment is located on the avionics pallet and returned to the ground with the crew module.

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 two-way link through the 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 the 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. 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 avionics 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 the state of health of the vehicle and passes this information to a "system manager" that is the mission management 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 1-2.4.8-5. 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-downtime" network.

Bus network types that are current networks or about to have space application include the shuttle 1-Mbps 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 using 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. Implementation costs are reduced, maintainability (high level of BIT and standard interfaces) increases, and resource utilization is maximized because the system uses only a few module types (less than 20). Some common types include the Space Station 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). The 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 and maintenance interface, and BIT coverage.

Controls and Displays. 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 because piloting skills may degrade after 6 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 LCDs 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.

1-2.4.9   Environmental Control and Life Support

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

Requirements. Top-level requirements for the environmental control, life support, and personnel provisions are given in Figure 1-2.4.9-1.

Hardware Description. Basic life support functions as applied to the STV can be grouped as shown in Figure 1-2.4.9-2. 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 1-2.4.9-3. Shown are interfaces with other vehicle systems (i.e., fuel cells) and 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 3.0 subsystem trades. Since an adequate supply of water is provided as byproduct 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.

Similarity of the STV life support system to that of the STS orbiter provides a credible (i.e., verifiable) description of the baseline hardware. Figure 1-2.4.9-4 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. For instance, there are three

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	Basing		Requirement Source		
	Space	Ground	Given	Derived	Provis.
Environmental Control					
Clean/decontaminate contaminated atmosphere	X	X	X		
Monitor, certify, revitalize atmosphere and water	X	X	X		
14.7 psi atmosphere, 21% O2 with flammability and EVA constraints	X	X	X		
Reject TBD kW metabolic heat, TBD kW equipment heat	X	X		X	
Personnel Provisions					
Medical	X	X	X		
Protect, maintain, monitor health of crew					
Capability of ill or injured crew transport to medical facilities					
NASA STD-3000 is applicable	X	X	X		
Crew habitat shall provide facilities for meals and recreation	X	X	X		
Provide for 4 crewmembers	X	X	X		
Provide zero-g commode and waste managment	X	X			X

Figure 1-2.4.9-1. ECLSS Design Requirements

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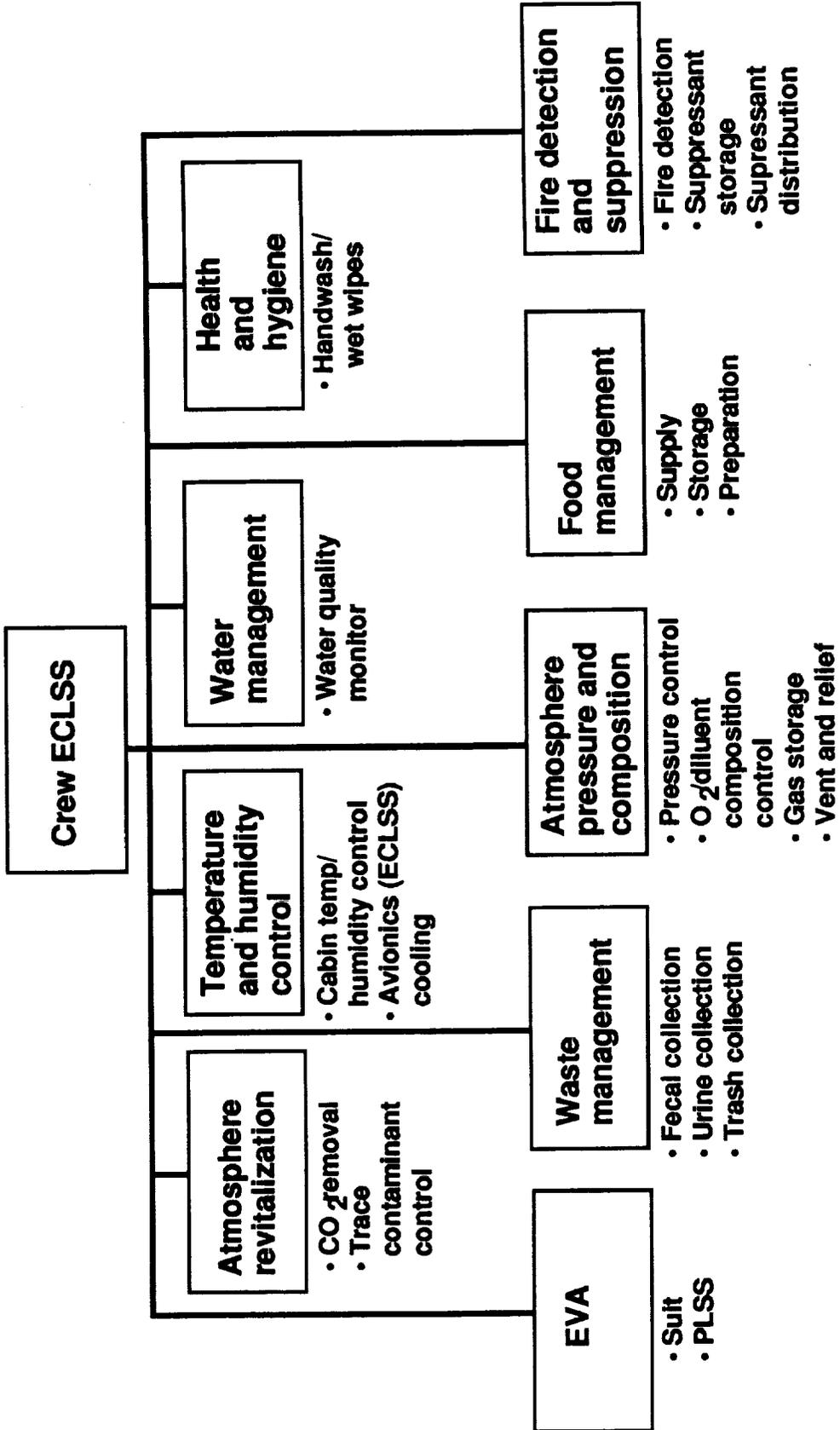
Figure 1-2.4.9-2. Basic ECLSS Functions
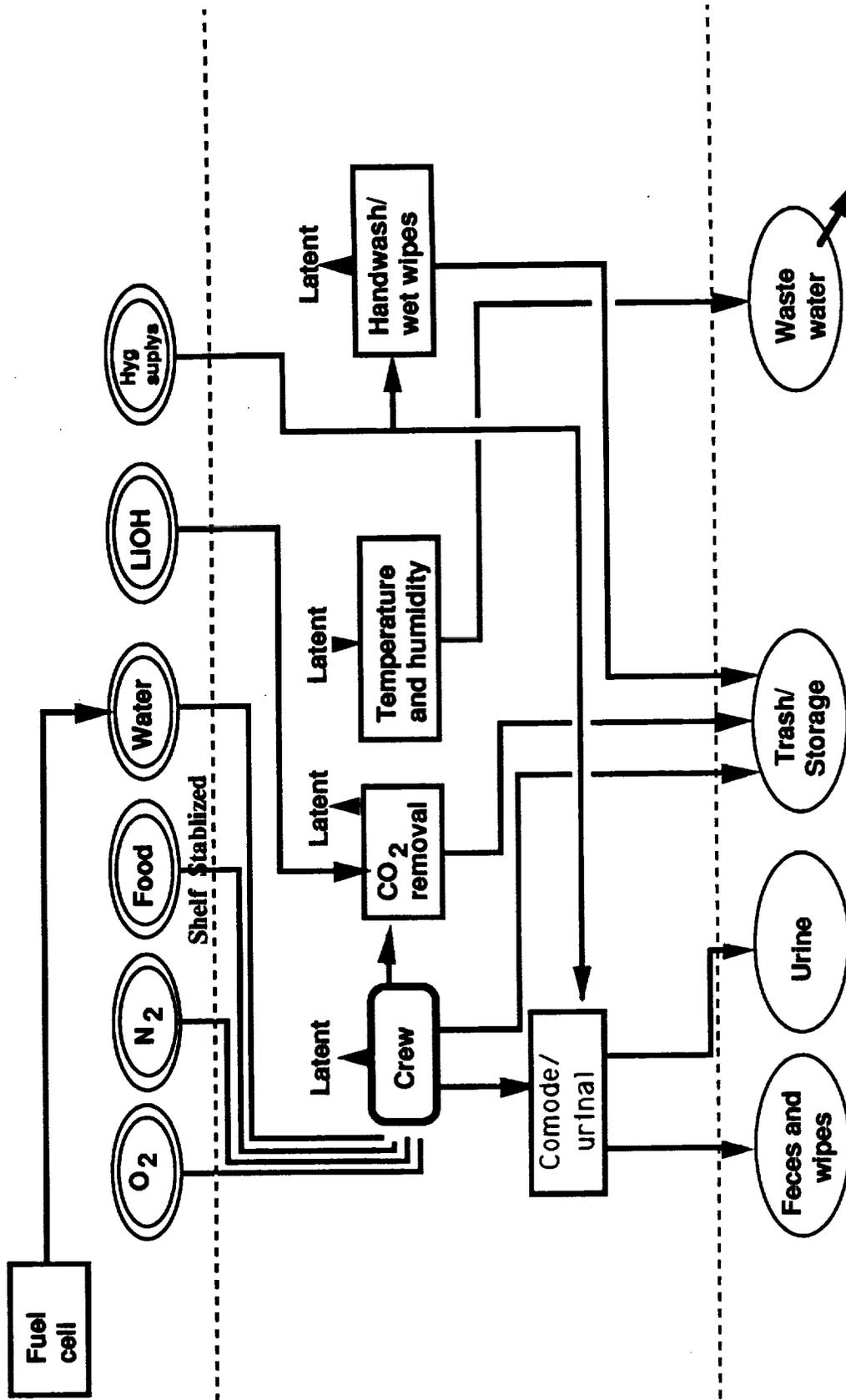
Figure 1-2.4.9-3.   ECLSS Configuration Diagram

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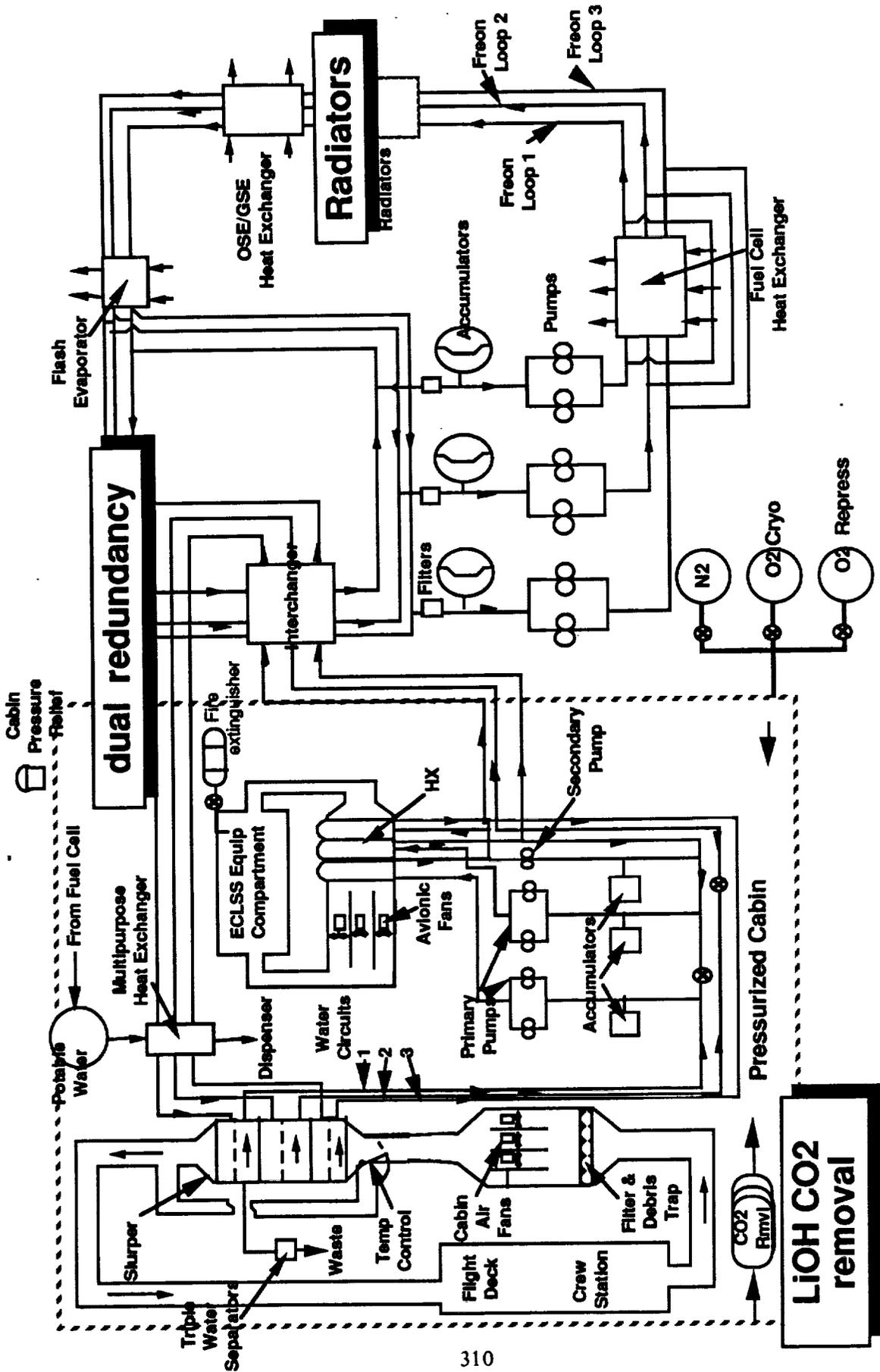
Figure 1-2.4.9-4. ECLSS Schematic

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.

Weight, volume, and power for the space-based STV ECLSS and personnel provisions are given in Figures 1-2.4.9-5 and 1-2.4.9-6. The data contained are consistent with the ECLSS functional diagram, the ECLSS approach diagram, and the ECLSS hardware schematic. Items considered critical are triply redundant and are indicated by a "(3)" after their name. Dual redundant items are indicated by a "(2)".

There is no ECLSS hardware difference between the space-based and ground-based STV configurations, but there are differences in storage requirements for consumables such as atmospheric gases, food, water, and expendables such as wipes and LiOH cartridges. These differences are shown in Figure 1-2.4.9-7. The power figures listed are approximate on the conservative side. Estimates were based on STS crew sizes (7 to 10 persons). Since the STV crew capacity is smaller than that for STS, the actual power consumption will probably be less.

Atmosphere Pressurization and Revitalization. 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

Subsystem	Qty	Mass (kg)	Volume (m3)	Avg Power (Kw)*	Peak Power (Kw)*	Connected Power Reqmnts
Cabin and Personnel System	1	491	2.26	0.662	0.696	
O2 Storage - Cryogenic			Included in fuel cell supplies			
O2 Storage - Gas (3 - repress)	3	76	0.41			
N2 Storage - Gas (3 - make-up & repress)	3	76	0.41			
Press Plumbing		16				
Cabin Pressure & Composition Control	3	88	0.23	0.021	0.055	28 vdc
CO2 Removal - 2-Bed LiOH		5	0.20			
LiOH Cannister Stowage (110% of req'd)		129	0.45			
Temperature & Humidity Control	3	86	0.55	0.641	0.641	115 vac/3p/400Hz
Trace Contaminant Control	3	5	0.01			
Ducting, Misc		9				
Equipment Cooling	11	82	0.01			
Coldplates		39	0.01			
Plumbing		36				
Coolant Tankage		6				
Heat Transfer Loop		159	0.28	0.611	0.611	115 vac/3p/400Hz
Primary + Secondary Water Pump Pkg	3	54	0.11	0.423	0.423	5 & 28 vdc
Heat Exchangers	6	52	0.09	MIN	MIN	
Avionics Cooling Assy	3	19	0.08	0.188	0.188	115 vac/1&3p/400Hz
Plumbing, etc		34				
Heat Rejection		187	0.52	0.338	0.338	
Flash Evaporator Assy	2	53	0.26	0.073	0.073	32 vdc
Topping Duct Assy	2	51				
High Load Duct Assy	2	22				
Freon Pump Package	3	61	0.26	0.264	0.264	115 vac/3p/400Hz
Radiators (3 - circuits)		200	2.21			
ECLSS Supt/Inst		112	0.53			
TOTAL		1229	5.80	1.611	1.645	

* Very crude estimate at this time - probably a little high)
(4-Person LSS; 4-Person load; Space Shuttle Life Support System; 14.4 - Day (abort) Mission Duration; 765 ft3 Cabin)

Figure 1-2.4.9-5. Space-Based Crew Module ECLSS Equipment

BOEING

Subsystem	Qty	Mass (kg)	Volume (m3)	Avg Power (Kw)*	Peak Power (Kw)*	Connected Power Reqmnts
Food Management		142	1.61	0.125	0.995	115 vac/3p400Hz, 28 vdc
Food Management - Galley		36	0.16	0.125	0.995	
Food Stowage		102	1.44			
Dry Wipes		4	0.01			
Water Management	2	51	0.38	0.000	0.004	28 vdc
Potable Water Storage Tank		45	0.38	MIN	0.004	
Handwash/Wet wipes		2				
Plumbing, valves, etc		5	0.00			
Waste Management		105	0.90	0.018	0.504	28 vdc
Waste Water Storage Tank		22	0.19	MIN	0.004	120 vdc
Commode/Urinal		75	0.68	0.018	0.500	
Emergency Waste Collection (bags)		7	0.03			
Fire Detection and Suppression		4	0.00	0.000	0.004	(5 vdc - est)
Smoke Detectors		3		MIN	MIN	(28 vdc - est)
Fire Suppression Tank		1		MIN	0.004	
Furnishing and Equipment		354	1.85			
Seats, Personnel Restraints	4	184				
Medical, Health Management		50				
Incidental Equipment	4	120				
Cabin Lighting				0.150	0.150	28 vdc
EVA - 4 crewpersons		441				
EMU (4 - STS SSAs & PLSSs)						
Other Supt/Inst		30	0.29			
TOTAL		686	3.18	0.293	1.657	

Figure 1-2.4.9-6. Space-Based Crew Module Personnel Provisions

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| Configuration/ Mission Duration | Consumables Weight (lbm) | | | | | Expendables Weight (lbm) | | | Total |
	Oxygen Gas [1]	Nitrogen Gas [2]	Food Wt	Food Vol	Potable Water [3]	LiOH Storage	Dry Wipes	Wet Wipes	
SB-1 / 14.4 days	159.6	183.1	224.6	50.7	305.9	299.2	8.2	4.0	1184.7
GB-1 / 13.8 days	142.1	138.5	215.3	48.6	293.1	290.2	7.9	3.9	1090.9
GO-1 / 13.8 days	155.1	182.9	215.3	48.6	293.1	290.2	7.9	3.9	1148.3

[1] Metabolic + leakage + repressurization. Metabolic and leakage may be supplied from crygenic storage.
[2] Leakage + repressurization
[3] Supplied from fuel cells but should allow for at least 2-day storage as backup.

Figure 1-2.4.9-7. ECLSS Consumables

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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.

Temperature and Humidity Control. 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 and 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 used 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 space-based vehicle is similar, but returns to the Space Station and does not require ammonia boilers.

The radiators used for these vehicles are deployable triple-loop metallic radiators covered with a high-reflectivity, high-emittance coating. The radiators on the ground-based vehicle are jettisoned with the core stage prior to atmosphere reentry, but those on the space-based vehicle are retracted behind the aerobrake during the aeromaneuver and are reused, as shown in Figure 1-2.4.9-8.

BOEING

SUN FACING	SURFACE	WASTE HEAT
	AREA (FT2)**	REJECTED (KW)
	115.7	3
	154.3	4
	192.9	5
	231.5	6
	270.1	7

SOLAR HEATING = 442 BTU/FT2-HR

TELESCOPING GUIDERAILS

DEPLOYABLE RADIATOR
COVERED WITH 10 MIL
SILVERED TEFLON*
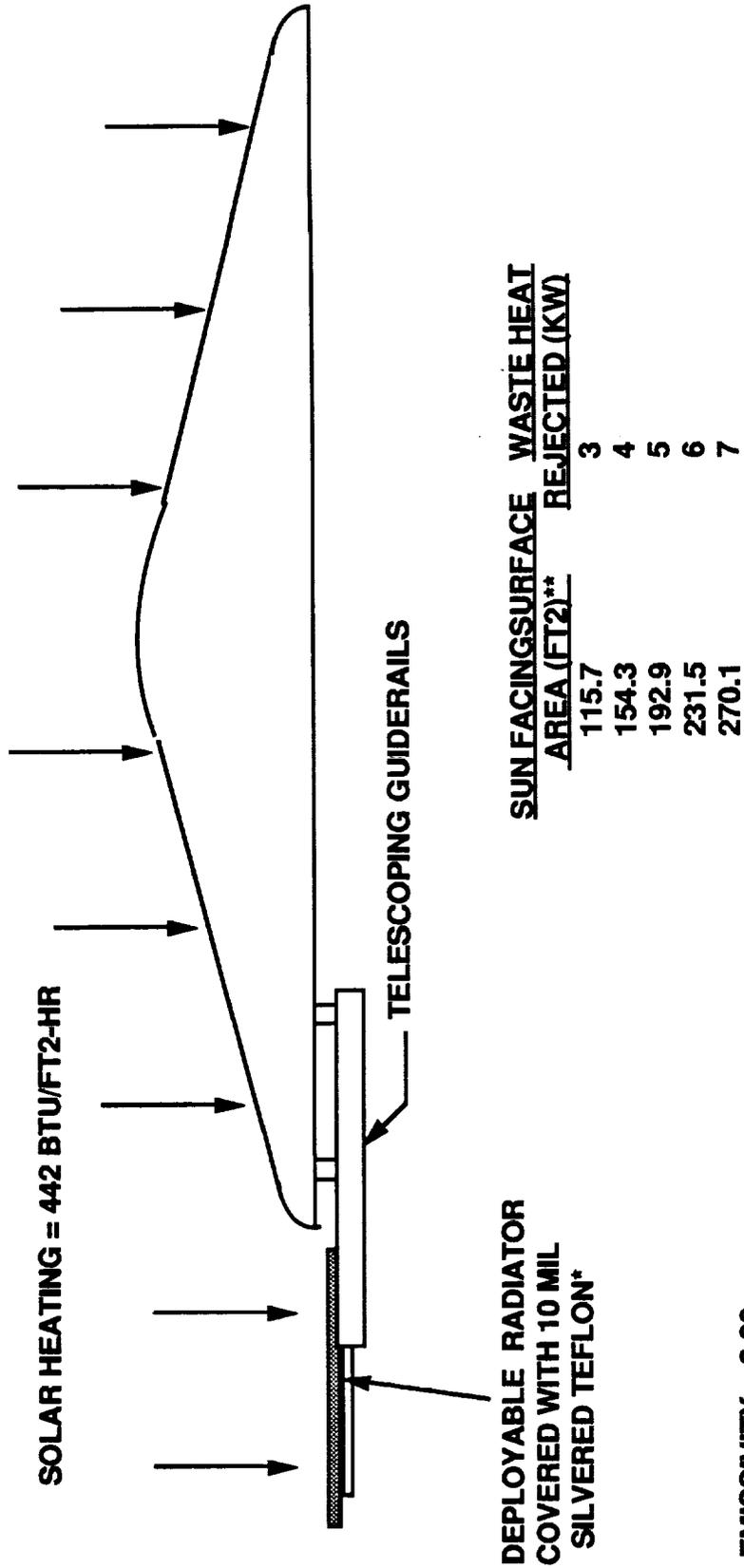
* EMISSIVITY = 0.09
ABSORPTIVITY = 0.88

** Assumes an average surface temperature of 80 deg F
and that the back side is adiabatic.

Figure 1-2.4.9-8. Space-Based Thermal-Rejection System

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Fire Detection and Suppression. 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 byproducts in the case of fire.

Food and Water Management. 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 6 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 food is prepared using warm water from an onboard galley and is cooked in a convection oven, also contained within the galley. A shuttle-type galley is included in the equipment list due to the length of the STV mission. Food and utensil storage volume will be provided for a crew of 4 for 12 days, assuming 4 lb/person/day.

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 byproducts, water dispenser, and tanks with a contingency water supply.

Waste Management. 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. 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.

Furnishings and Equipment. 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 en route to or from the lunar surface.

1-3.0   CONFIGURATION AND SUBSYSTEM TRADE STUDIES

During the STV study, the vehicles underwent a series of trade studies and analyses. This work can be divided into overall system (i.e., configuration) trades and specific subsystem (i.e., power, propulsion, and so forth) trades. As part of this analysis, it is important to remember the history of the STV contract. The first portion of the study was focused on the Skunkworks, which is more fully discussed in Appendix A. This portion of the study addressed a performance-driven configuration composed of a lunar transfer vehicle (LTV) for transit between the Earth and the Moon and a lunar excursion vehicle (LEV) for traveling between the surface of the Moon and low lunar orbit (LLO). Following the completion of this first phase, the philosophy of the LTV/LEV system was more fully analyzed and options of this architecture were more fully explored.

However, the configuration is driven by criteria other than pure performance. As previously discussed (section 2-1.1.3) cost, margins and risk, other mission capture, and benefits to the Mars mission were felt to be important selection criteria. With these criteria factored in, the study entered a second phase. During this second phase, the work on the configuration was focused on supporting the evaluation of the new architecture options.

The outcome of this architecture analysis was a selection of three distinct configurations and then a refinement to two distinct configurations (and three operational concepts). It is during this downselect process that further work was performed on defining and refining the subsystems required for the vehicles.

1-3.1   CONFIGURATION TRADES

Following the Skunkworks, the study continued to explore the options of the LTV/LEV system. The configurations were assessed based on the groundrules that the STV would use cryogenic propellants and be capable being launched (in pieces) from the Kennedy Space Center. Along with these groundrules went the assumptions that the vehicle would be space based (at the Space Station), reusable, and composed of modular core elements that can be evolved to perform other missions. Using these groundrules and assumptions three distinct families were developed for further analysis (Figures 1-3.1-1 and 1-3.1-2).

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These three families were explored further to trade various configuration options to determine which of these vehicles had the best performance. These configurations and the operational concept for those vehicles are shown in Figures 1-3.1-3 through 1-3.1-22. These figures show the significant features of each of these vehicles, the pros and cons of the configuration, and recommendations on the direction the work on those vehicles would take.

The performance of these configurations was then analyzed to determine the differences between the configurations and the families. Looking at the performance of the three families, in this case the amount of propellant required in LEO (Figure 1-3.1-23), there is a noticeable difference between the performance of the families. A closer look at the propellant requirements for the individual concepts is shown in Figure 1-3.1-24. Along with the options that are configuration dependent, other options exist that can be used through the spectrum of the families. These generic options and their impacts on vehicle performance are addressed in Figure 1-3.1-25. Based on the development and analysis of these three families and the subsequent 20 configurations in this phase of the study several conclusions can be drawn. These conclusions and the resulting recommendation are shown in Figure 1-3.1-26.

Contingency operations directly impact the configuration in areas such as how the cargo is carried and handled. The vehicle cannot return to Earth with the cargo still attached. In the event of a mission abort, the cargo must be jettisoned for crew safety. This means that the configurations must be laid out in a manner that does not preclude this jettisoning of cargo. Using the criteria of performance and contingency aborts, several configurations were considered good candidates for elimination in a downselect (Figures 1-3.1-27 and 1-3.1-28).

During this phase of the study, the concept of the STV was a modular, evolutionary system where the vehicle is built up from a set of common components. As such the various configurations were taken apart and then recombined in different ways to capture the other missions being considered as part of the STV. To determine the validity of these various recombinations, one configuration from each family was chosen as representative of that entire family and used to determine what components would have to be used to allow

FAMILY A

Two Vehicles (LTV & LEV)
Two Propulsion & Avionics Modules

FAMILY B

One Vehicle (LTV only)
One Propulsion & Avionics Module

FAMILY C

One Vehicle (LTV only)
Two Propulsion & Avionics Modules
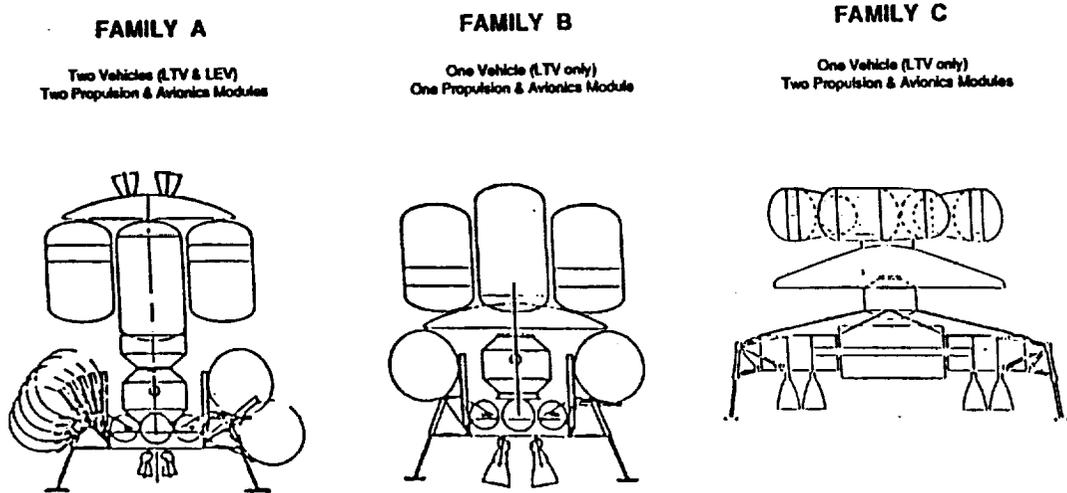
STEADY STATE PILOTED MISSION (DRM L2 & L3)

Figure 1-3.1-1. Vehicle Concept Family Definition

FAMILY A

Two Vehicles (LTV & LEV)
Two Propulsion & Avionics Modules

FAMILY B

One Vehicle (LTV only)
One Propulsion & Avionics Module

FAMILY C

One Vehicle (LTV only)
Two Propulsion & Avionics Modules

**SAME AS
FAMILY B**

FAMILY A COMPONENTS
ARE RECONFIGURED TO
A ONE VEHICLE, ONE
P/A MODULE DESIGN
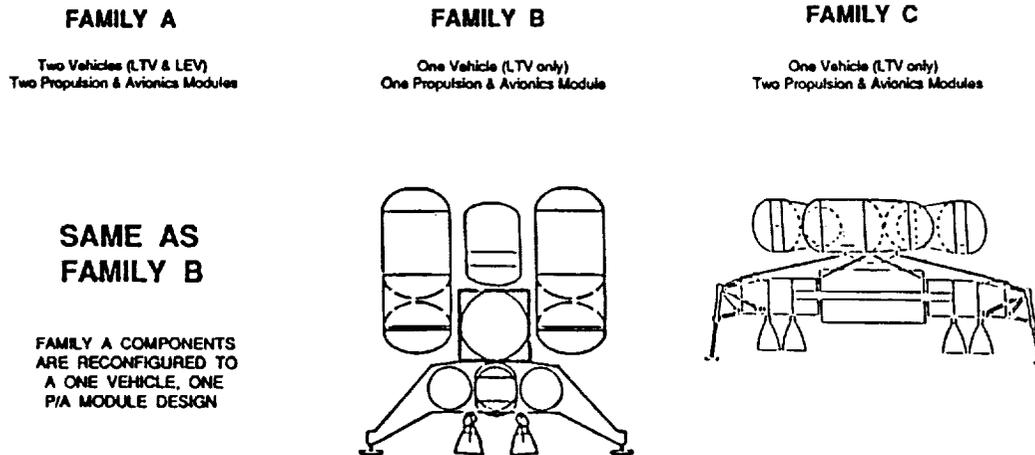
FULLY EXPENDABLE CARGO MISSION (DRM L1 & L4)

Figure 1-3.1-2. Vehicle Concept Family Definition

Features:

- Two crew modules
- Two P/A modules
- Dock with LEV on LLO
- LTV remains on LLO
- Two cargo containers

PRO	CON
• Max P/L to Lunar Surface	• 4 Large Engine Penetrations Thru Aerobrake
• Cargo Accessible for Unloading/Jettison	• Not Readily Adaptable for Single Payload
• Good Landing Visibility	• Requires Tall Support Tower for LEV Crew Module

RECOMMENDATION:- Retain further for consideration

Figure 1-3.1-3.   Mission Scenario - Configuration #1
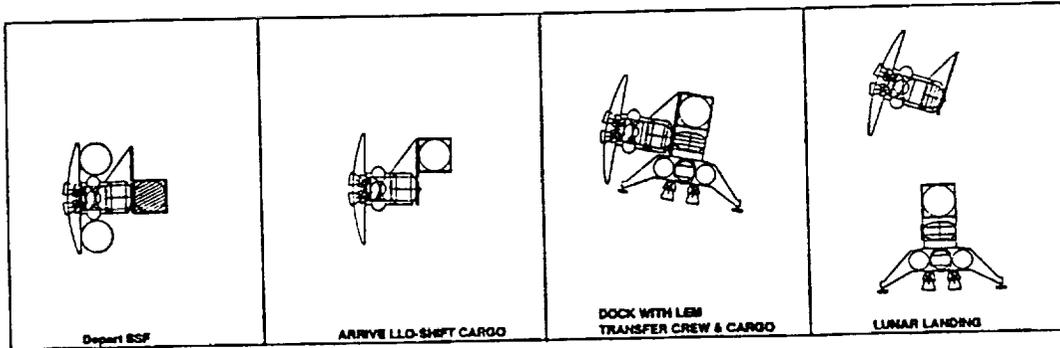
Features:

- Two crew modules
- Two P/A modules
- Dock with LEV on LLO
- LTV remains on LLO
- Single cargo container
- Cargo moves to the side for transfer to LEV

PRO	CON
• Max P/L to Lunar Surface	• Not suitable for dual cargo missions
• Good landing visibility	• Not suitable for LEV delivery misson
• Good cargo access for payload unloader	• 4 large engine penetrations thru aerobrake

RECOMMENDATION:- Retain for further consideration

Figure 1-3.1-4.   Mission Scenario - Configuration #2
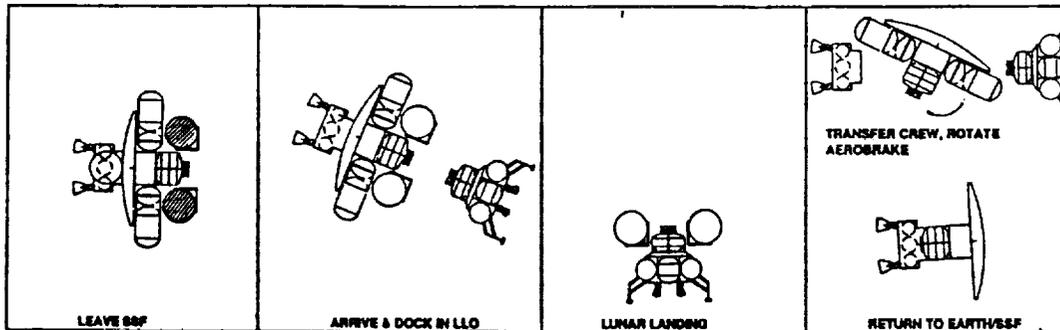
Features:

- Two crew modules
- Two P/A modules
- Dock with LEV on LLO
- LTV remains on LLO
- Two cargo containers
- Large tanks req'd to lunar orbit

PRO	CON
• No aerobrake penetration	• Large tanks reduce lunar P/L by approx 500 kg.
• Good landing visibility	• Additional LLO maneuver to prepare for earth return
• Good access to P/L	

RECOMMENDATION:- Abandon further work with large tank concepts

Figure 1-3.1-5. Mission Scenario - Configuration #3
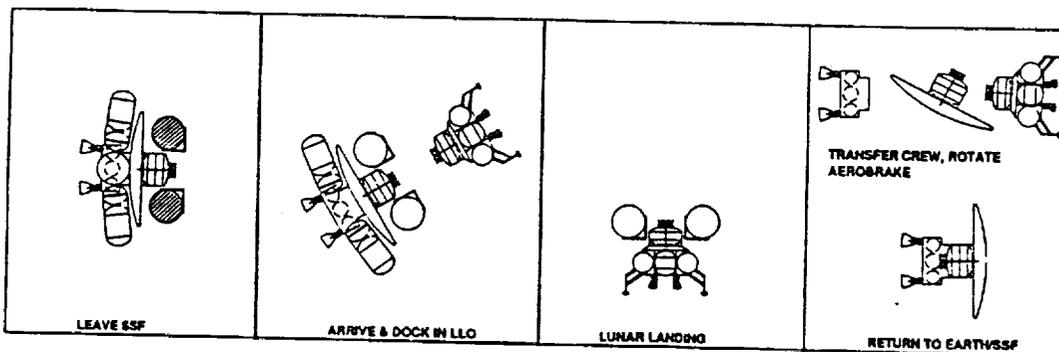
Features:

- Two crew modules
- Two P/A modules
- Dock with LEV on LLO
- LTV remains on LLO
- Two cargo containers
- Fill LEV tanks with line thru re-entry shield

PRO	CON
• Max P/L to lunar surface	• Additional LLO maneuver required to prepare for earth return
• Small propellant line penetration thru aerobrake	
• Accessible cargo	
• Good landing visibility	

RECOMMENDATION:- Retain for further consideration

Figure 1-3.1-6. Mission Scenario - Configuration #4
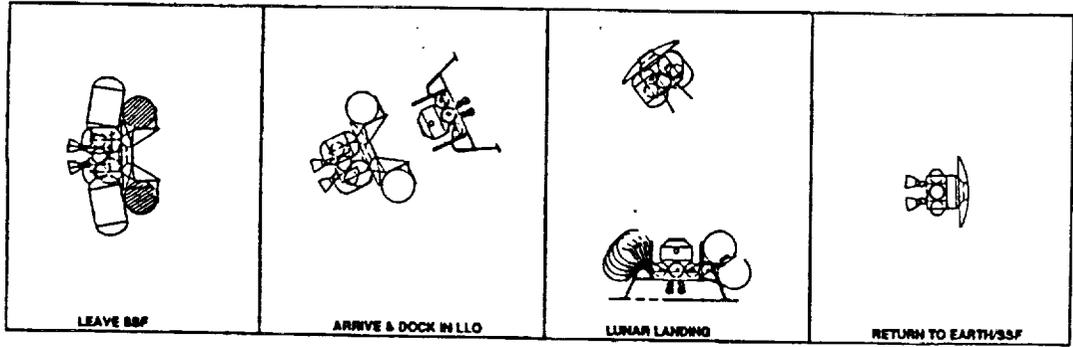
323

PRO	CON
• Max payload to lunar surf.	• Crew access hole must be sealed for re-entry
• Good landing visibility	
• Good payload access	• Crew access penetration Thru aerobrake

Features:

- Two crew modules
- Two P/A modules
- Dock with LEV on LLO
- LTV remains on LLO
- Two cargo containers
- Transfer crew thru 1 re-entry shield
- View thru hatch cover for lunar docking
- Propellant lines thru re-entry shield to fill LEV tanks

RECOMMENDATION:- Retain for further consideration

Figure 1-3.1-7. Mission Scenario - Configuration #5

PRO	CON
• No aerobrake penetrations	• 2 lunar orbit maneuvers req'd
• Good landing visibility	
• Good cargo jettison capability	• Special adapter req'd above aerobrake to deliver LEV to Lunar Surface
• Maximum payload to lunar surface	
• Cargo accessible for payload unloader	
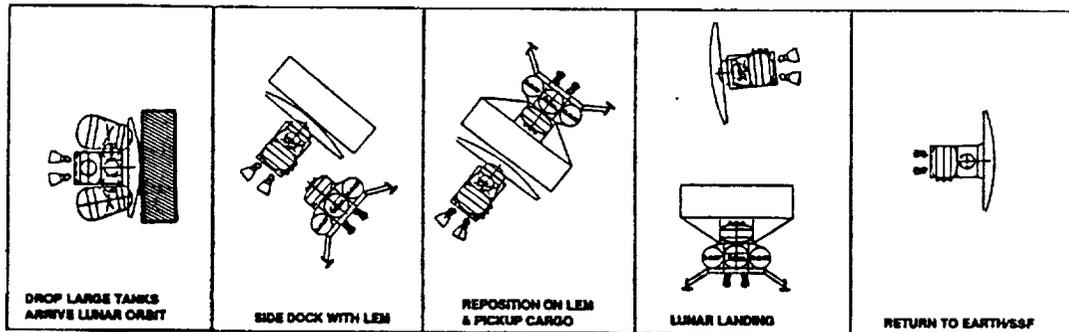
Features:

- Two crew modules

- Two P/A modules

- Dock with LEV on LLO

- LTV remains on LLO

- One or two cargo containers

- Cargo carried above aerobrake

RECOMMENDATION:- Retain for further consideration

Figure 1-3.1-8. Mission Scenario - Configuration #6
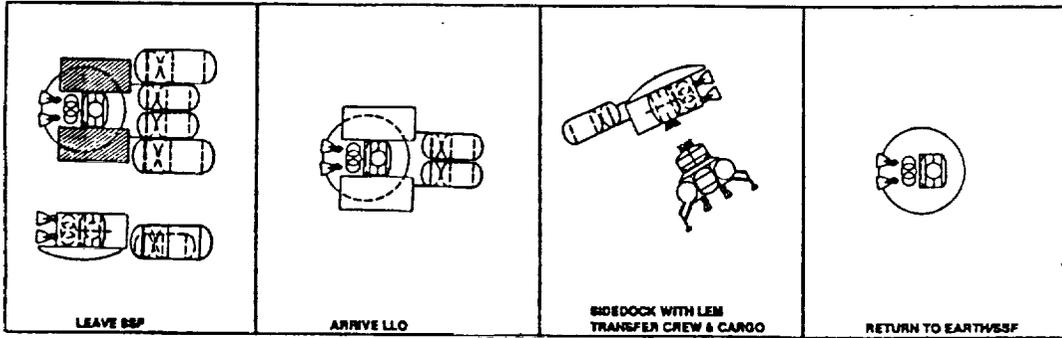
Features:

- Two crew modules
- Two P/A modules
- Dock with LEV on LLO
- LTV remains on LLO
- Two cargo containers
- Aerobrake fixed on side

PRO	CON
• Max payload to lunar surf.	• Not suitable for single cargo & LEV delivery missions
• No penetrations thru aerobrake	
• Stable aerobrake entry due to low C.G.	• Support of 4 drop tanks is difficult
• Good cargo jettison	

RECOMMENDATION:- Retain for further study - Also consider concept with tanks at side & cargo in front

Figure 1-3.1-9. *Mission Scenario - Configuration #7*
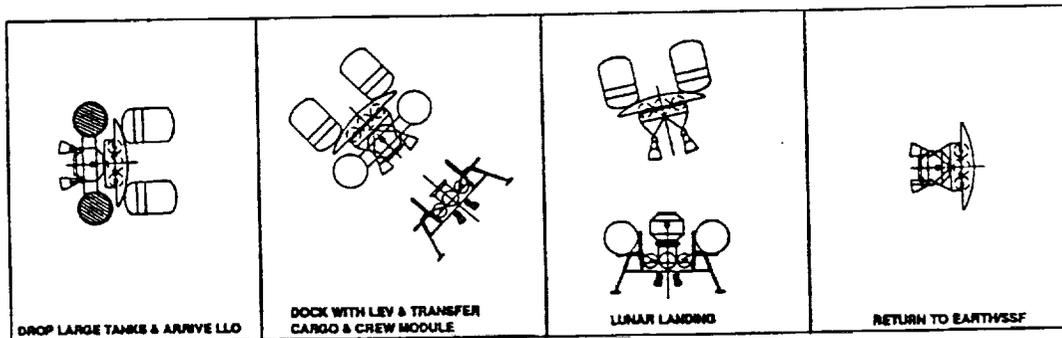
Features:

- One crew module
- Two P/A modules
- Dock with LEV on LLO
- LTV remains on LLO
- Two cargo containers
- Transfer crew module from LTV

PRO	CON
• One crew cab should reduce cost	• Propellant lines thru aerobrake
	• Taking the large crew module to the lunar surface will reduce lunar P/L by 2 Kg
	• Propellant line penetration thru aerobrake
	• Engines cannot thrust thru CG.

RECOMMENDATION:- Abandon further consideration of this concept

Figure 1-3.1-10. *Mission Scenario - Configuration #8*
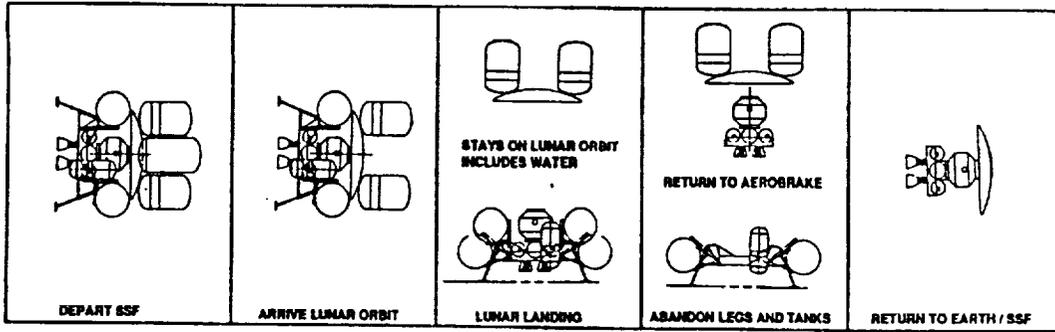
325

		STAYS ON LUNAR ORBIT INCLUDES WATER	RETURN TO AEROBRAKE	
DEPART SSF	ARRIVE LUNAR ORBIT	LUNAR LANDING	ABANDON LEGS AND TANKS	RETURN TO EARTH / SSF

Features

- One crew module
- One P/A module
- Small tanks, aerobrake & shield water stay on LLO
- Two cargo containers
- Abandon landing legs & lunar descent tanks on lunar surface

PRO	CON
• Most P/L to lunar surface by a single P/A module vehicle • Good access to cargo on lunar surface • Good cargo jettison capability	• Landing legs & descent tanks must be replaced on every mission • Propellant line penetration thru aerobrake • Not readily adaptable to single cargo mission

RECOMMENDATION :- Retain for further consideration

Figure 1-3.1-11. Mission Scenario - Configuration #9

	STAYS ON ORBIT INCLUDING SHIELD WATER	RETURN TO AEROBRAKE	
DROP LARGE TANKS & ARRIVE AT LLO	LUNAR LANDING	ABANDON TANKS	RETURN TO EARTH/SSF
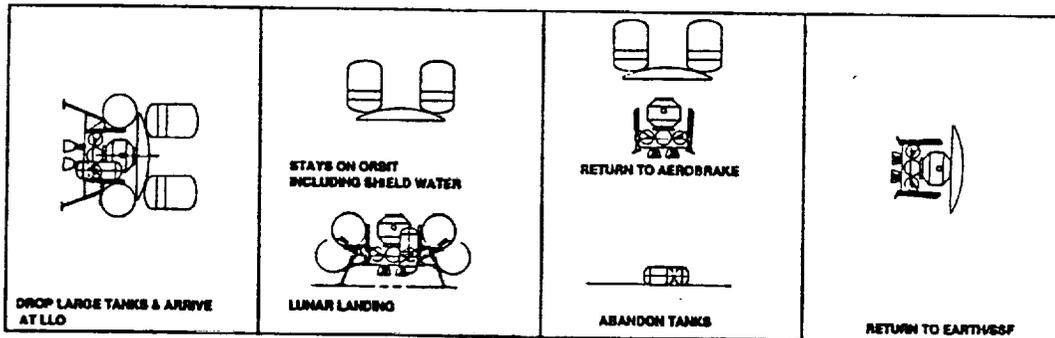
Features:

- One crew modules
- One P/A modules
- Small tanks, aerobrake & shield, water stay on LLO
- Two cargo containers
- Abondon Lunar descent tanks on lunar surface

PRO	CON
• Good access to cargo on lunar surface • Good cargo jettison	• 1530 KG P/L less than best 1 - P/A module concept • Descent tanks must be replaced on every mission • Propellant line penetration thru aerobrake • Not readily adaptable to single cargo mission

RECOMMENDATION:- Retain for further consideration

Figure 1-3.1-12. Mission Scenario - Configuration #10

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DROP LARGE TANKS & ARRIVE AT LLO | LUNAR LANDING | RETURN TO AEROBRAKE | RETURN TO EARTH/SSF

Features:

- One crew module

- One P/A module

- Small tanks, aerobrake & shield, water stay on LLO

- Two cargo containers

PRO	CON
• Good access to cargo on lunar surface	• 3225 KG P/L less than best 1-P/A module concept
• Good cargo jettison	• Propellant line penetration thru aerobrake
• May be cost advantage to save legs & tanks	• Not readily adaptable to single cargo mission

RECOMMENDATION:- Retain for further consideration

Figure 1-3.1-13. Mission Scenario - Configuration #11
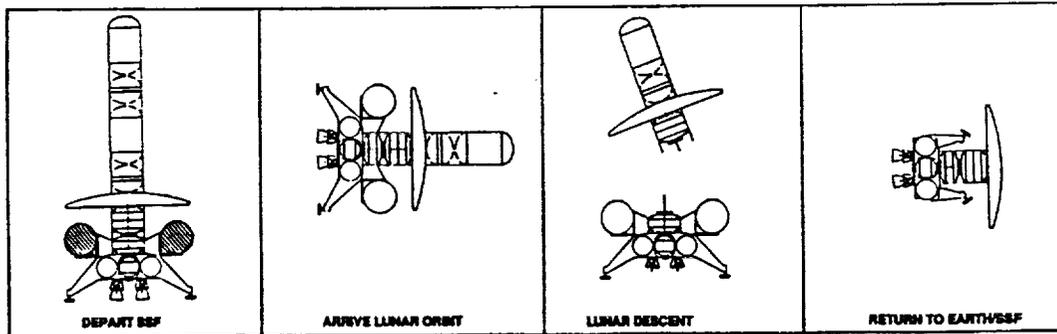
DEPART SSF | ARRIVE LUNAR ORBIT | LUNAR DESCENT | RETURN TO EARTH/SSF

Features:

- Two crew modules

- One P/A module

- Small tanks, aerobrake & shield water stay on LLO

- Two cargo containers

- Similar to 9,10,11 except tanks are in line

PRO	CON
• Good cargo access on lunar surface	• Long propellant lines req'd
	• Prop. lines thru aerobrake
• Good landing visibility	• More RCS control req'd than shorter section
	• Upper tank loads must be supported by tank walls
	• Not suitable for single cargo mission
	• 5000 KG less P/L to the lunar surf. than best single P/A module concept

RECOMMENDATION:- Retain for further consideration of in line tank concept

Figure 1-3.1-14. Mission Scenario - Configuration #12
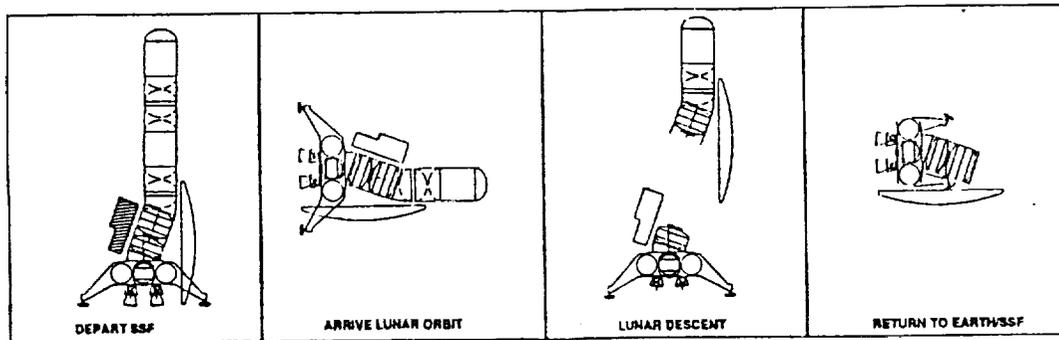
DEPART SSF	ARRIVE LUNAR ORBIT	LUNAR DESCENT	RETURN TO EARTH/SSF

Features:

* Two crew modules

* One P/A module

* Small tanks, aerobrake & shield water stay on LLO

* One cargo container

* Similar to 9, 10, 11 except tanks are in line & the crew modules are angled for C.G. Placement

PRO	CON
• Good cargo access on lunar surface - adaptable for easy self unload • No propellant lines thru aerobrake	• Long propellant lines req'd • More RCS control req'd than shorter version • Upper tank loads must be supported by lower tank walls • Not suitable for single cargo mission • 5000 KG less P/L to the lunar surf. than best 1 P/A module concept

RECOMMENDATION:- Retain for further consideration of in line tank concept

Figure 1-3.1-15.   Mission Scenario - Configuration #13
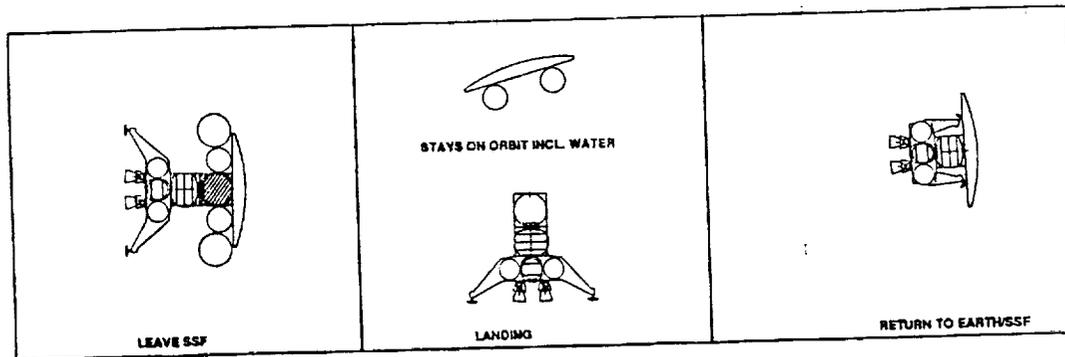
LEAVE SSF	LANDING	RETURN TO EARTH/SSF

STAYS ON ORBIT INCL. WATER

Features:

* One crew module

* One P/A module

* Small tanks, aerobrake & shield water stay on LLO

* One cargo container

* Cargo located between crew module & aerobrake

PRO	CON
• Good cargo visibility • Good cargo access for payload unloader • No aerobrake penetrations	• Unable to jettison cargo prior to crew module separation • 3225 KG less P/L delivered to lunar surface than best single P/A vehicle

RECOMMENDATION:- Abandon further consideration due to lack of abort capability

Figure 1-3.1-16.   Mission Scenario - Configuration #14

Figure showing three panels: LEAVE SSF | STAYS ON ORBIT INCL. WATER / LUNAR LANDING | RETURN TO EARTH/SSF
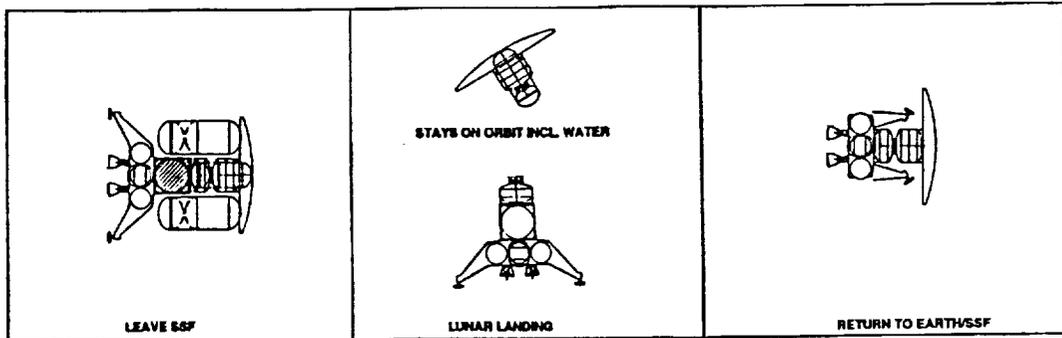
Features:

- Two crew module

- One P/A module

- Small tanks, aerobrake, 1 crew module & shield water stay on LLO

- One cargo container

- Cargo located between crew module & P/A module

PRO	-CON
• Good cargo visibility	• Unable to jettison cargo prior to lunar landing
• Good cargo access for payload unloader	• 5000 KG less P/L delivered to lunar surface than best single P/A vehicle
• No aerobrake penetrations	
• Readily adpatable for unmanned missions	

RECOMMENDATION:- Abandon further consideration due to lack of abort capability

Figure 1-3.1-17. Mission Scenario - Configuration #15

Figure showing three panels: DEPART SSF | LUNAR LANDING | RETURN TO EARTH/SSF
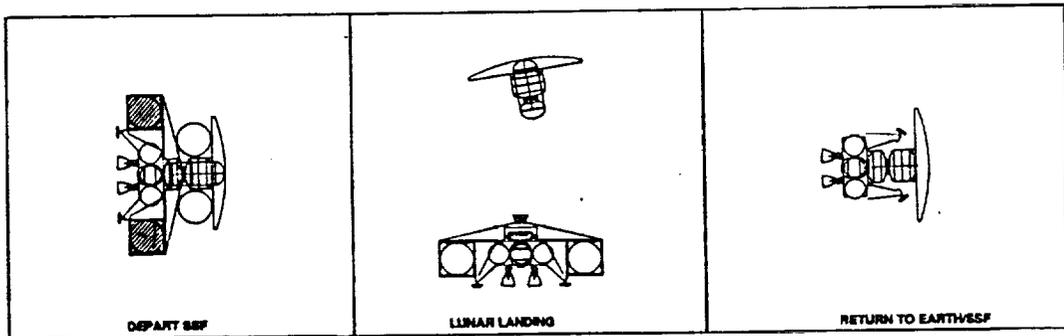
Features:

- Two crew modules

- One P/A module

- Aerobrake, small tanks, one crew module and shield water stay on LLO

- Two Cargo Containers

PRO	CON
• No aerobrake penetrations	• 5325 KG less P/L to the lunar surface than the best single P/A vehicle
• Good landing visibility	
• Good cargo self-unloading capability	
• Best cargo jettison capability	

RECOMMENDATION:- Retain for further consideration - could be similar to concept #9 with one crew module and abandon legs and tanks

Figure 1-3.1-18. Mission Scenario - Configuration #16
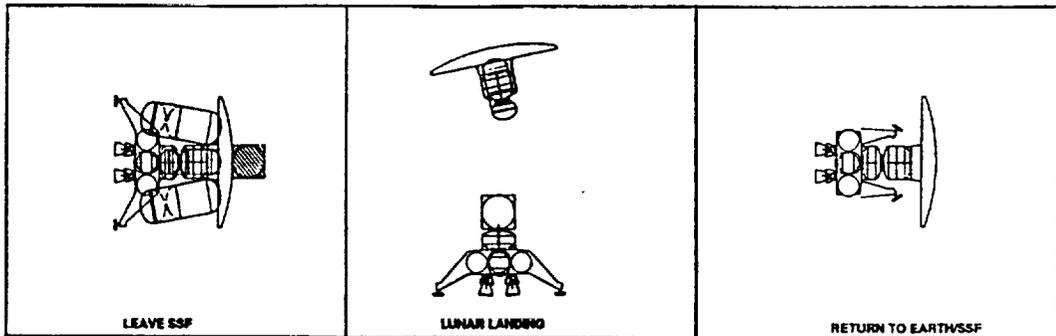
BOEING

Features:

- Two crew modules
- One P/A module
- Small tanks, one crew module, aerobrake & shield water to stay on LLO
- One or two Cargo Containers
- Cargo mounted on top of Aerobrake

PRO	CON
• No aerobrake penetrations • Readily adapted for 1 or 2 cargo containers • Good landing visibility	• LEM must maneuver to pick-up cargo • Cargo jettison on lunar descent is difficult • Crew entrapment danger if unable to unload cargo 5145 KG less cargo to lunar surface than best single P/A version.

RECOMMENDATION:- Retain for further consideration - could be similar to concept #9 with 1 crew module and abandon legs & tanks

Figure 1-3.1-19. Mission Scenario - Configuration #17
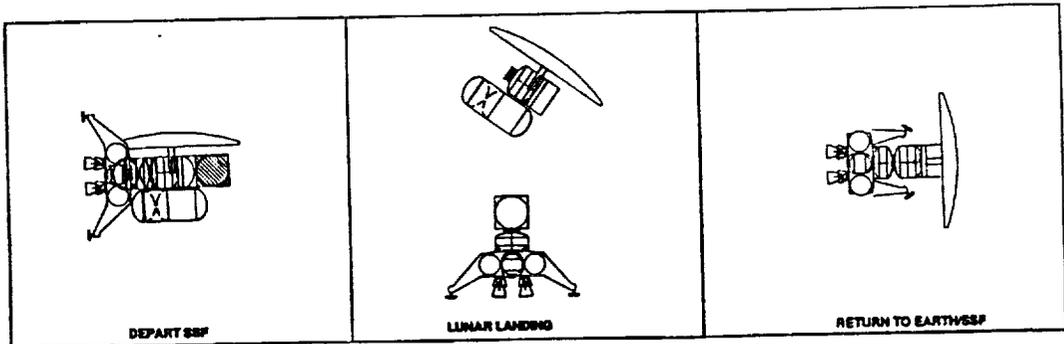
Features:

- Two crew modules
- One P/A module
- Small tanks, one crew module, aerobrake & shield water to stay on LLO
- One Cargo Container
- Pick up cargo with LEM maneuver on LLO Aerobrake
- Pivot aerobrake into place prior to re-entry

PRO	CON
• No aerobrake penetrations • Good cargo access for payload unloader	• 5640 KG less P/L to lunar surf. than best single P/A version • Difficult to pivot and lock large aerobrake • Large aerobrake could make P/L inaccessible to LEV • Crew entrapment danger if unable to unload cargo

RECOMMENDATION:- Abandon further consideration due to poor performance and structural complexity

Figure 1-3.1-20. Mission Scenario - Configuration #18

D180-32040-2

DEPART SSF   LUNAR LANDING   RETURN SSF
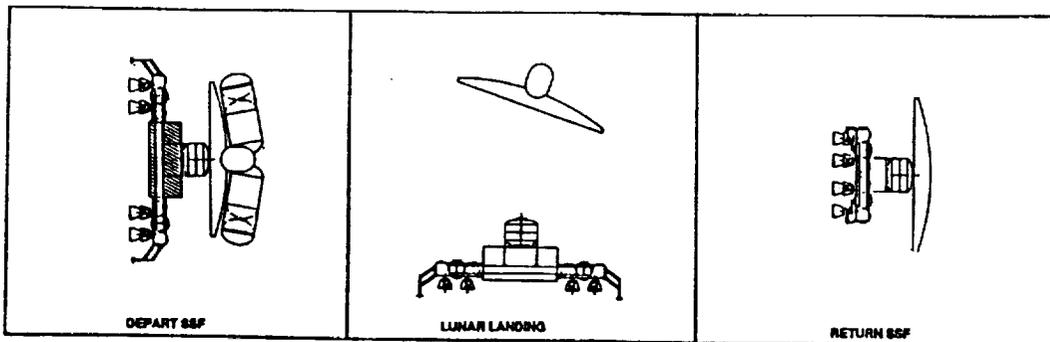
Features:

- One crew module
- Two P/A modules
- Small tanks, aerobrake and shield water stay on LLO
- 1 cargo container
- Cargo mounted between P/A modules
- P/A modules retract for return

PRO	CON
• Easy cargo unload at lunar surface	• Can only transport 6960 KG to lunar surface • Cannot survive 2 engine failuresunless more engines are added • Propellant line penetrations thru aerobrake • Large RCS thruster req'd

RECOMMENDATION:- Abandon further consideration due to poor performance

Figure 1-3.1-21. Mission Scenario - Configuration #19

DEPART SSF   LUNAR LANDING   RETURN SSF
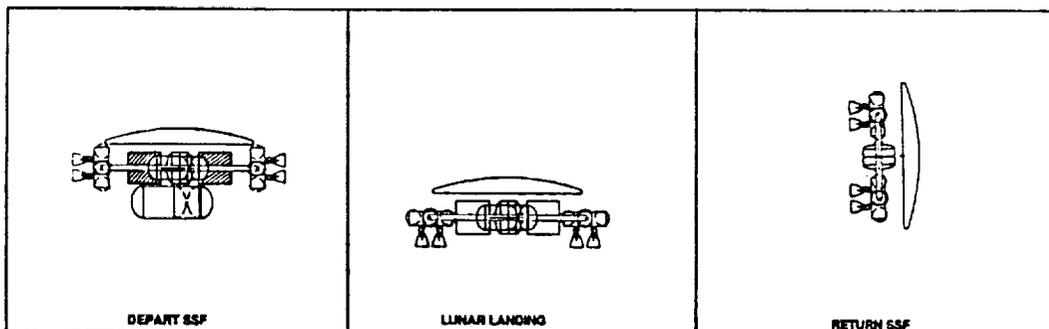
Features:

- One crew module
- Two P/A modules
- Small tanks, aerobrake and shield water stay on LLO
- 2 cargo containers
- Cargo mounted between P/A modules
- P/A modules retract for return
  P/A modules rotate to thrust thru CG (1 set dormant)

PRO	CON
• Easy cargo unload at lunar surface	• Can only transport 6960 KG to lunar surface • Cannot survive 2 engine failures unless more engines are added • Large RCS thruster req'd

RECOMMENDATION:- Abandon further consideration due to poor performance

Figure 1-3.1-22. Mission Scenario - Configuration #20

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STV

MSFC- **BOEING**

PERFORMANCE RESULTS CHART
SYSTEM CONCEPTS & EVOLUTION FAMILIES

PROPELLANT LAUNCH MASS TO LEO
13,000 KG Lunar Delivered Cargo - Steady State Piloted Mission
(DRM L2 & L3)

VEHICLE CONCEPT NUMBER
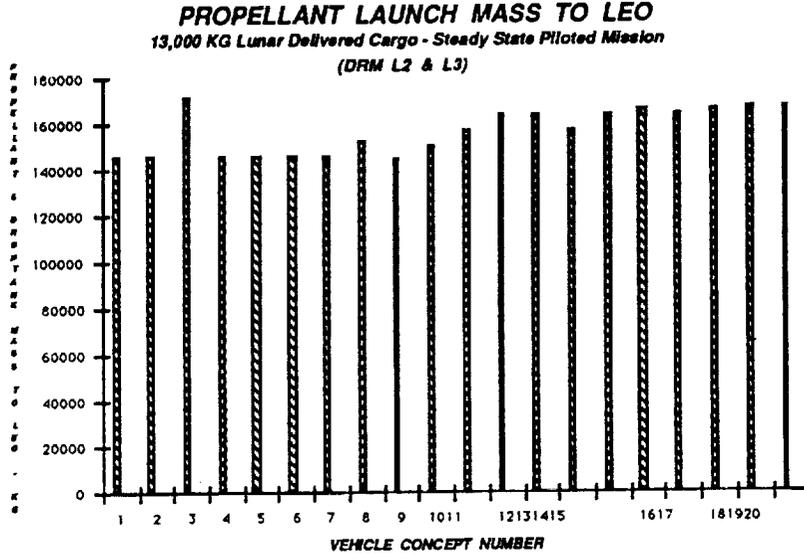
Figure 1-3.1-23.   Propellant Launch Mass to LEO

STV

MSFC- **BOEING**

PERFORMANCE RESULTS CHART
SYSTEM CONCEPTS & EVOLUTION FAMILIES

LUNAR SURFACE DELIVERABLE CARGO
Deliverable Cargo Limited to HLLV Launch Capability

VEHICLE CONCEPT NUMBER
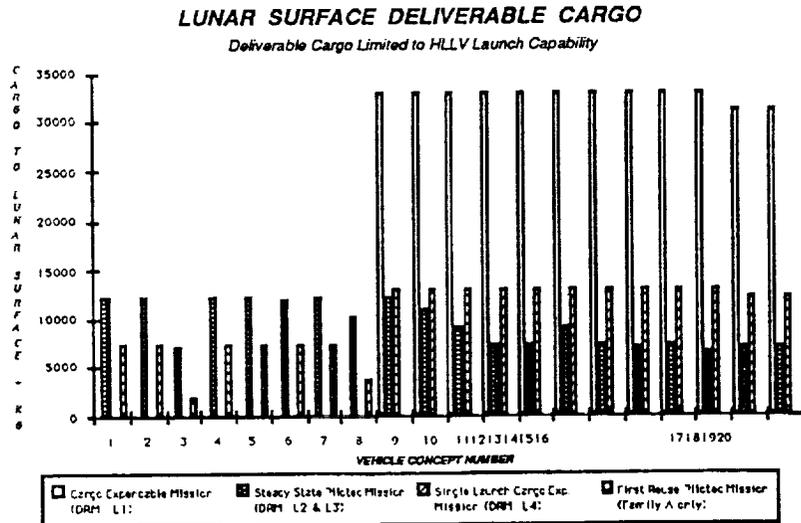
Figure 1-3.1-24.   Lunar Surface Deliverable Cargo

STV

PERFORMANCE BY FAMILY
System Concepts & Evolution Families
MSFC- *BOEING*

PERFORMANCE COMPARISON OF FAMILIES
Propellant & Droptanks to LEO vs. Lunar Surface Delivered Cargo
Steady State Piloted Mission

LUNAR SURFACE DELIVERED CARGO - KG

--- FAMILY A - Fully Reusable, Two Cabs (C1)　　◆FAMILY B - Fully Reusable, Single Cab (C11)　　✦FAMILY C - Fully Reusable, Single Cab (C20)
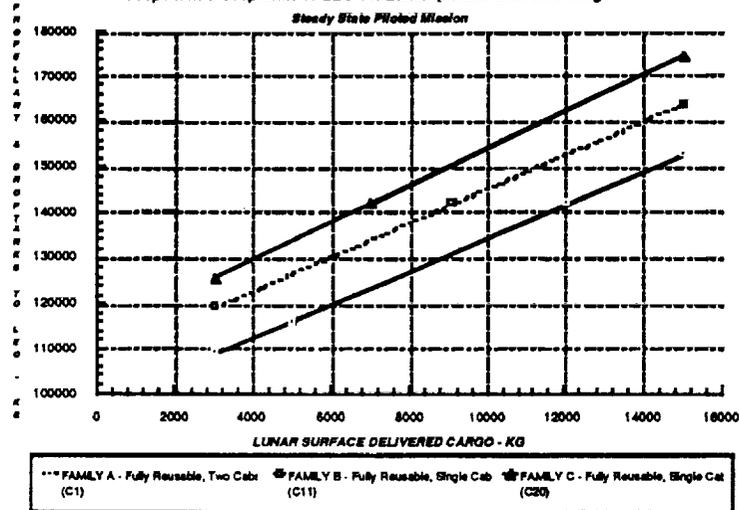
Figure 1-3.1-25. Performance Comparison of Families

STV

ADDITIONAL TRADE ISSUE EFFECTS
SYSTEM CONCEPTS & EVOLUTION FAMILIES
MSFC- *BOEING*

ADDITIONAL TRADE EFFECTS ON MISSION PERFORMANCE

TRADE ISSUE - Steady State Piloted Mission	PERFORMANCE DELTA (KG) * BY CONFIGURATION FAMILY		
	A	B	C
• LEO Staytime, +30 Days (Boiloff effects)	-405	-415	-350
• Include effects of Airborne Support Equipment (approx. 6.5 mt from 142 mt)	-1860	-1750	-1610
• Two Independent Crew Cabs rather than one cab	+1845	-1775	-1505
• Hybrid Crew Cab rather than one cab	+1445	+1525	+1600
• Lunar Descent Droptanks Included	+960	+1225	+1285
• Abandon Lunar Landing Gear	-530	+1530	+1605
• Radiation Protection Water Stowed In LLO with Aerobrake (1.8 mt)	N/A	+2275	+2345
• Engine Specific Impulse variations from an ASE Engine (481s/ 465s)			
• Enhanced RL10 (460s/ 445s nozzle retracted)	-3595	-3645	-3265
• "Current" RL10 (449s)	-4650	-4725	-4135
• Including effects of Power Reactants Consumed by Fuel Cells			
• Fuel Cells consume Propellant directly (assumed 2.38 KG/hr)	-4500	-4500	-4500
• Fuel Cells consume Propellant Boiloff prior to direct consumption	-2200	-2200	-2200
• Direct Lunar Landing Scenario for the Expendable Cargo Mission	+3395	+3395	+2000

* NOTE: Stated Mass Deltas assume all other variables are constant and are relative approximations

Figure 1-3.1-26. Additional Trade Effects on Mission Performance

STV CONCLUSIONS & RECOMMENDATIONS
SYSTEM CONCEPTS & EVOLUTION FAMILIES
MSFC- *BOEING* ————————————————————————————

PERFORMANCE ANALYSIS CONCLUSIONS & RECOMMENDATIONS

OVERALL PERFORMANCE CONCLUSIONS :

- Family A concepts provide the best overall performance (current performance front-runners)
- Family A configurations can be retofitted to a Family B configuration, but not the converse
- Fuel Cell consumption of propellant will significantly impact performance; further study pending
- Shorten LEO staytime to reduce propellant boiloff losses
- Higher Specific Impulse Engines produce a significantly higher performance
- Cargo Expendable Missions increase performance using a Direct Lunar Landing scenario

PERFORMANCE RECOMMENDATIONS BY FAMILY :

FAMILY A (Two Vehicles, LEV & LTV, Two P/A modules) :
- Use Two Crew Cabs, one for each vehicle, rather than a single cab transfered in LLO
- LEV should be fully reusable (no Lunar descent droptanks & no expendable lander legs, to avoid assembly and maintenance operations in LLO)

FAMILY B (One Vehicle, LTV, One P/A module) :
- Use a Hybrid Cab rather than either one or two independent crew cabs
- Radiation Protection Water should be stowed in LLO with the Aerobrake during the Lunar stay
- Lander should be fully staged with Lunar descent droptanks & expendable lander legs (subject to a pending cost analysis)

FAMILY C (One Vehicle, LTV, Two P/A modules) :
- All provide poor overall performance (currently the bottom of the list of considered concepts)

Figure 1-3.1-27. Performance Analysis Conclusions and Recommendations

the vehicle to capture a specific mission. These vehicle incarnations are shown in Figures 1-3.1-29 through Figures 1-3.1-34.

During the course of this work, it became important to assess the sensitivity of the STV to weight growth at various points through the lunar mission. As shown in Figure 1-3.1-35, the location of the increased weight determines the overall system impact. This turned out to be the case because of the use of staging and droptanks through the mission. If the weight growth is in a component that is dropped early (e.g., TLI tanks), the system impact is less than if the growth is in a component used for the entire mission duration (e.g., the crew module).

After this phase of the study, there were significant criteria not being taken into account in this analysis. As discussed earlier (section 2-1.1), the new criteria used to assess the configuration are cost, margins and risk, other mission capture, and benefits to the Mars mission. With this change in configuration scoring, it became apparent that other mission architectures needed to be considered. The configuration work during this phase was focused on finding the configuration differences between the architectures. Following this architecture study (section 2-1.1.4) the configurations were narrowed to a three distinct configurations (Figure 1-3.1-36).

The first of these configurations was the GB (ground-based) configuration Figure 1-3.1-37). This vehicle is fully integrated on the ground and launched aboard one very large launch vehicle ($\approx$250 metric tons to LEO). This vehicle has a very similar operational concept to that used for the Apollo missions. The majority of the vehicle is expended with only the crew module being returned to the Earth, where it reenters and recovers to the ground. It is important to note that in this ground return the vehicle is maneuverable enough to land at a prepared landing site. This is possible primarily because the final descent is performed under a parafoil, which allows the descent to be very accurately controlled.

The second configuration was the GO (ground-based, orbitally assembled) configuration (Figure 1-3.1-38). In this vehicle, a much smaller launch vehicle ($\approx$71 metric tons) is used to place the vehicle component parts in orbit. These components then rendezvous and self-assemble (Figure 1-3.1-39). As can be
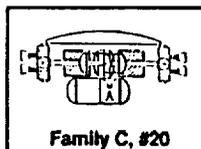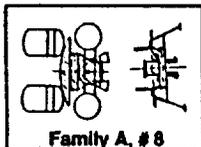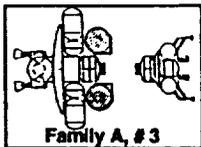
CONCEPT	REASONS
 Family A, #3	• High IMLEO Propellant Requirement/ Poor Lunar Cargo Delivery Capability - This concept requires both large droptanks to be retained though the Lunar Orbit Insertion burn to avoid Aerobrake penetrations. Added mass during LOI degrades performance.
 Family A, #8	• High IMLEO Propellant Requirement/ Poor Lunar Cargo Delivery Capability - A single crew cab transfered in Low Lunar Orbit must utilize the LTV crew cab, rather than the smaller LEV cab. Payload mass to the lunar surface is reduced due to the larger crew cab.
 Family C, #19	• High IMLEO Propellant Requirement/ Poor Lunar Cargo Delivery Capability - Family C dual P/A module designs require structural spars and added mechanisms to translate or pivot the two P/A modules on the spars. This additional mass is present during all mission phases and significantly impacts performance.
 Family C, #20	• High IMLEO Propellant Requirement/ Poor Lunar Cargo Delivery Capability - Same reasons as above..

Figure 1-3.1-28. Concept Downselection Candidates
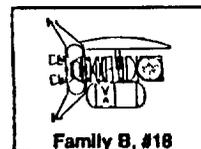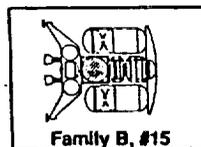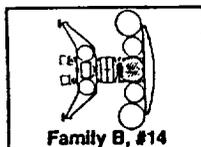
CONCEPT	REASONS
 Family B, #14	• Poor Mission Abort Capability - Containerized cargo cannot be easily jettisoned during some mission abort scenarios.
 Family B, #15	• No Mission Abort Capability - Containerized cargo cannot be jettisoned during lunar landing mission abort scenarios.
 Family B, #18	• Poor docking & cargo transfer capability - A close fit with the large aerobrake creates interference during docking maneuvers. Also, additional mass from the aerobrake pivot and locking mechanisms will marginally decrease deliverable payload performance.

Figure 1-3.1-29. Concept Downselection Candidates

Mission: - P1. Launch 16000kg to Planetary Unmanned , - Single P/L
G1. Deliver 10000kg to GEO - Unmanned, - Single P/L
M1 Deliver 5000 KG (molniya) - Unmanned, - Single P/L

	CONFIG. #1	CONFIG. #9	CONFIG. #19
LEAVE SSF			
DEPOSIT P/L			
RETURN TO EARTH/SSF			
REMOVE STRUCTURE FROM LUNAR CONFIG.	Entire LEV + 1 crew module. 2 P/L supports 1 set (2) tanks	Crew module, 1 set lunar tanks, landing legs, 1 set (2) maintanks 2 P/L supports	Crew module, 1 set (2) tanks, landing legs
NEW STRUCTURE FOR SECONDARY MISSION	Single P/L support in place of crew module	Single P/L supt. above aerobrake, structure between P/A module & aerobrake	Structure to attach main beams to aerobrake

Figure 1-3.1-30. Payload Delivery Mission

Mission: - G2: - Servicing Mission at 35750 km orbit - manned
S1: Servicing Mission at 824 KM Orbit - manned

	CONFIG. #1	CONFIG. #9	CONFIG. #19
LEAVE SSF			
SERVICE S/C			
RETURN TO EARTH/SSF			
REMOVE STRUCTURE FROM LUNAR CONFIG.	Entire LEV 1 set tanks (2) 2 P/L supports	2 P/L supports 1 set tanks (2) lunar descent tanks lander legs	1 set tanks (2) lander legs
NEW STRUCTURE FOR SECONDARY MISSION	Remote maniput. arm manned maneuvering unit. Propellant transfer capability	Remote manipul. arm manned maneuvering unit propellant transfer capability	Remote manip. arm manned maneuvering unit propellant transfer capability

Figure 1-3.1-31. Spacecraft Servicing Mission

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Mission C1: - Retrieve 500kg from a 407KM x 62968 km elliptical orbit,manned.
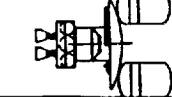
	CONFIG. #1	CONFIG. #9	CONFIG. #19
LEAVE SSF			
RETRIEVE P/L			
RETURN TO EARTH/SSF			
REMOVE STRUCTURE FROM LUNAR CONFIG.	Entire LEV 1 set tanks (2) 2 P/L supports	2 P/L supports 1 set tanks (2) lunar descent tanks lander legs	1 set tanks (2) lander legs
NEW STRUCTURE FOR SECONDARY MISSION	Support for small P/L remote manipul. arm manned maneuvering unit	Support for small P/L remote manipulator arm manned maneuvering unit	Support for smallP/L remote manipulator arm manned maneuvering unit

Figure 1-3.1-32. Small Payload Retrieval Mission

Mission: - T1. Retrieve 71 t from LEO - Unmanned, No Aerobrake Entry

	CONFIG. #1	CONFIG. #9	CONFIG. #19
LEAVE SSF			
RETRIEVE P/L			
RETURN TO SSF			
REMOVE STRUCTURE FROM LUNAR CONFIG.	All components except for RCS thrusters & RCS tanks	All components except for RCS thrusters & RCS tanks	All components except for RCS thrusters & RCS tanks
NEW STRUCTURE FOR SECONDARY MISSION	Payload Retrieval & Support Structure	Payload Retrieval & Support Structure	Payload Retrieval & Support Structure

Figure 1-3.1-33. Payload Retrieval Mission

BOEING

Mission D1: - Capture 2500Kg at a 1296km orbit & boost to solar orbit - unmanned & expendable
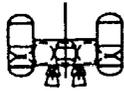
	CONFIG. #1	CONFIG. #9	CONFIG. #19
LEAVE SSF			
CAPTURE P/L & BOOST TO ESCAPE ORBIT			
REMOVE STRUCTURE FROM LUNAR CONFIG.	1 set (2) tanks, entire LEV, 1 crew module 2 P/L supts., aerobrake.	1 set (2) tanks, landing legs, lunar descent tanks, 2 P/L supports, crew module, aerobrake	1 set (2) tanks, landing legs, aerobrake, crew module
NEW STRUCTURE FOR SECONDARY MISSION	Single P/L retrieval & supt. sturcture in place of crew cab.	Structure for tank support (2), P/L retrieval & support structure in place of crew cab	Structure for tank support (2) P/L retrieval mechanism
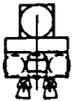
Figure 1-3.1-34. Debris Disposal Mission

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STV CARGO SENSITIVITY TO INERT WEIGHT
System Concepts and Evolution Families
MSFC- *BOEING*

VEHICLE CONCEPT 1 - STEADY STATE PILOTED MISSION
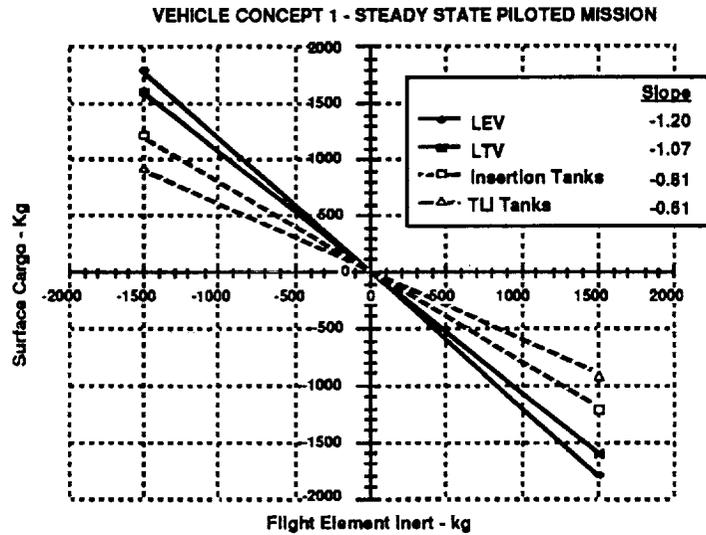
Figure 1-3.1-35.   Vehicle Concept 1 - Steady-State Piloted Mission

STV STV CONFIGURATIONS
MSFC- *BOEING*
Vehicle Design
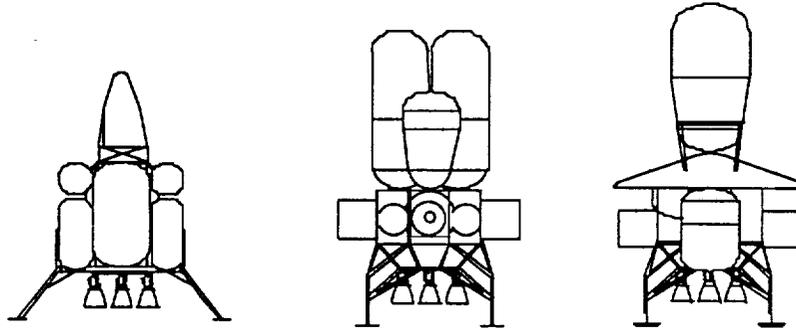
Ground based (GB)
IMLEO   235 mt (manned)
        259 mt (unmanned)
Cargo   11.1 mt (manned)
        46.4 mt (unmanned)

Ground based/orbital assy (GO)
IMLEO   244 mt (manned)
        272 mt (unmanned)
Cargo   10.6 mt (manned)
        48.7 mt (unmanned)

Space based (SB)
IMLEO   255 mt (manned)
        288 mt (unmanned)
Cargo   9.9 mt (manned)
        52.77 mt (unmanned)

Figure 1-3.1-36. STV Configurations

STV GROUND BASED (GB) CONFIGURATION
MSFC- *BOEING*
Vehicle Design

Crew compartment

66.5 ft

58.6 ft
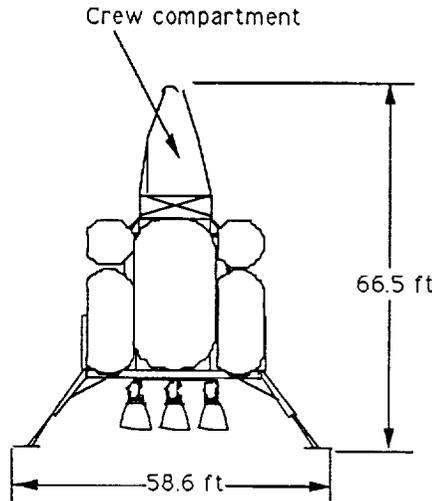
Features
40 ft x 15 ft dia payload envelope
TLI droptanks
Descent LH2 droptanks
Self-unloadable
6 15klbs thrust engines
Crew module land recoverable
Crew modules refurbishable

Figure 1-3.1-37. Ground-Based Configuration

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STV GROUND BASED/ORBITAL ASSEMBLY VEHICLE

MSFC- *BOEING* ─────────── Vehicle Design ───────────
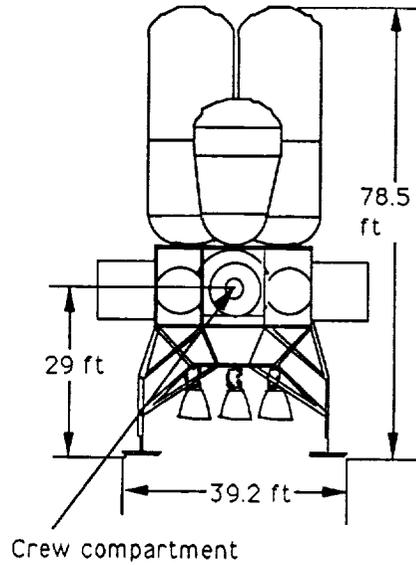
Features

Heaviest element 61 mt
(TLI tankset)

30 ft dia launch shroud

6 15klbs thrust engine

15 ft dia x ∞ cargo
envelope

Self unloadable

Crew module ground
recoverable

78.5 ft

29 ft

39.2 ft

Crew compartment

Figure 1-3.1-38. Ground-Based and Orbital Assembly Vehicle

GO ASSEMBLY SEQUENCE

Vehicle Design

STV

MSFC- *BOEING*

1st launch
Core stage
Empty crew module

2nd launch
TLI tank

3rd launch
TLI tank

4th launch
Descent tanks

5th launch
Crew
Cargo
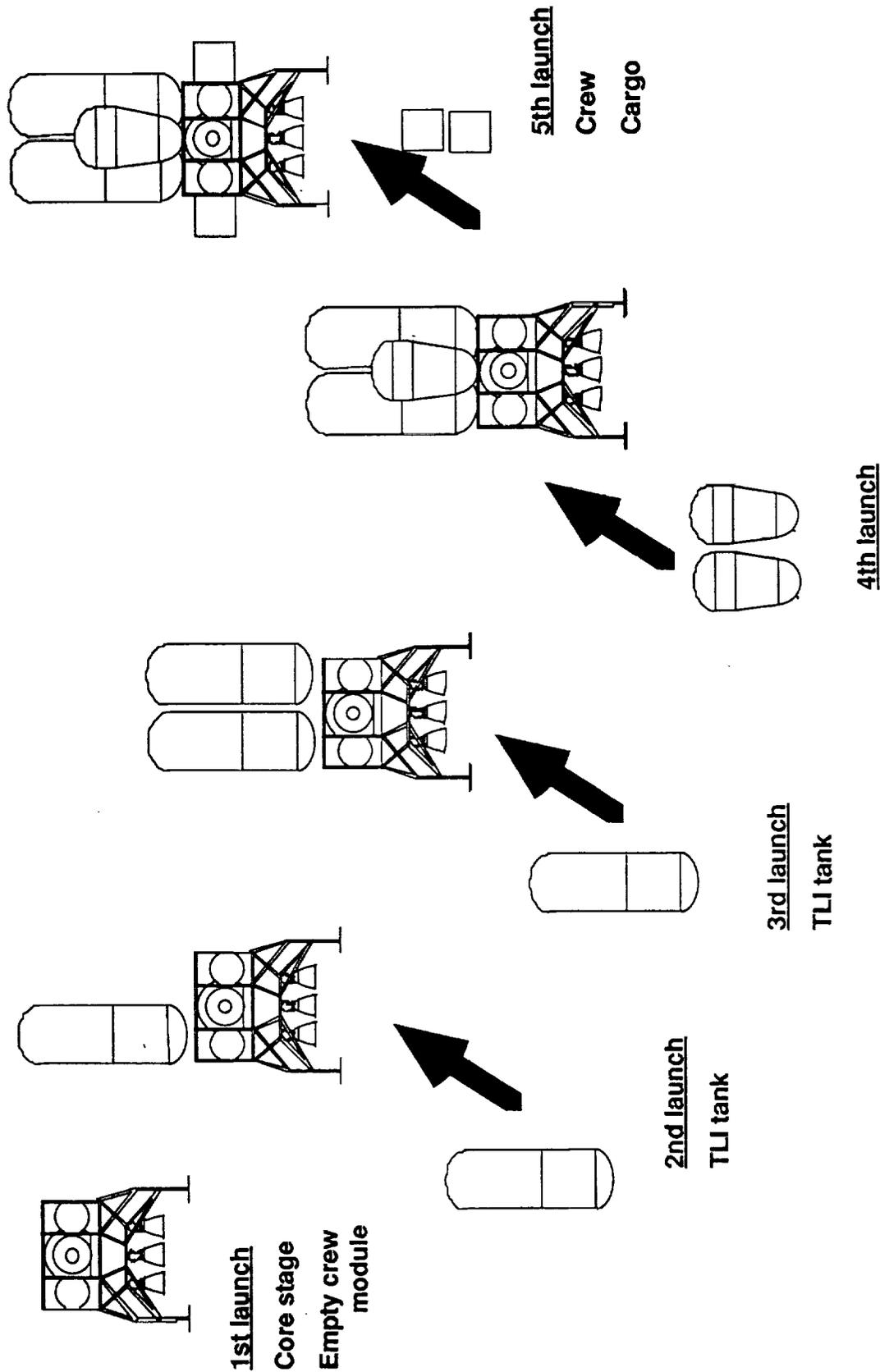
Figure 1-3.1-39. GO Assembly Sequence

seen, after the vehicle has assembled and self-tested itself, the crew rendezvous with the vehicle and the mission begins. Upon completion of the mission, this vehicle is expended except for the crew module, which (like the GB configuration) recovers to a prepared landing site.

The final vehicle considered was the SB (space-based) configuration (Figure 1-3.1-40). Based at the Space Station, this vehicle carries four droptank sets (two TLI and two descent tanksets). With the exception of these however, the vehicle is completely reusable. The most obvious feature of this configuration is the aerobrake. Following the completion of the mission, this vehicle aerobrakes back to the Space Station where it is refurbished and readied for the next flight.

While these configurations were being developed, concerns about the GO configuration began to arise. The most significant of these concerns was the cost and complexity of the equipment necessary for this vehicle to self-assemble. Also, more than 50% of the entire vehicle mass was the LOX carried on board. Using this information, the three configurations shown earlier were refined into just two configurations (Figure 1-3.1-41). In this new scheme of two configurations, the three options of GB, GO, and SB were maintained by using three unique operational concepts. The GB and SB configurations and operational concepts remain the same. The GO configuration is a variation of the GB configuration. In the new GO, the vehicle is identical to the GB vehicle, but the GO is launched with only 20% of its LOX load. The remaining LOX is launched aboard a separate tanker, which rendezvous with the main vehicle and transfers the remaining 80% of the LOX required to perform the mission (Figure 1-3.1-42). This split of STV and tanker allows the system to avoid any on orbit assembly problems associated with the first GO configuration, yet can be launched in a smaller launch vehicle (≈125 metric tons).

1-3.1.1   Lunar Surface Configuration

During the course of the mission model used in the STV study, the vehicle must have the capability to land 418 metric tons to the lunar surface (section 2-1.4). The flexibility in cargo packaging determines how much flexibility in the vehicle design is allowed. Cargo packaging and location become more important considerations on the ground-based vehicles. This is because on these
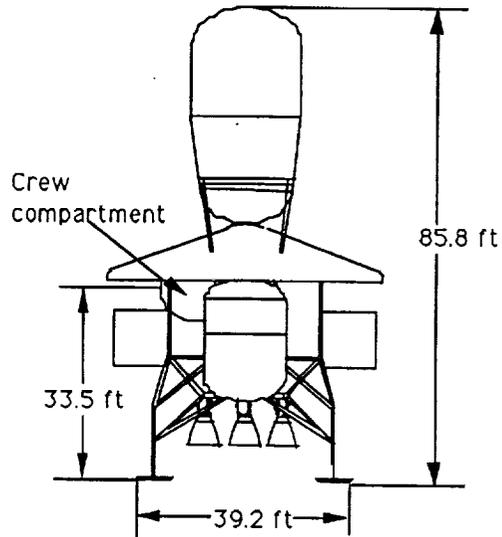
STV

SPACE BASED VEHICLE

Vehicle Design

MSFC- *BOEING*

Features

50 ft dia aerobrake

Largest element 65 mt (TLI tankset)

Reentry L/D >.2

Asymmetric vehicle

Launchable in 30 ft shroud

15 ft x ∞ cargo envelope

Recovery to SSF

Crew module fits in Shuttle cargo bay

Self unloadable

Figure 1-3.1-40. Space-Based Vehicle

STV Concepts
Configuration Splinter

MSFC- **BOEING**

STV
Space Transfer Vehicle

**Ground Based
Single Launch**
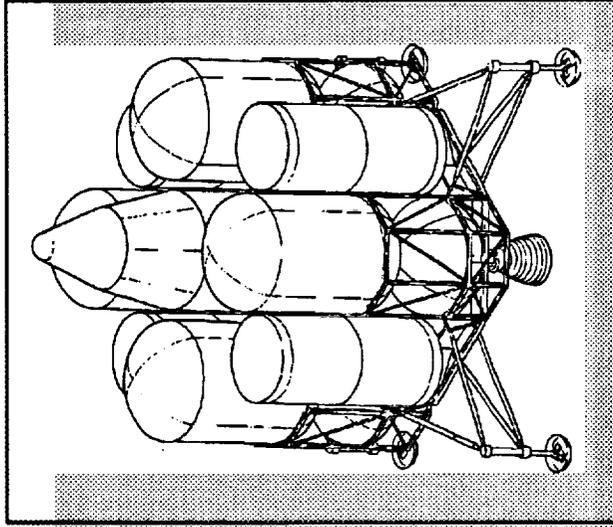
1 Launches
250 mt booster
No assembly
required

**Ground Based
Multiple Launch**
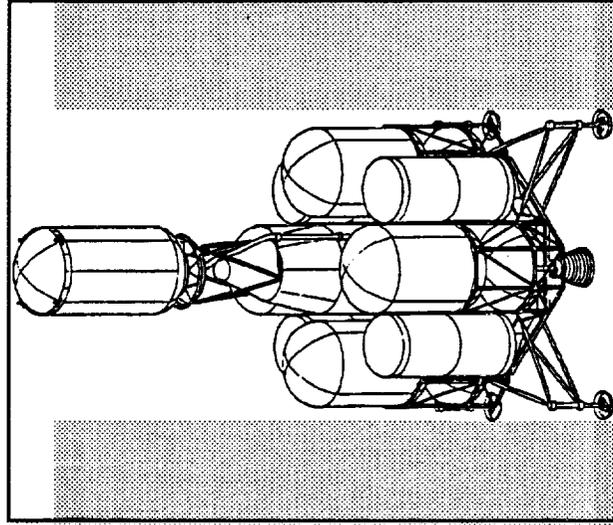
2 Launches
125 mt booster
Docking and
refuelling

Space Based (SB)
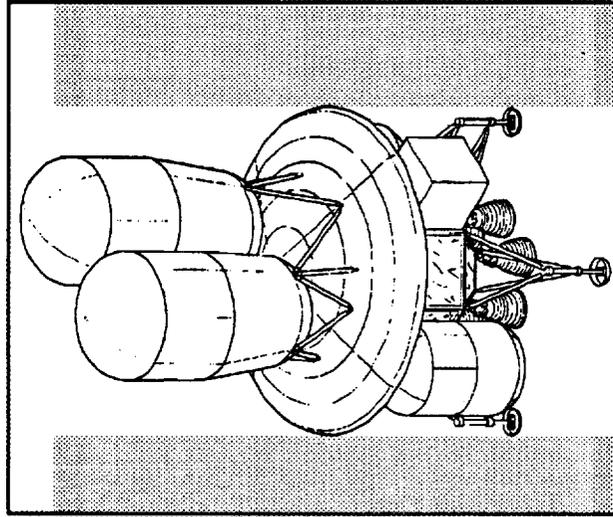
5 Launches
70 mt booster
Assembly
required

Figure 1-3.1-41. STV Concepts

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BOEING

STV
Space Transfer Vehicle

Minimize On Orbit Assembly
Configuration Splinter

MSFC- BOEING
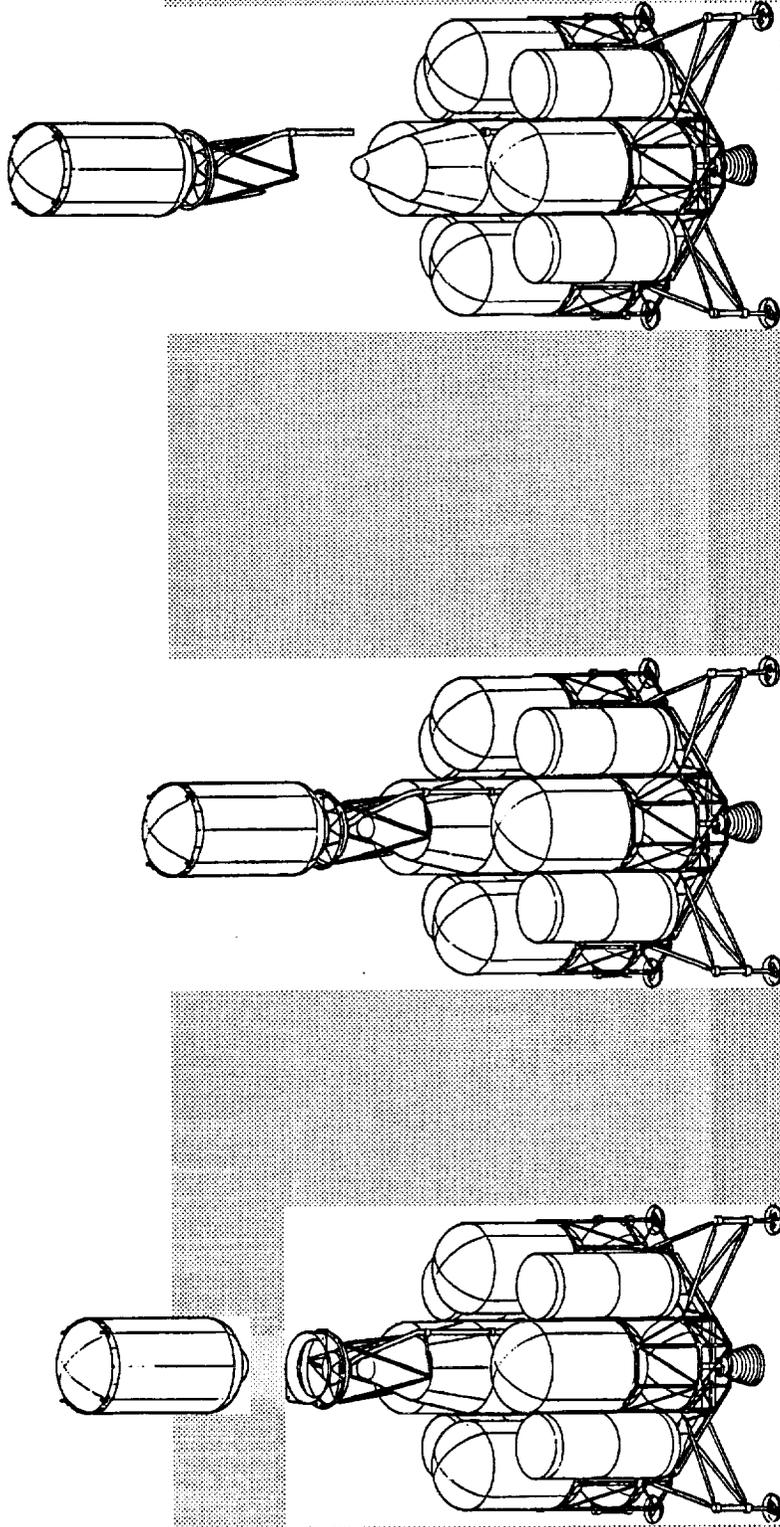
- Dock with tanker using LES support structure
- Perform settling burn
- Transfer liquid oxygen
- Disconnect LES / tanker berthing structure & transfer umbilical
- Back STV away from tanker
- Tanker performs deorbit burn

Figure 1-3.1-42. Minimize On-Orbit Assembly

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vehicles the cargo is launched from the ground integrated with the rest of the vehicle. This integration must in no way interfer with the launch escape system, and at the same time it must allow clear visibility of the lunar surface during landing (Figure 1-3.1.1-1).

As currently envisioned by the planetary surface system studies, the STV will have a payload unloader capable of unloading the vehicle on the lunar surface as is shown in Figure 1-3.1.1-2. This, in effect, determines what the STV surface configuration will be because this unloader is incapable of unloading many of the vehicles envisioned during the course of the study. To alleviate these limitations, work was undertaken to determine what the options were for unloading the configurations. These options are shown in Figures 1-3.1.1-3 through 1-3.1.1-6. Notice that the "lumber carrier" described in Figure 1-3.1.1-4 is essentially a variation of the payload unloader described earlier. How well these payload unloading options would integrate into the configuration is shown by two possible configurations (Figure 1-3.1.1-7 and 1-3.1.1-8).

1-3.1.2 Aeromaneuver Configurations

The two configurations assessed in this study, the ground based (GB, GO) and the space based (GB), have very different means of returning to the Earth. As was previously discussed, the ground-based configuration (upon leaving the Moon) does not establish any type of Earth orbit but instead directly enters the Earth's atmosphere and recovers to the ground (Figure 1-3.1.2-1).

The space-based configuration, upon entering the Earth's vicinity, enters the atmosphere and uses the atmosphere to bleed off some of the vehicle's energy and allow it to achieve a stable orbit at the same altitude as the Space Station (Figure 1-3.2.1-2).

Critical to both of these return configurations is the portion that executes the aeromaneuver. Although these aeromaneuvers are very different with one being an aerobraking maneuver and the other an atmospheric reentry, the problems associated with the aeromaneuver are the same. During this period of atmospheric flight, the vehicle uses the atmosphere to dissipate the majority of the energy gained during cislunar flight. As the vehicle enters the atmosphere, it

STV
Space Transfer Vehicle ━━ MSFC- *BOEING* ━━

Ground Based Cargo Layout
Payload/PSS Interfaces

Crew FOV

□ - Foot Pads

Cargo

TLI Drop Tanks
Jettisoned

Ground base vehicle design influenced
by requirement to see two foot pads
and horizon during Lunar landing
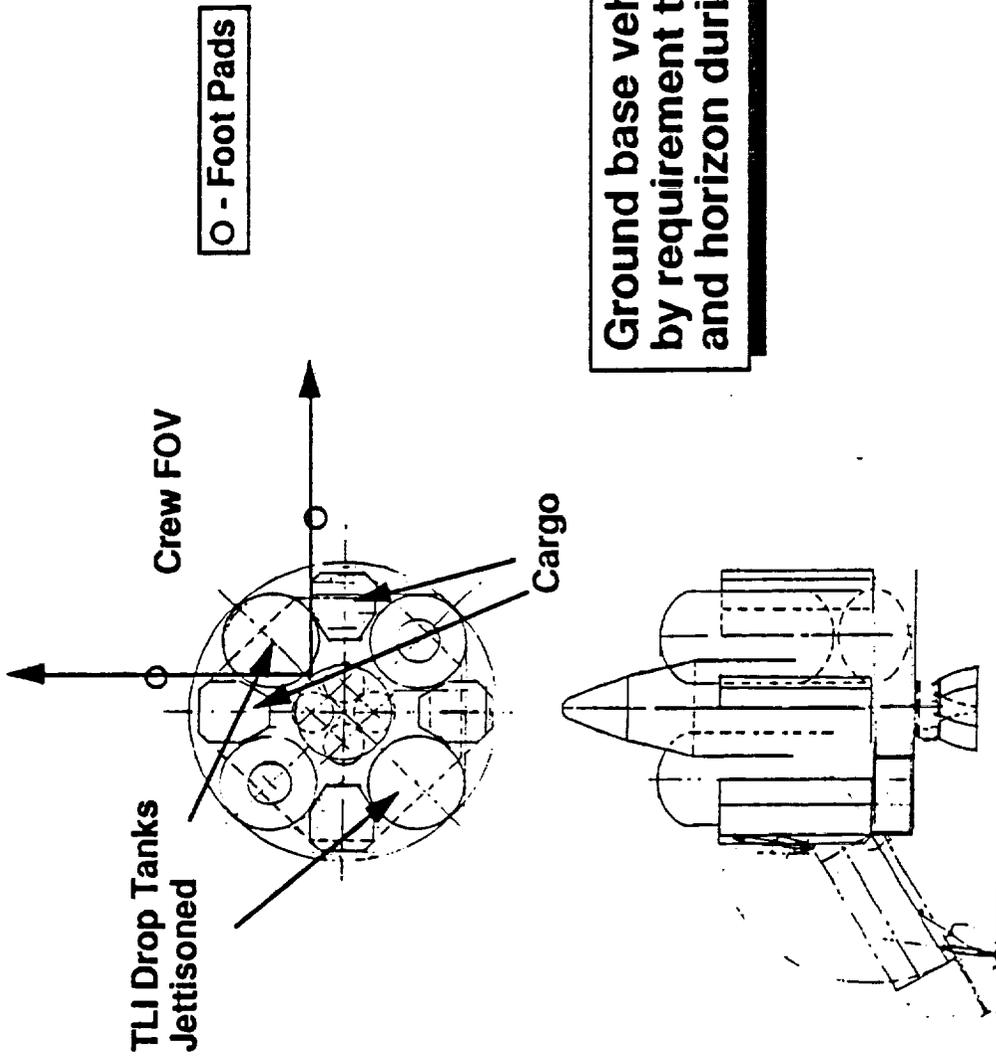
Ground Based Vehicle

Figure 1-3.1.1-1. Ground-Based Cargo Layout

STV

MSFC- *BOEING*

PAYLOAD UNLOADER
Vehicle Design
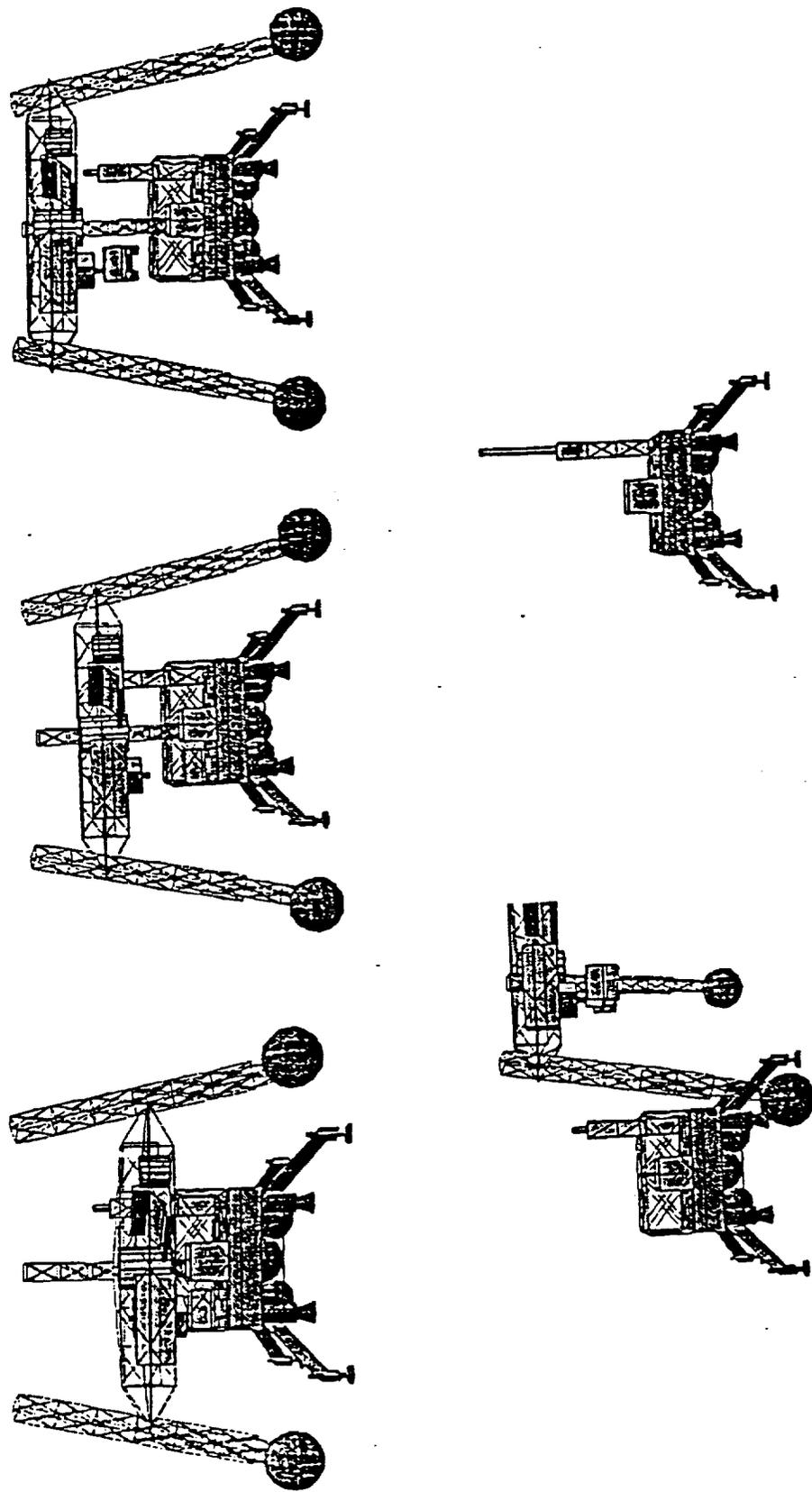
Figure 1-3.1.1-2. Payload Unloader

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Payload Unloading Options - 1
Payload/PSS Interfaces —— MSFC- *BOEING*—

STV
Space Transfer Vehicle

| Self Unloading / Dual Modules |
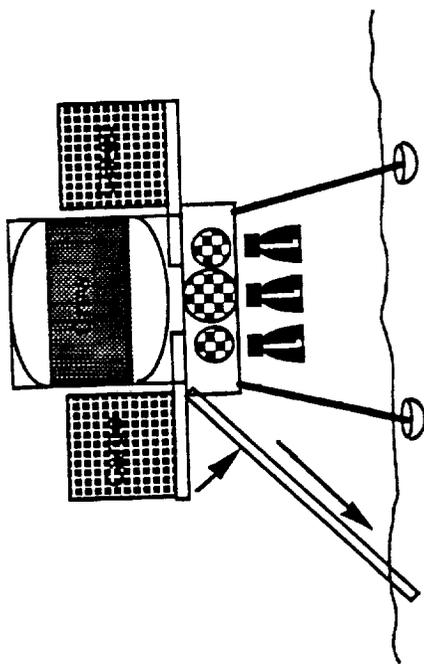
Self-Contained Ramp
- Accommodates cargo transfer
- Ramp weight
- Cargo higher – better in-flight stability
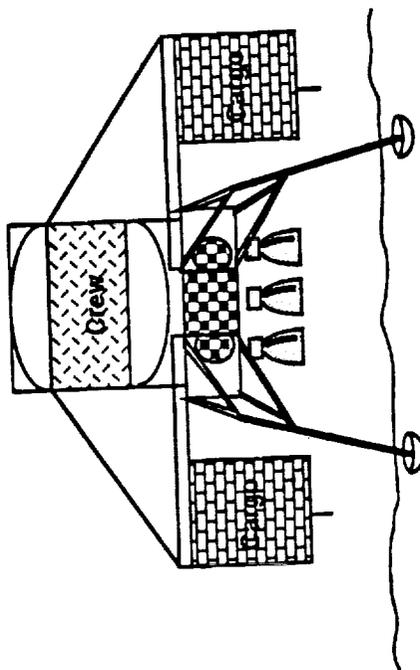
Simple Lowering
- Simplest concept
- Room for additional fuel
- Cargo low for ground stability

Figure 1-3.1.1-3. Payload Unloading Options .-1

Payload Unloading Options - 2
Payload/PSS Interfaces

MSFC- *BOEING*

STV
Space Transfer Vehicle

Lumber Carrier Unloading

Crew
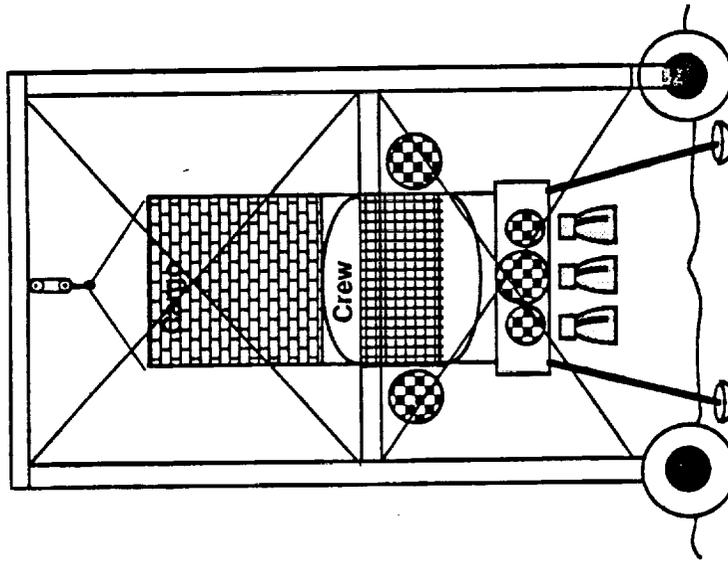
- Requires prepared surface
- Allows flexible placement
- Abort jettison
- Cargo flexibility
- In-flight stability

Backpacker Unloading

FOV
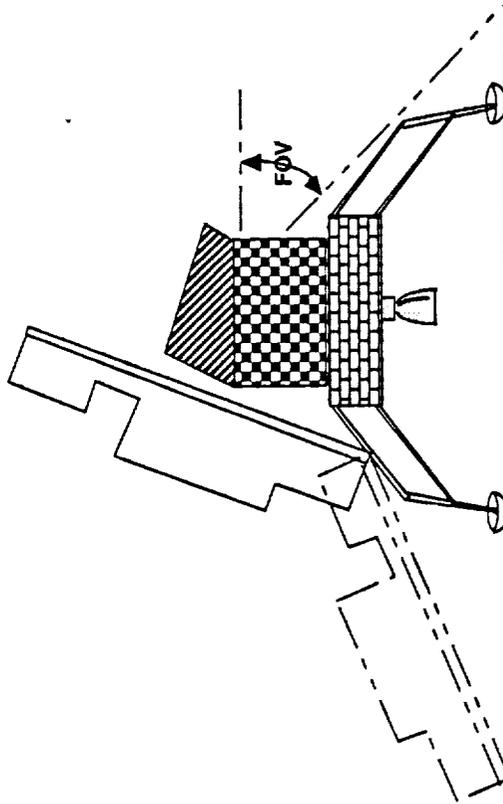
- Landing flexibility
- Not sensitive to cargo wt. or length
- Ramp weight
- CG control
- Abort jettison

Figure 1-3.1.1-4. Payload Unloading Options -2

BOEING

STV
Space Transfer Vehicle

Payload Unloading Options - 3
Payload/PSS Interfaces

MSFC- *BOEING*

A-Frame Integral Crane
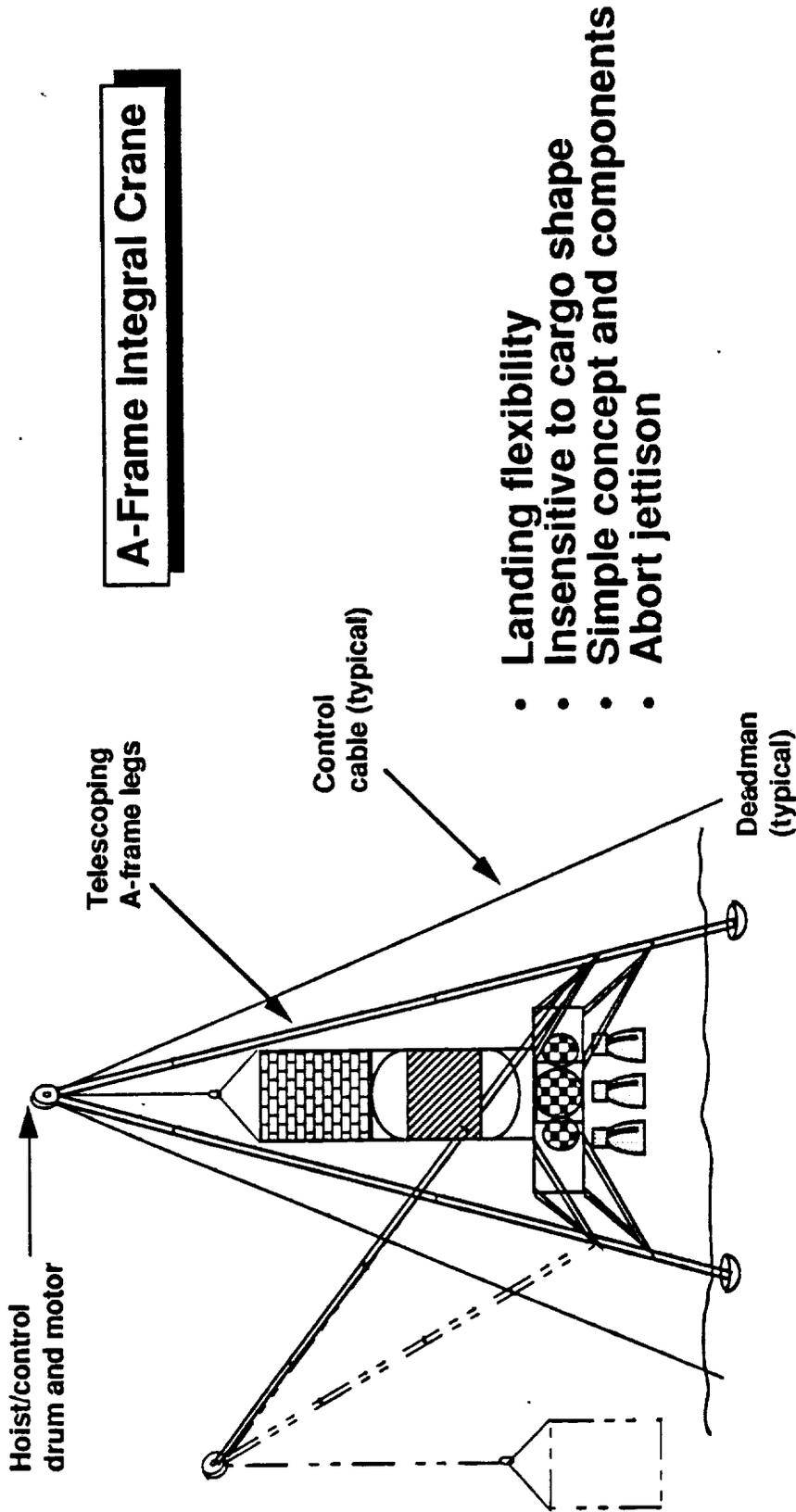
- Landing flexibility
- Insensitive to cargo shape
- Simple concept and components
- Abort jettison

Telescoping
A-frame legs

Control
cable (typical)

Hoist/control
drum and motor

Deadman
(typical)

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Figure 1-3.1.1-5. Payload Unloading Options -3

JSC Unloading Scheme
Payload/PSS Interfaces

MSFC - *BOEING*

STV
Space Transfer Vehicle
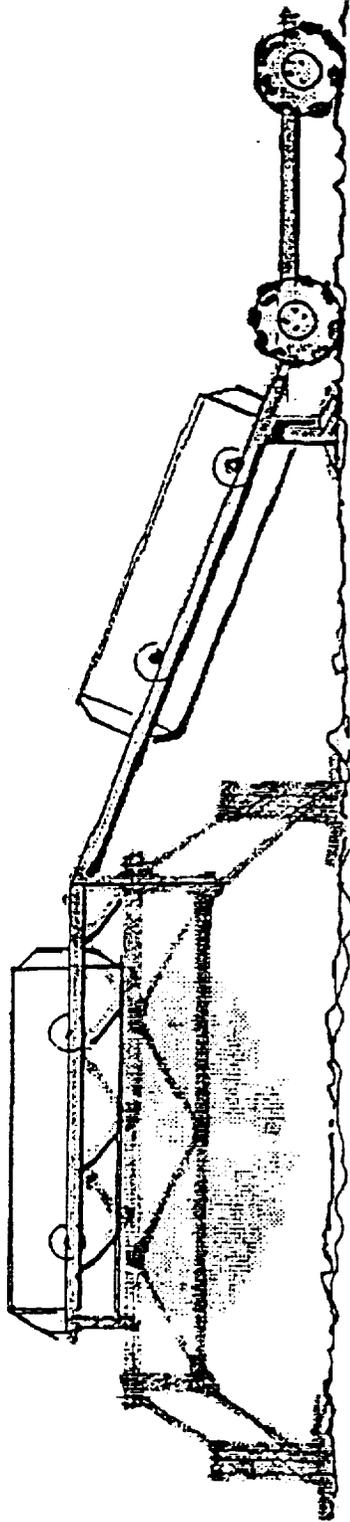
- Alternate to payload unloader
- PSS provided ramp and cart with rails
- Uses existing trunion pins

Figure 1-3.1.1-6. JSC Unloading Scheme
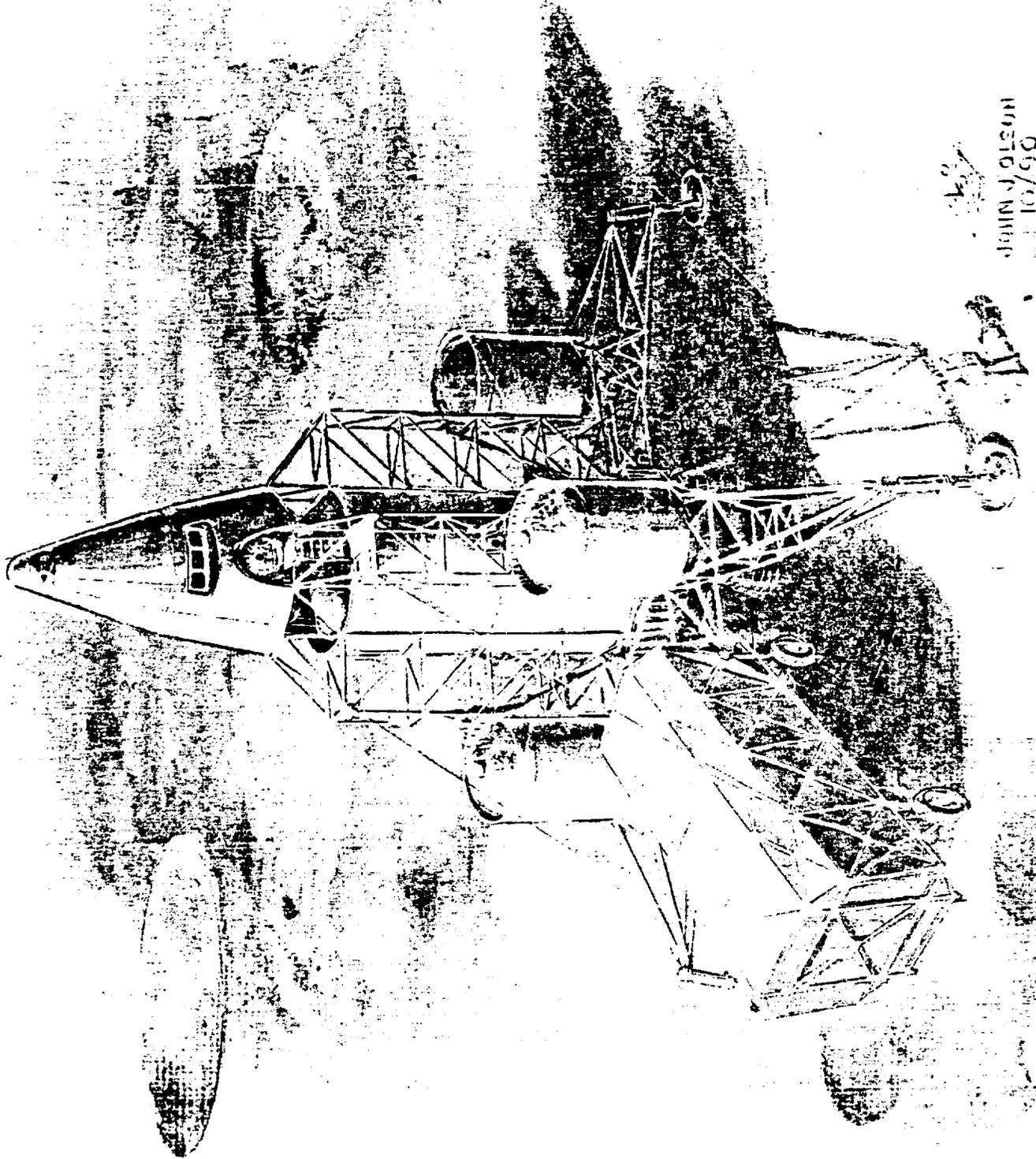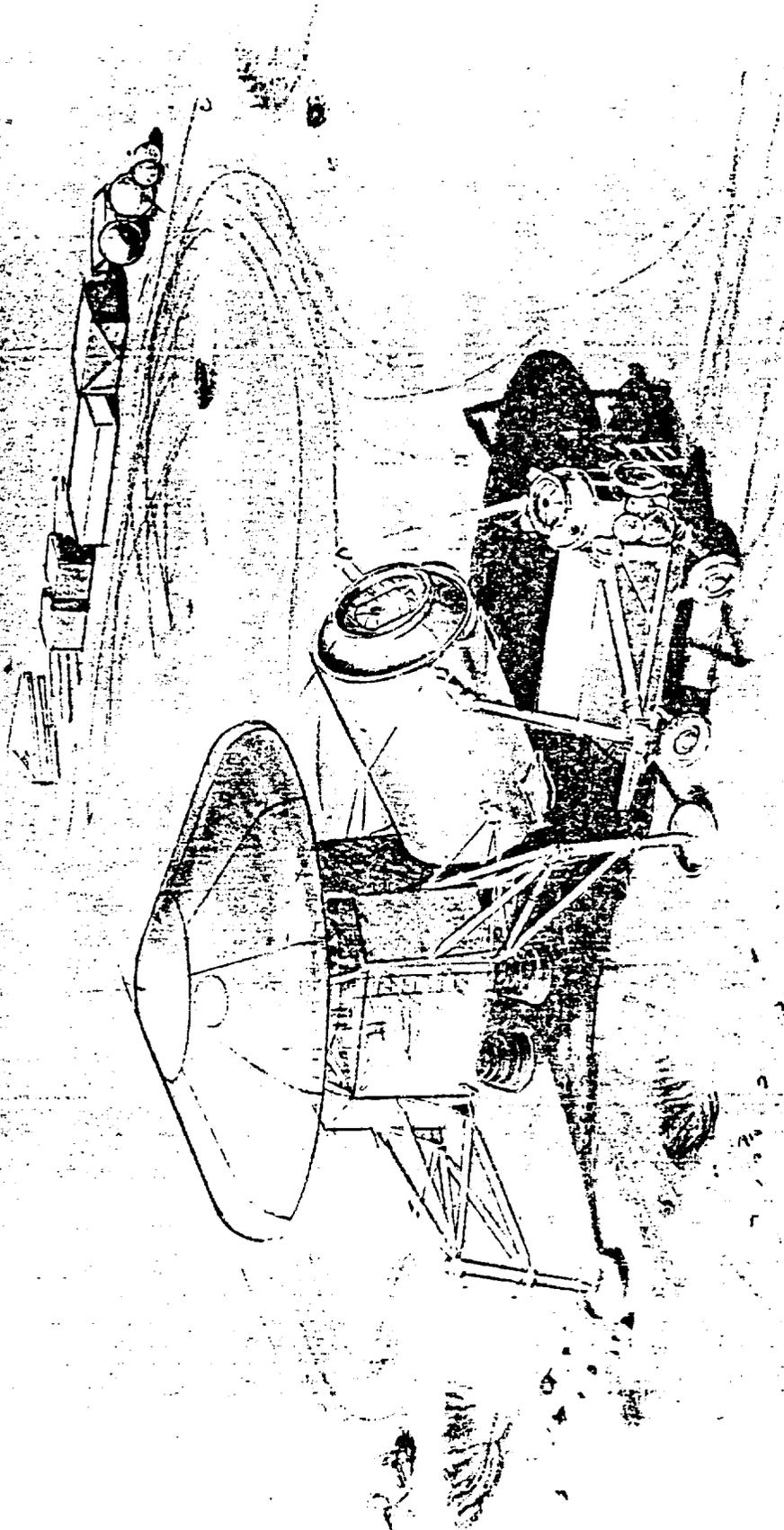
Figure 1-3.1.1-7.

Figure 1-3.1.1-8.

Space Based Bomber — Undocking
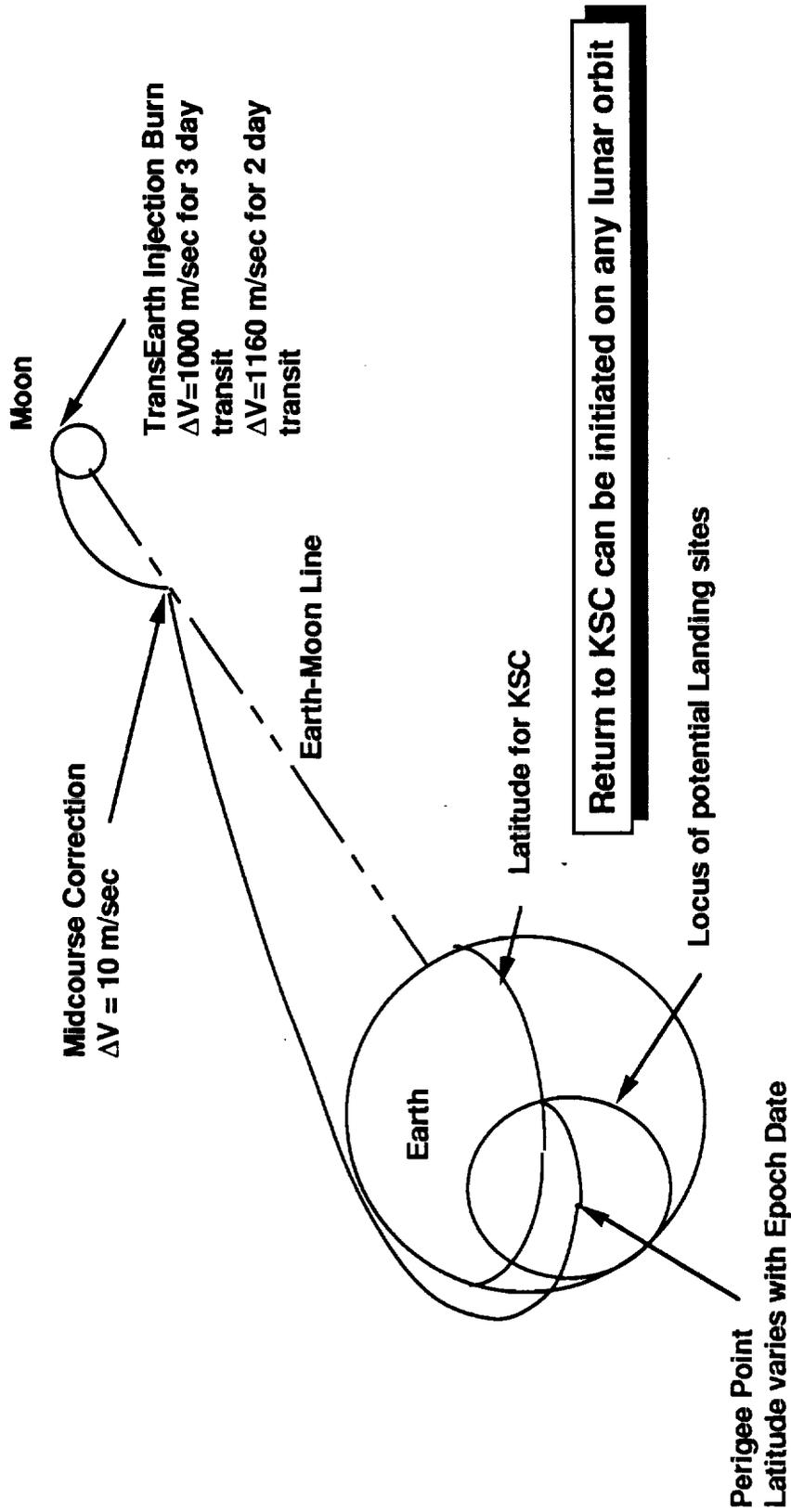
Figure 1-3.1.2-1. *Ground-Based Lunar Return Trajectory*

Moon

TransEarth Injection Burn
ΔV=1000 m/sec for 3 day transit
ΔV=1160 m/sec for 2 day transit

Midcourse Correction
ΔV = 10 m/sec

Earth-Moon Line

Latitude for KSC

Return to KSC can be initiated on any lunar orbit

Locus of potential Landing sites

Earth

Perigee Point
Latitude varies with Epoch Date
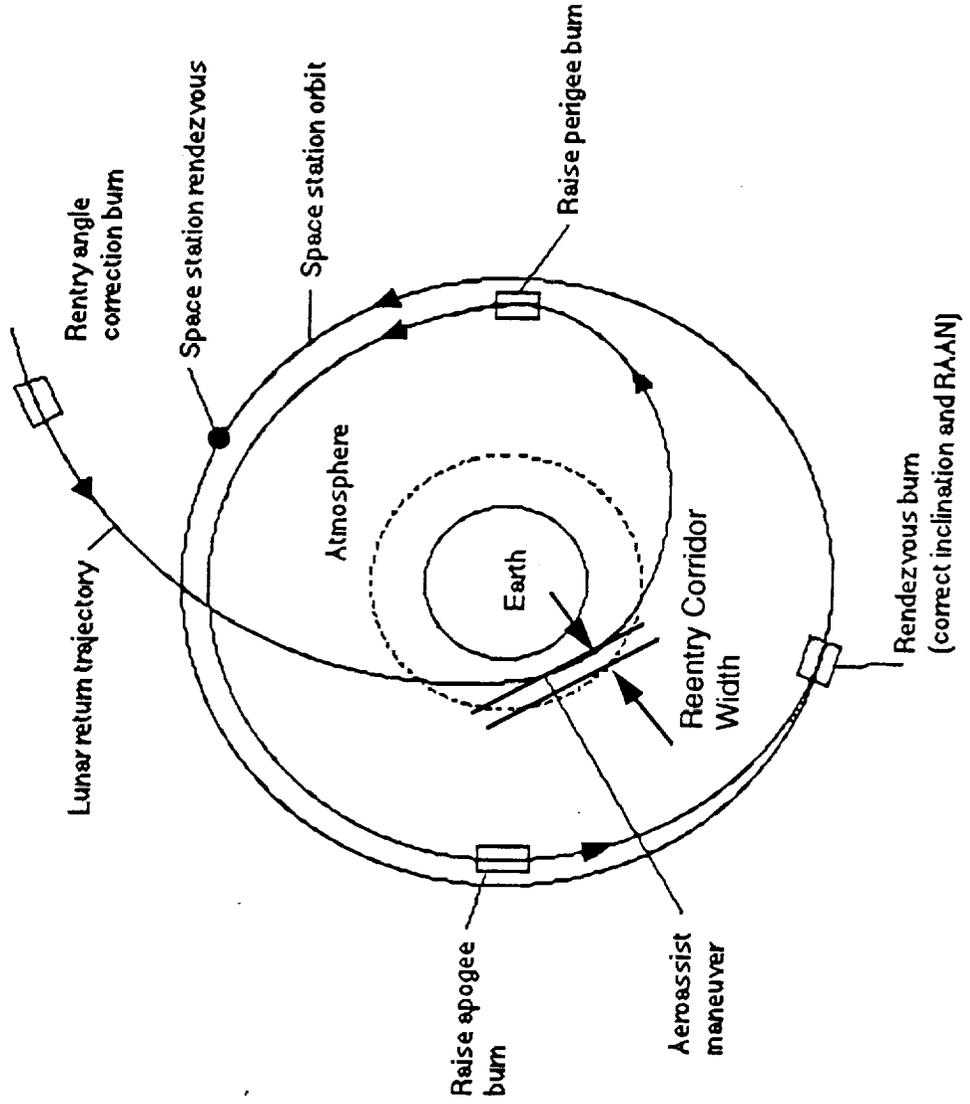
Figure 1-3.1.2-2.

is subjected to very high thermal loads, the highest dynamic pressures experienced during the mission, and possibly high gravity loads.

The difference in the mission for each of these vehicles determines the shape of the aerodynamic configuration. The space-based vehicle is designed for the entire vehicle (with the exception of droptanks) to be refurbished and reused. To meet this goal, the configuration that reenters is required to protect a large mass and volume during the reentry. To meet the large volume return goal, the aeromaneuver configuration chosen was an aerobrake, which will be discussed in more detail shortly.

For the ground-based vehicle, the only portion of the vehicle that is refurbished and reused is the crew compartment. Therefore, this configuration does not have the requirement for a large return volume. The major constraint on this configuration is that it must be integrated into a launch vehicle on the ground.

Aerobrake Configuration. On the space-based configuration, several issues had to be addressed in the aerobrake design. These issues included launch vehicle integration and its impact on the aerobrake design, the structural concept used, and subsequently the TPS scheme that goes along with that structural concept. Along with those overall configuration issues, other issues such as the penetrations through the aerobrake and the refurbishment and reuse of the aerobrake have to be assessed to develop an operational concept.

Lift-to-Drag Trends. The aerobrake used in the space-based configuration has the goal of reducing the overall system weight by reducing the propellant load required to achieve a stable orbit at the Space Station altitude. To achieve this goal, it is necessary to keep the aerobrake mass fraction to a minimum. Minimizing the aerobrake mass comes primarily from two sources: reducing the aerobrake structural mass and reducing the aerobrake TPS mass. In achieving either of these mass reductions, it is necessary to determine what impact the aerobrake performance has on either the aeroheating (and its subsequent impact on TPS mass) or the impact on the aerodynamic loading (and that impact on the structural mass of the brake). As can be seen in Figure 1-3.1.2-3 for a given aerobrake ballistic coefficient (=W/CdA), dynamic pressure, load
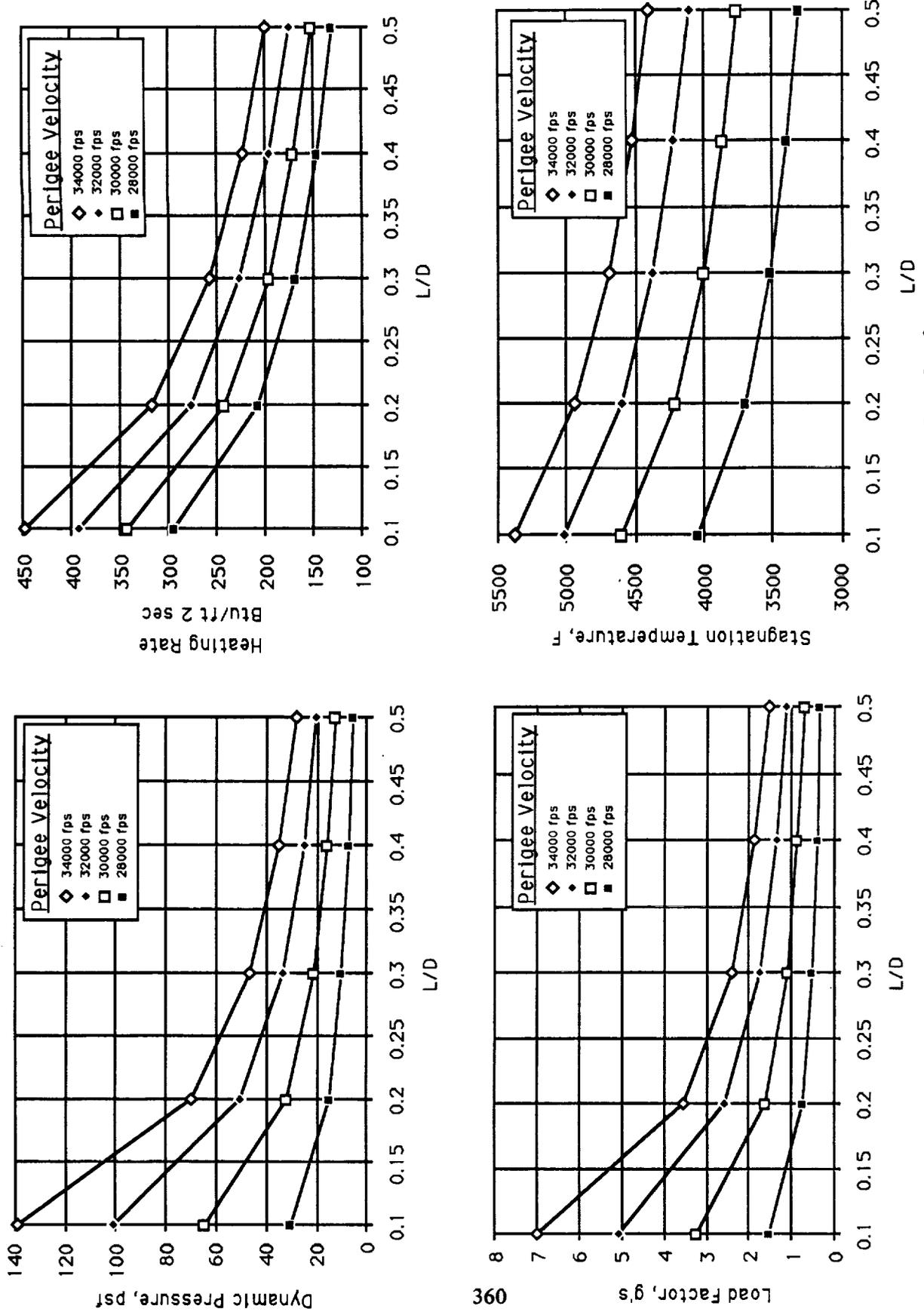
Figure 1-3.1.2-3. Impact of Life-to-Drag Ratio on Aerobrake

BOEING

factor (a measure of the structural loads), and the heating rates are all reduced with increasing lift to drag (L/D) from the aerobrake.

Along with this decrease in the severity of the reentry environment, the vehicle L/D increases as does the width of the reentry corridor that the vehicle can fly (Figure 1-3.1.2-4), giving the vehicle more room for error in the guidance, navigation, and control (GN&C) system.

Configuration Options. As shown in Figure 1-3.1.2-5, the choice of vehicle reentry L/D dictates to a large extent the aerobrake configuration and complexity. On the low end of the L/D scale is the ballute (L/D ≈ 0). Although the least complex operationally with the inflatable decelerator, this configuration has the highest heating and exposes the crew to the highest gravity loads.

The symmetric brake is next on the L/D scale (L/D ≈ 0.15). It is in this region that the reentry corridor is starting to widen enough to provide a sufficient margin of error and the loads on the brake have dropped significantly. This configuration is operationally more complex than the ballute. This operational complexity arises from the fact that although the frontal area is approximately the same size as the ballute, it has an internal rigid frame that is to large to be placed in orbit intact. This complexity is mitigated somewhat by the fact that the aerobrake symmetry makes manufacturing and refurbishing simpler by limiting the number of complex curves.

Moving further up the L/D scale brings the shaped brake (L/D ≈ 0.3). This configuration has a wider reentry corridor and lower loadings than either the ballute or the symmetric brake shapes. However, this configuration shares the requirement for space assembly with the symmetry brake. Along with this space assembly requirement goes an increase in the complexity of manufacture and refurbishment because this vehicle has a large number of complex curves.

Towards the top end of the L/D scale (L/D ≈ 0.8) is the biconic configuration. This configuration, as was mentioned earlier, is severely volume limited in the space-based case because of the requirement to return the entire vehicle to the Space Station.

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Figure 1-3.1.2-4. Reentry Corridor Widths
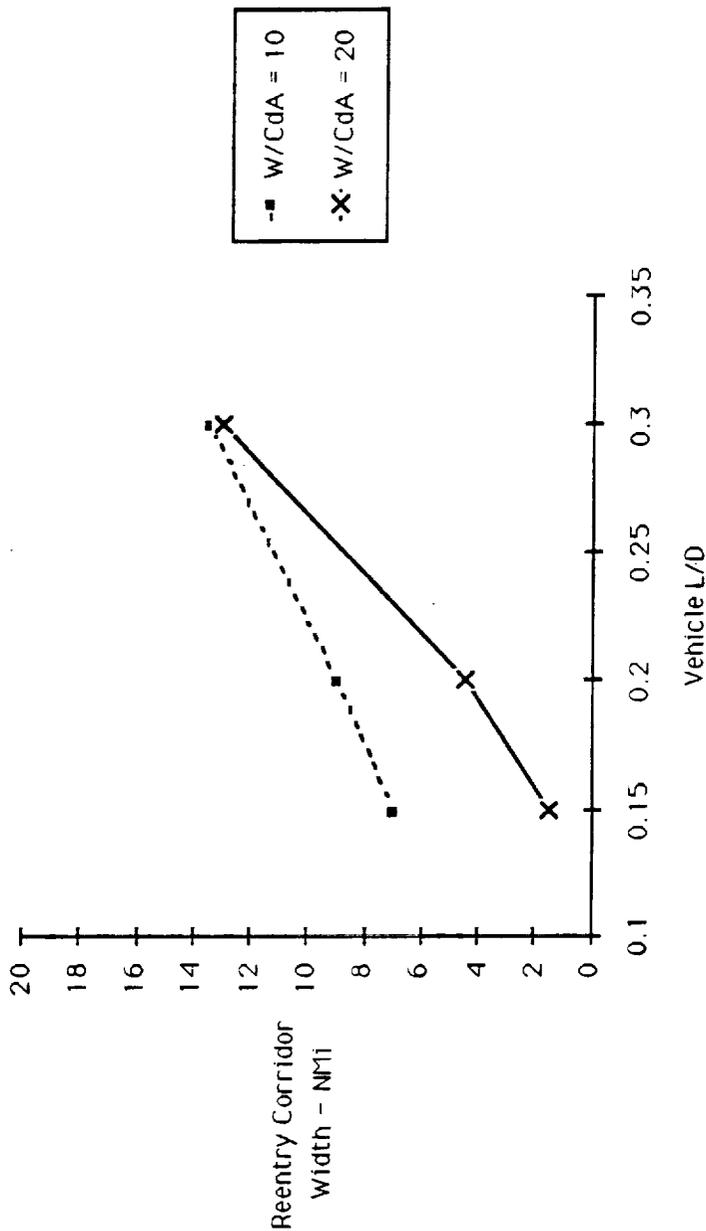
Note: The reentry corridor is defined as the distance between the minimum transfer orbit perigee and the maximum transfer orbit perigee which result in required velocity corrections less than 10 percent of the minimum required velocity correction for the particular vehicle configuration.
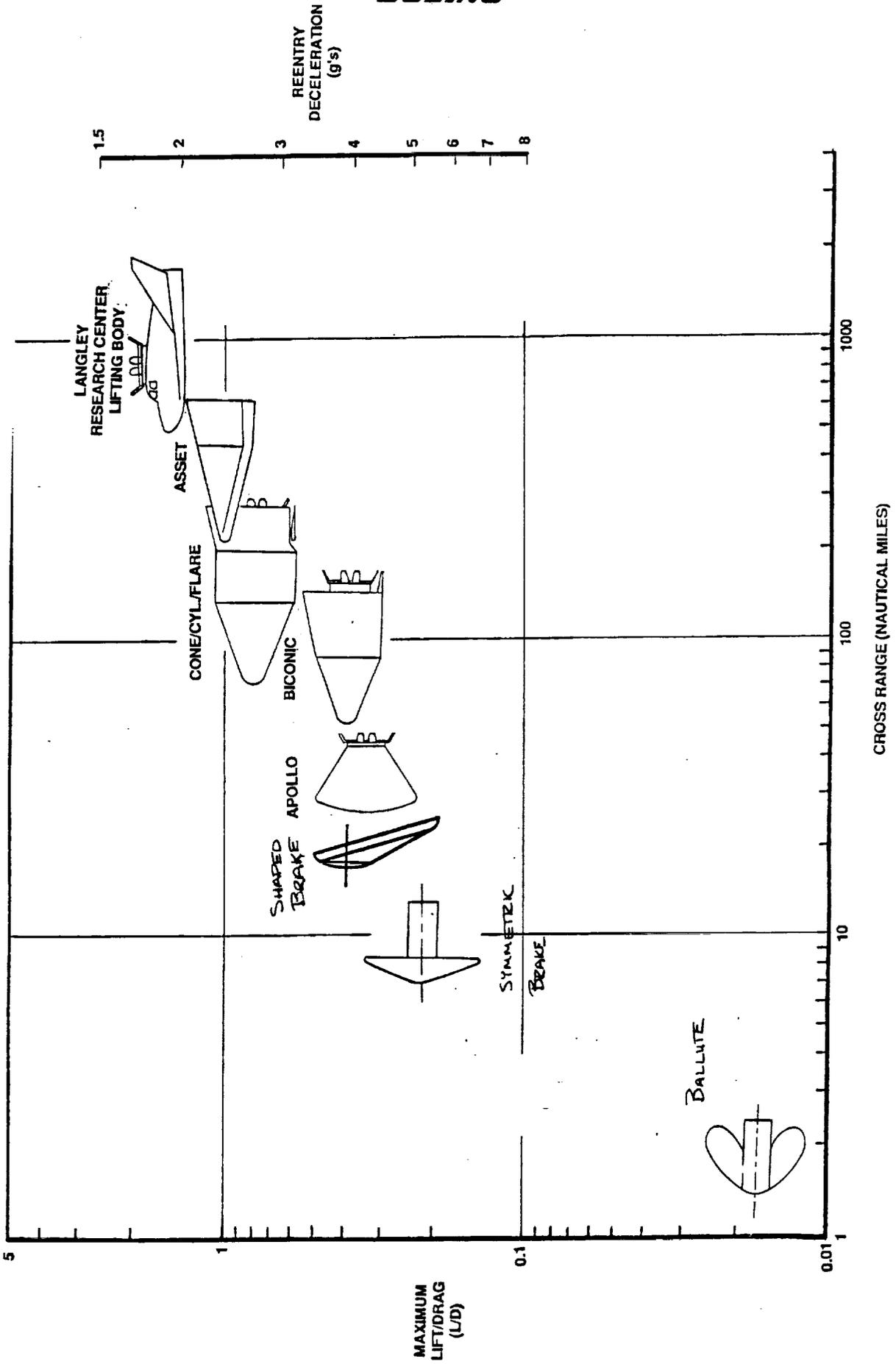
Figure 1-3.1.2-5. Aeromaneuver Configurations

Structural Concepts. The configuration chosen is very dependent on basic structural configuration used in the vehicle. As shown in Figure 1-3.1.2-6, there are very few materials that can be used in the high-temperature environment that the aerobrake will experience. These materials have different structural schemes in which they would be used depending on their mechanical properties.

Two structural schemes were examined in this study to determine if they could meet the aerobrake needs. The first scheme is a "cold" structure where, like the space shuttle, insulative tiles are overlaid on a conventional structure (which is aluminum in the case of the orbiter). In this concept, the tiles take the thermal load and the underlying structure takes any of the aerodynamic loads. As has been proven, this system is very effective. Difficulties also arose in maintaining the bond between the thermal tiles and the underlying structure. This difficulty in bonding the two materials (with their dissimilar expansion properties) has lead to the system used on the orbiter. This system uses a strain isolation pad (SIP) to eliminate the thermal expansion mismatch between the tiles and the aluminum surface.

The second scheme addressed was "hot" structure, where a high-temperature material such as carbon-carbon or a refractory metal (such as columbium or Rene 41) is used as the outer skin of the vehicle. In this concept, the aerodynamic loads and the thermal loads are both handled by the same structure. Although this system appears to reduce the difficulty of attaching the thermal protection system (TPS), this method introduces the problem of high-temperature fasteners and secondary heating of the vehicle behind the aerobrake. Although the vehicle is protected from the thermal environment at the face of the aerobrake, it is now subjected to heating from the backface of the aerobrake.

Thermal Protection Options. If the vehicle uses "cold" structure, the question of the TPS that will overlay the structure must be addressed. In the previous lunar return vehicle (i.e., Apollo), the structural temperatures were maintained using an ablative TPS. In an effort to keep the system reusability as high as possible, reusable thermal protection methods were baselined into the concepts, if possible.

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Material	Density (lb/ft3)	Max Temp °F	Issues
Advanced Carbon-Carbon	101	3200	Oxidation protective coating damage High temperature fasteners and insulation required
Advanced Space Shuttle Tile (AETB)	12	2750	Small, bonded tiles, Durability and repair issues SIP required
Silicon Carbide/Silicon Carbide Composites	156	<3500	High temperature fasteners required
Boeing Zirconia/Hafnia Layered Fibrous Ceramics	16-30	>3500	Testing required Certification required
Flexible Insulation Blanket (TABI)	8	2600	Potential heat shorts at attachments Dynamic flutter
Black Glas Ceramic Matrix Composite (Allied Signal)	137	<3500	High temperature fasteners required Large components must be demonstrated Design allowables require developmental material

Figure 1-3.1.2-6. High-Temperature Material Shopping List

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The current experience with reusable TPS is the tiles used on the space shuttle orbiter. It was this TPS that was initially looked at as the aerobrake TPS. As previously stated, in the case of the space shuttle, there is a mismatch in the coefficients of thermal expansion (CTE) between the tile insulation and the underlying aluminum skin. In the case of the orbiter, the problem is solved with the use of an SIP that allows for differential expansion between the two materials.

This problem can be alleviated if the underlying structure's CTE is more closely matched with that of the tiles. This can be done with a graphite composite (such as graphite polyimide) whose expansion characteristics can be tailored to a desired level.

Because of the difficulty associated with matching the expansion rates between the ceramic tiles and the underlying structure, the tiles themselves are bonded onto the SIP, which is in turn bonded to the outer mold line of the orbiter using room temperature vulcanizing (RTV) adhesives. This is a labor intensive and time-consuming task.

In an effort to reduce that operational complexity, this study looked at a material currently under development at The Boeing Company. This material is a fiberous ceramic composed in a large part of zirconia. This zirconia tile shows great promise of being capable of sustaining temperatures up to 1,975°C (3,500°F).

In addition to this high-temperature capability, the zirconia ceramic has the potential to be mechanically attached to the underlying structure. The current shuttle tiles are processed at extremely high temperatures to sinter the silica fibers in the tile. In the zirconia tile, the temperature that the material is processed at is considerably lower. It is low enough that a honeycomb of a high-temperature metal (such as titanium) can be cast in the ceramic. Following processing of the tile, a facesheet can be attached to the honeycomb and the entire assembly can then be mechanically attached to the aerobrake structure (section 3.1.2, Aeroshell Configuration).

Launch Vehicle Integration. In all the configurations looked at, the aerobrake was >12m in diameter. At this size it is obvious that the intact brake was significantly larger than the launch shrouds available for use. The configuration chosen for the aerobrake uses a rigid structural concept as opposed to a flexible, deployable fabric brake. This rigid concept was chosen because the flexible brake has increased complexity and no weight advantage.

With the choice of the rigid aerobrake, launch vehicle integration becomes a driver in vehicle design. As shown in Figures 1-3.1.2-7 through 1-3.1.2-10, the aerobrake must be packaged differently for the different launch vehicles. Remembering that each seam or penetration must be gas tight before reentry begins, the reduction of these penetrations is critical in the design of the brake. Also important to reducing gas leaks is the reduction of breaks in the TPS in the aerodynamic flow direction. By running the seams across the break as shown in the figures, these flow paths are reduced or eliminated.

It is obvious that the 45-foot launch shroud on the ALS has the minimum impact on the design of the aerobrake. This configuration, however, is a launch shroud of ≈40 metric tons. This mass penalty was considered too severe to be used on the space-based configuration. The option that provided the least configuration impact on launch mass and a lowered on-orbit assembly requirement was the 30-foot ALS shroud.

Chosen Concept. The configuration chosen for the space-based aerobrake is a 15.2m (50-foot) symmetric aerobrake that will fly at a L/D = 0.2. Using the symmetric configuration and flying the trajectory shown in Figure 1-3.1.2-7, the temperatures experienced for a symmetric aerobrake are shown in Figures 1-3.1.2-8 and 1-3.1.2-9. Using this information, it is obvious that the temperatures experienced on the Earth return are going to require some advances in material technology along the lines of the Boeing-developed zirconia tile.

The choice of the L/D = 0.2 was determined by the desire to minimize the manufacturing and refurbishment complexities and use a symmetric aerobrake while at the same time reducing the severity of the reentry environment as much as possible.

D180-32040-2

4.5 m (15 ft) bay width

View A-A 3rd Launch

View A-A 1st Launch

View A-A 2nd Launch

View B-B 3rd Launch

View B-B 1st Launch

View B-B 2nd Launch
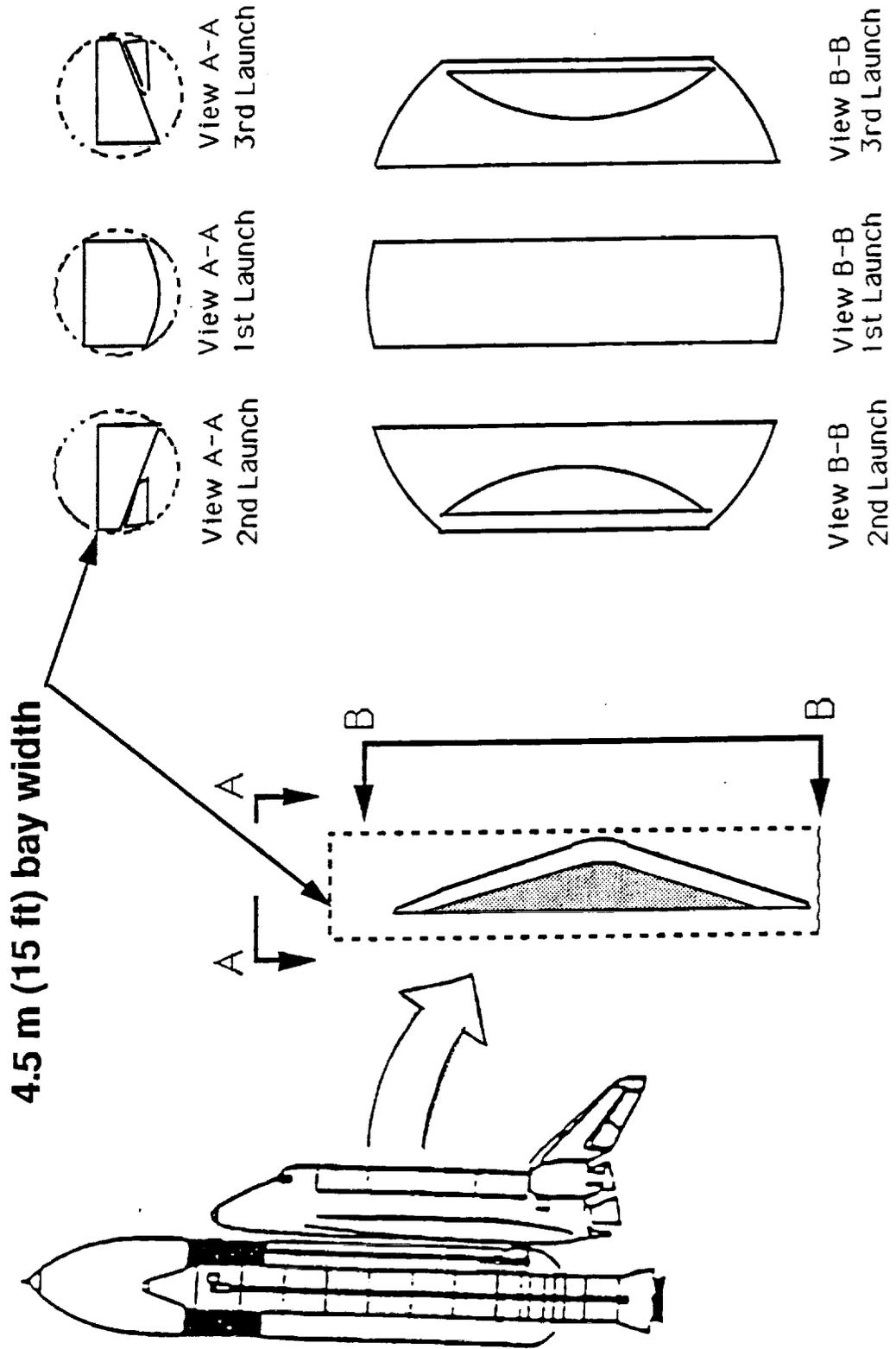
Figure 1-3.1.2-7. Aerobrake Packaged for STS Launch
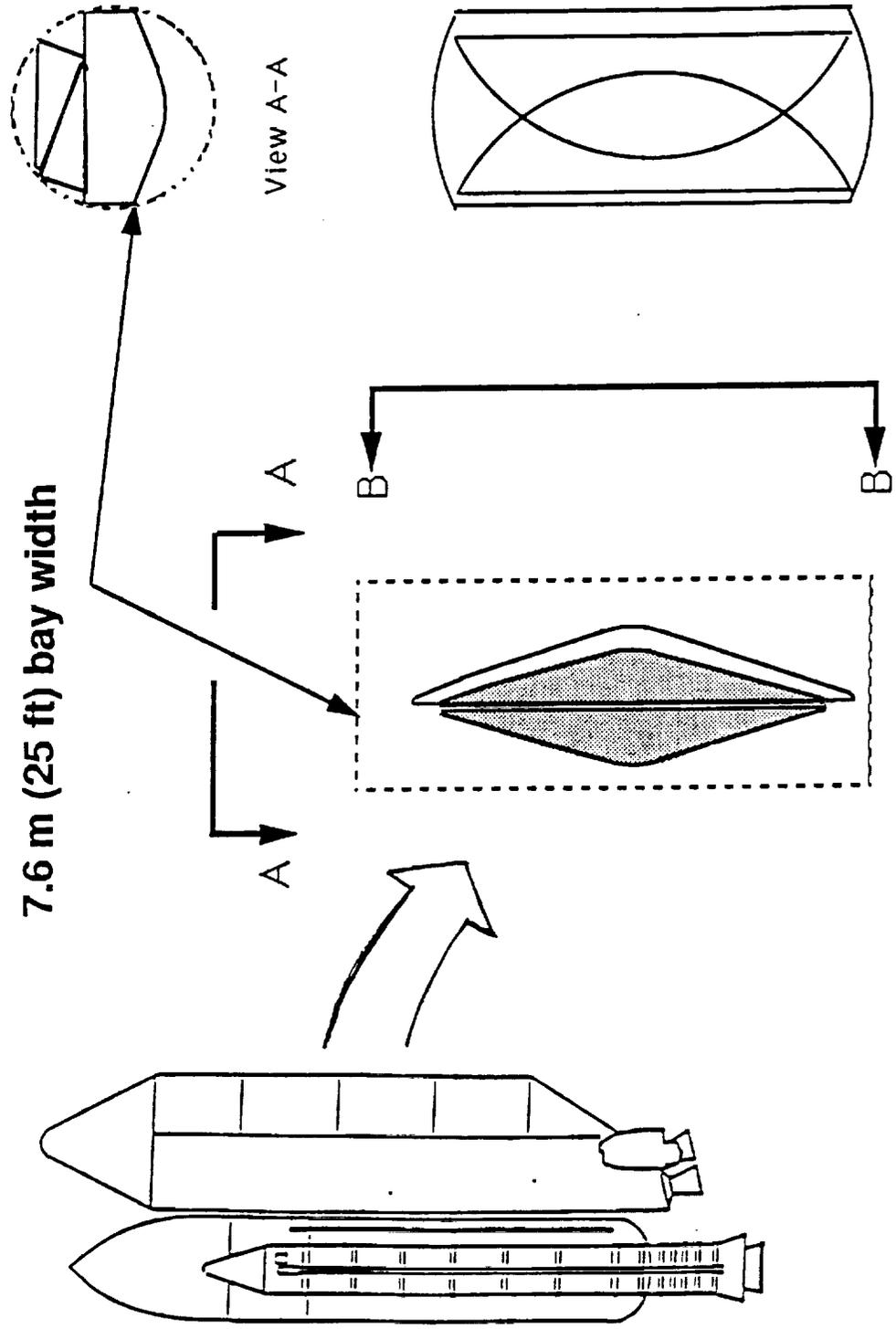
7.6 m (25 ft) bay width

View A-A

View B-B

Figure 1-3.1.2-8. Aerobrake Packaged for STS-C Launch

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30 ft shroud Diameter

View A-A

View B-B

A

A

B

B
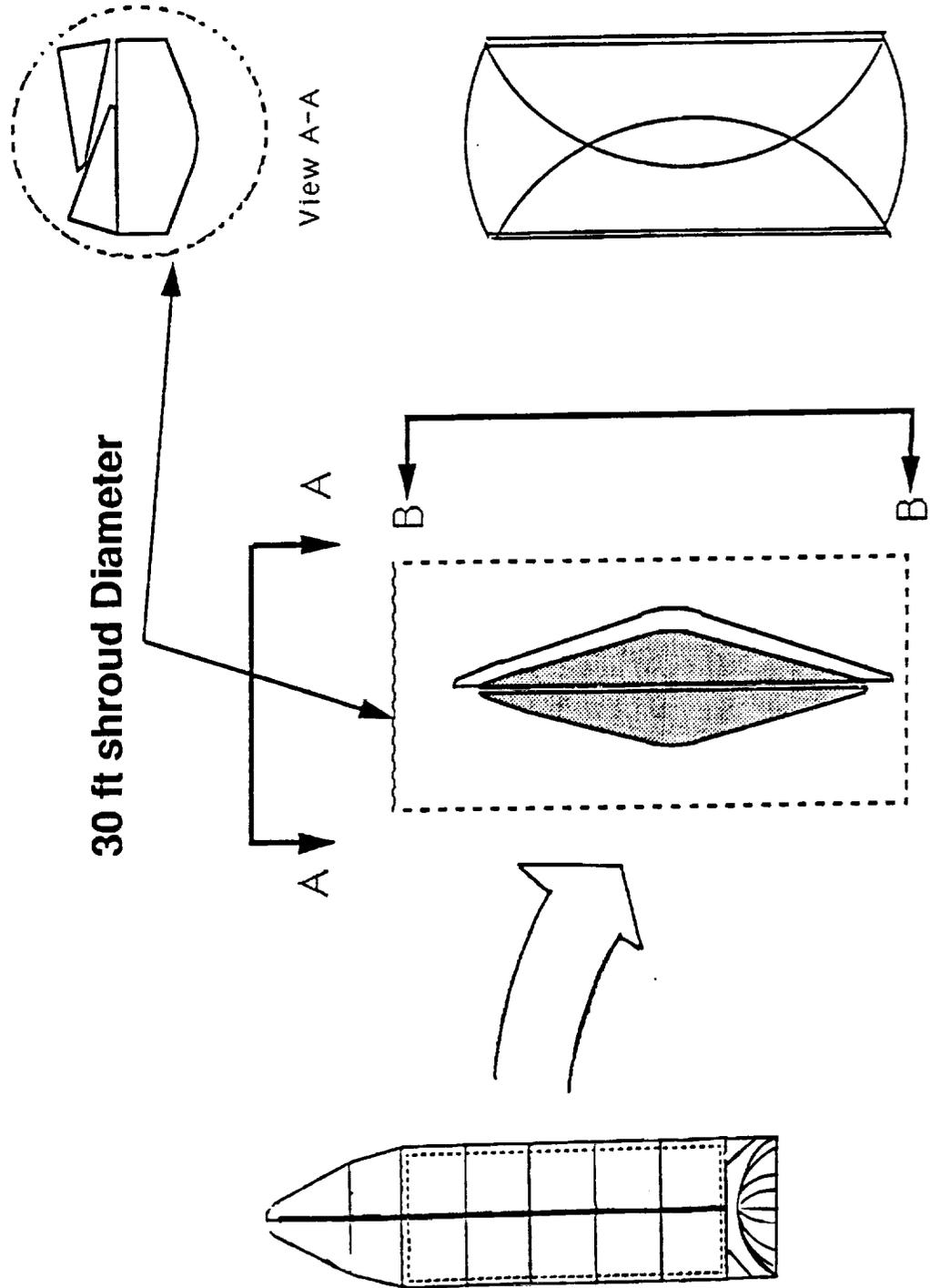
Figure 1-3.1.2-9. Aerobrake Packaged for 30-Foot ALS Shroud

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45 ft. payload shroud

View A-A

View B-B

A

A

B

B
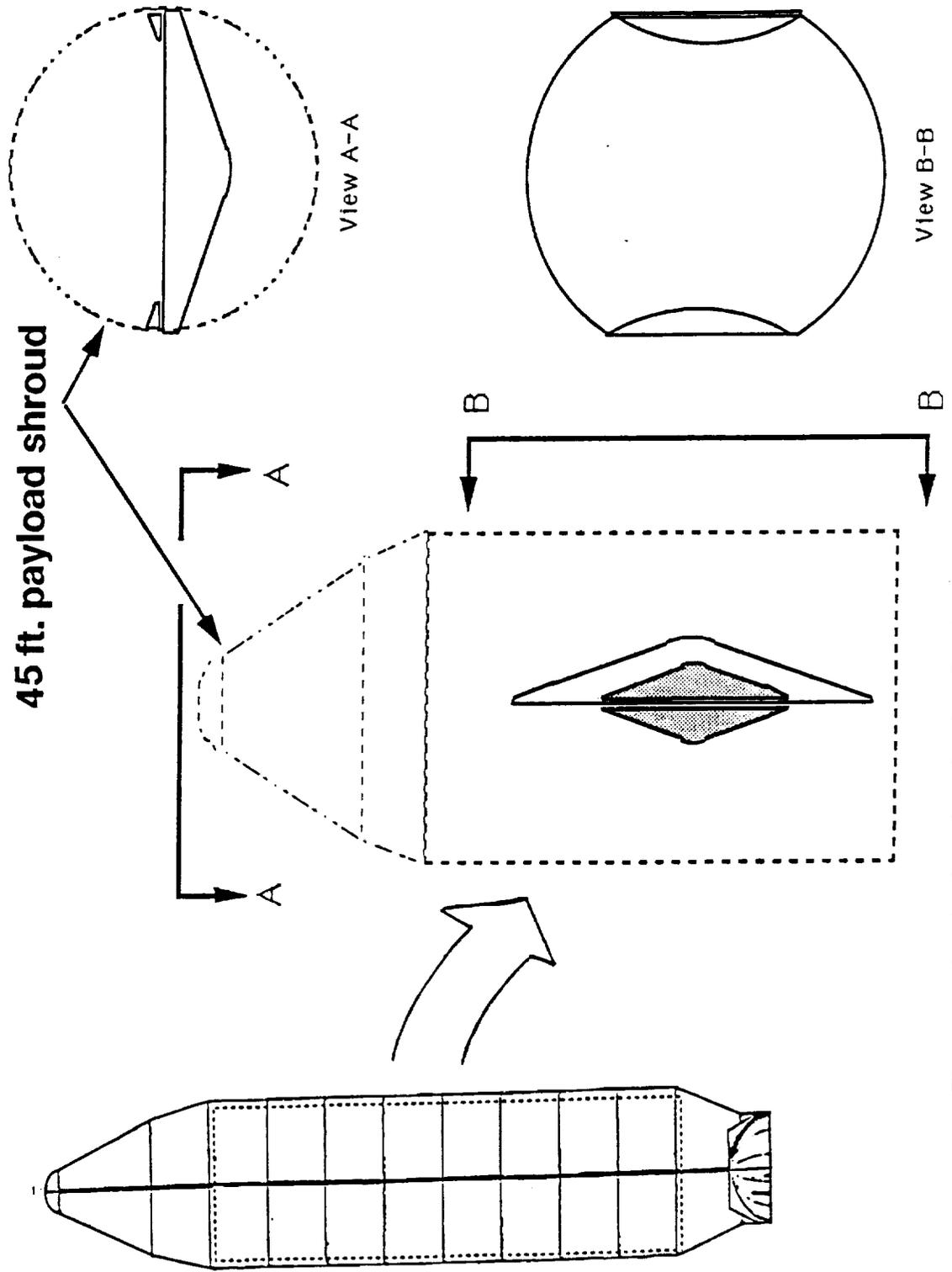
Figure 1-3.1.2-10. Aerobrake Packaged for 45-Foot ALS Shroud

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Aeroshell Configuration. In the ground-based configuration, the vehicle operates in a mode very similar to the familiar Apollo missions. In aeroshell configuration, the majority of the vehicle is expended during the course of the mission with only the crew module being returned to the Earth. As such, the requirements for the aeromaneuver portion of the vehicle are very different from that of the space-based configuration. Although the aerobrake for the space-based vehicle is large enough to make it nearly impossible to launch intact, it is important to remember that the aerobrake is launched only once every 5 years and can take advantage of the assembly capabilities of the Space Station. The ground-based configuration's aeromaneuver configuration does not have this operational flexibility and is, by definition, launched intact.

Because the majority of the vehicle is expended, the ground-based aeromaneuver vehicle does not have the volume constraints on its return cargo as the space-based configuration. This allows the return module to be similar in size to the Apollo command module.

Configuration Options. The options for the return portion of the ground-based STV are shown in Figure 1-3.1.2-1. These configurations cover a number of reentry shapes that have been used successfully many times. It is also important to note the future directions that these configurations will be going. In the case of the lifting body shown, this configuration is being considered in ongoing studies as a PLS vehicle to service the Space Station. Another configuration being considered as a PLS candidate is the biconic shape. This configuration is also part of an ongoing study considering it as an ACRV vehicle.

Selection Criteria. In the ground-return configurations, unlike the space-based vehicle, other issues beside thermal and structural loading drove the overall configuration. Early in the analysis, integration of these modules into the overall configuration, especially during vehicle launch, was a significant discriminator. Because the mass of the return capsule was taken through all the burns of the mission, the mass penalty paid for the aeromaneuver configuration must be minimized. Another important consideration for this return configuration is commonality with other existing systems (such as PLS and ACRV).

Structural Concepts. The configuration chosen is very dependent on basic structural configuration used in the vehicle. As in the aerobrake, there are two overall structural schemes that were looked at in this study. One used "cold" structure where, like the space shuttle, insulative tiles are overlaid on conventional aluminum structure. In this concept, the tiles take the thermal load and the aluminum takes any of the aerodynamic loads. The second scheme addressed was "hot" structure, where a high-temperature material such as carbon-carbon or a refractory metal (such as columbium or Rene 41) is used as the outer skin of the vehicle. In this concept, the aerodynamic loads and the thermal loads are both handled by the same structure.

A third concept that was addressed is more of a hybrid of these two concepts than a a new concept. In this concept the structure is "warm." The outer skin of the vehicle is a high-temperature metal but with a lower maximum temperature capability than the refractories required in the hot structure. This lower temperature capability is possible because the structure is actively cooled.

TPS Options. The TPS options for the ground-return vehicle depend on the structural concept chosen. The hot structure requires the high temperatures of the refractory metals or one of the new high-temperature composites (such as silicon carbide/silicon carbide and carbon/carbon). The materials considered and some of their properties are shown Figure 1-3.1.2-2.

For the cool structure concepts, the material options are somewhat less. Some type of highly insulative ceramic material is required. The currently used shuttle tiles are an example of such a system. This material, however, has several operational problems that greatly add to the complexity of the overall system. The most significant of these problems is the method for attaching the individual tiles to the underlying structure.

Ongoing work at Boeing has developed high-temperature tile, with equal or greater temperature capability than the current tiles. The biggest advantage that this tile has over the current one is the ability to mechanically attach this material to the underlying structure. As shown in Figure 1-3.1.2-3, this material can have a metallic honeycomb cast into the ceramic. After processing, this honeycomb can be brazed onto a metallic facesheet and the the entire

assembly can then be fastened in place on the structure of the vehicle as shown in Figure 1-3.1.2-4.

Active Cooling. Several options exist for the warm structure concept. The major difference between them is the method of cooling the structure. The structure can either be actively cooled with high-temperature heat pipes, with backwall fluid flow, or with transpiration cooling or it can be passively cooled with an ablative material used in previous manned capsules. Only transpiration cooling and backwall cooling were addressed in this study. One concept for this type of cooling is addressed in Figure 1-3.1.2-5. Important to note in this figure is that should the active cooling fail, the dispersant medium functions as a backup TPS provides an additional level of redundancy.

Chosen Configuration. The configuration chosen for study as the ground-return crew vehicle was the biconic shape shown in Figure 1-3.1.2-6. This configuration was chosen for several reasons. First, it offers enough cross-range capability to allow the vehicle to return to one of several specified landing sites throughout the world at any time (Figure 1-3.1.2-7). Second, it places a minimum number of constraints on the overall system because of integration problems. Lastly, it has a great deal of commonality with ongoing ACRV and PLS studies.

One possible trajectory the vehicle would fly on Earth return is shown in Figure 1-3.1.2-8. As shown, the vehicle returns to KSC where it is ground recovered and prepared for the next mission. It also stops at M≈1.0 on this trajectory. It is at this point that the vehicle deploys a drogue chute and starts its ground-recovery sequence as described in section 3.8. The temperatures that the vehicle will experience are shown in Figure 1-3.1.2-9. Overlaid with these temperatures are the temperature capabilities of the various materials assessed in this study. As can be seen, the temperature capabilities of these materials is exceeded at the stagnation point on the nose. It is essential to have some method of cooling this region. The baseline TPS concept is shown in Figure 1-3.1.2-10. Work to this point has not given a clear indication of how the nose should be cooled and the figure demonstrates this uncertainty. Work, however, has been undertaken to determine if active cooling is feasible. Figure 1-3.1.2-11 shows that if the entire vehicle is transpirationally cooled, the amount of water required for the reentry

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Coolant Required(lbs H$_2$O)

Loc	Backwall cooling only	Backwall + Transpiration
1	922	292
2	1723	380
3	2922	375
4	1044	6
total	6610	1053
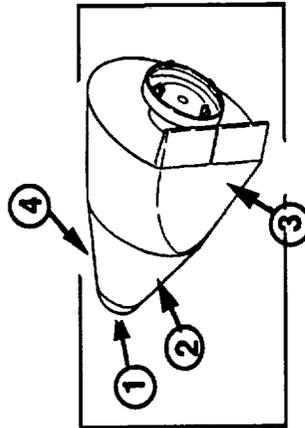
Figure 1-3.1.2-11. Transpiration Coolant Requirements

profile shown is less than the amount of water currently being carried for the crew's radiation shelter.

1-3.2   SUBSYSTEM TRADES

When the STV study contract was expanded to involve the consideration of a large number of mission scenarios and the vehicles that would perform this mission, work began on defining the subsystems that would be required by the vehicle to perform the mission. This work was used to define the differences between the vehicles and determine the impacts of those differences on criteria such as weight, cost, and performance.

Once the vehicle options had been narrowed down to the three chosen configurations (SB, GB, and GO), the subsystem trades and analysis work was to refine and further define the subsystems used in the architecture studies.

1-3.2.1   Propellant Management

One of the groundrules of the STV study was that the vehicle use liquid oxygen and liquid hydrogen as propellants. During the architecture trades, it was obvious that the time the cryogenics needed to be stored was a function of both the configuration and tankset. Because the lift capability of the launch vehicle dictated that the tanks be brought up on several launches (Figure 1-3.2.1-1) different storage requirements for the different tanksets were needed.

The difference in storage times was also apparent in the configurations chosen for further definition in the latter stages of the study. The two vehicles retain the different requirements for the propellant storage. In the space-based configuration (SB), the vehicle is assembled at the Space Station over approximately 6 months. During this period, as propellant tanks from Earth arrive and are attached to the vehicle, the propellant boiloff, by necessity, must be kept to a minimum. After the vehicle has been assembled, the next long-duration storage period for the propellant is when the vehicle is on the lunar surface. It is assumed that after 1 month on the surface the planetary systems will provide support (either refrigeration or reliquefaction facilities) for the remainder of the stay.
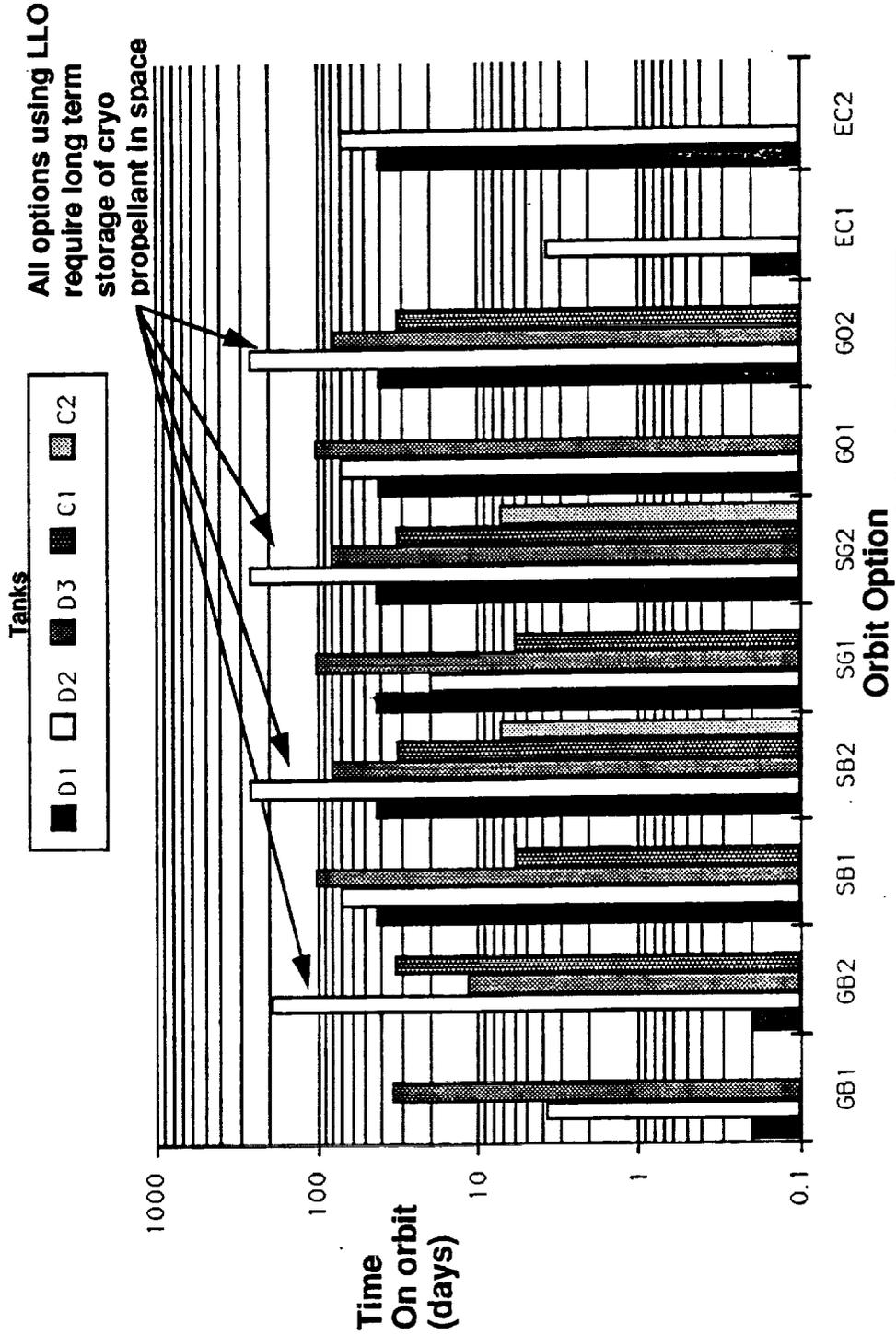
Figure 1-3.2.1-1. Orbit Options

The ground-based vehicles (GB and GO) do not have the long-duration LEO storage penalties of the space-based vehicle because they have very short duration LEO stays. In the case of the GB configuration, the stay in LEO is negligible (≈3 hours) as the vehicle is fully fueled and integrated on the ground.

The GO configuration has a slightly longer duration stay than the GB configuration because it,unlike the GB, is placed in orbit in two launches. The first launch is a tanker loaded with liquid oxygen. This tanker maintains the liquid oxygen for up to 40 days while the second launch is readied. This second launch places the fully integrated vehicle near the tanker. The GO vehicle then rendezvous with its oxygen tanker and offloads the LOX. In this configuration, the liquid hydrogen storage durations approximates that of the GB configuration (it is launched in the second launch), but the liquid oxygen must be stored for the 40 days.

Once the ground-based vehicles leave LEO, the propellant management and cryogenic storage is essentially the same as the space-based vehicle; a 3-day trip time to the Moon and 6 months on the lunar surface (1 month without ground support).

Tankage. Main propellant tanks are designed to permit room temperature proof testing to ensure service life requirements. Typically, LO2 tank membranes vary in thickness from tank bottom to top, sized by a hydrostatic proof test to simulate the variation in tank head pressure. LH2 is much less dense, so LH2 tank membranes generally have a constant thickness from tank bottom to top, sized by pneumostatic proof-test conditions. Maximum room temperature proof pressures are determined as shown in Figure 1-3.2.1-2 by multiplying the maximum expected operating pressure, including head pressures, by a life cycle factor and adjusting for room temperature to cryogenic material property ratios. (For expendable tanks, the life cycle factor is about 1.05. For reusable tanks, the proof factor is increased as a function of expected pressure cycles of the tank).

Both fracture and tensile property ratios were considered for both tank weldments and base metal. The material temperature ratios of 0.952 for -320°F LO2 temperature and 0.909 for -423°F LH2 temperature were based on fracture

Figure 1-3.2.1-2. Cryotank Proof Pressure Sizing
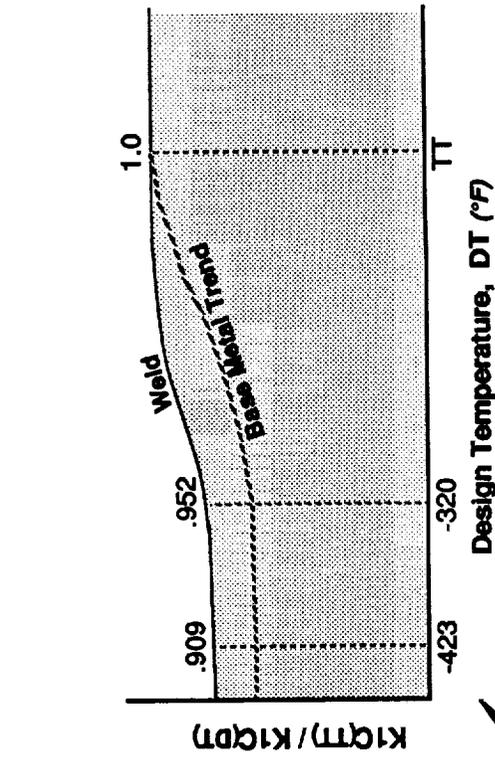
Material Temperature Ratio For
2219 Weldments With 2319 Filler

Produces same flaw tolerance at room
temperature as cryo temperature

$$\text{Proof Pressure} = 1.05 \times \frac{K1C(TT)}{K1C(DT)} \times MEOP$$

Where:

1.05 - Proof Test Factor
MEOP - Maximum Expected Operating Pressure
K1C(TT) - Fracture Toughness at Test Temperature
K1C(DT) - Fracture Toughness at Design Temperature
TT - Test Temperature
DT - Design Temperature

Proof Pressure Include 2 PSI Test Tolerance

- Proof pressure reduces as cryo toughness increases (constant R.T. toughness)
- Proof pressure sizes wall thickness (R.T. conditions)

Example Lox Tank Pressure

properties of the tank weldments. A comparison of typical material properties for 2090-T81 aluminum lithium, WL049-T8 Weldalite, and 2219-T87 aluminum rolled plate are given in Figure 1-3.2.1-3. WL049-T8 has the highest ultimate strength but the lowest fracture toughness. 2090-T81 has high ultimate strength, good fracture toughness, and low material density and is currently the baseline tank material pending further analysis.

Translated to mass ratios in Figure 1-3.2.1-4, 2090 aluminum lithium appears to be best for both minimum gage applications and thick tank wall applications (such as booster tank walls). For thin-gage tank walls in the range of STV tankage, more test data are needed to compare these materials. Most of the test data to date have been for application in thicker gage tanks. Assuming similar properties, aluminum lithium appears to have the greatest overall mass benefit.

Figure 1-3.2.1-5 is a sample mass breakdown comparison of an LH2 tank from the space-based TLI droptank module (mass in kilograms). Overall, the 2219 aluminum tank is 25% heavier than the 2090 aluminum lithium tank. The mass ratio of tanks with a larger number of stringers and frames would tend to be less than 25%.

The 25% difference in tank mass due to material differences has minimal impact on the total vehicle mass. Considering the total inert mass breakdown of the space-based STV shown in Figure 1-3.2.1-6, it can be seen that the propellant tanks are only a small percentage of the overall mass. These percentages take into account the equivalent mass of the TLI and descent droptanks (dropped during the mission) in relation to the core vehicle mass (taken all the way). A 25% increase in tank material mass translates to only a 1.7% increase in equivalent vehicle inert mass, resulting in a total vehicle mass increase (e.g., extra propellant) of only 1%.

Insulation. Both the space-based and ground-based vehicles must handle the cryogenics in two environments: on the ground and in orbit. The ground environment must be considered because on all three of the configurations have tanks that are launched full from the ground. Because of this requirement for thermal isolation in two distinctly different environments, the major trades centered on the type of multilayer insulation (MLI) to use and whether to use a

	Ultimate Strength (ksi)		Fracture Toughness (K1c)		K1c (DT)⁄K1c (RT)	Density lb / in3
	RT	-453°	RT	-453°		
2090 - T81	83	113	33	47	1.42	0.093
WL049 - T8	92	127	22	24	1.09	0.098
2219 - T87	68	102	26	40	1.54	0.103

Typical properties - average of data from NIST tests of production rolled plate material

Figure 1-3.2.1-3. Material Properties Comparison

	Membrane Weight Ratio		
	Minimum Gage Tankage	Thicker Gage made from plate mat'l stock: t > 0.25 in.	Thin Gage made from sheet mat'l stock: t < 0.25 in.
2090 - T81	1.00	1.00	TBD
WL049 - T8	1.05	1.014	TBD
2219 - T87	1.11	1.35	TBD

Figure 1-3.2.1-4. Cryotank Membrane Ratios

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BOEING

LH2 TANK	2090-T81	WL049-T8	2219-T87	REMARKS
	599	**606**	**749**	
FWD DOME MEMBRANE, WELDS	137	139	181	
FWD RING	39	39	39	2219 Design
CYLINDER MEMBRANE, WELDS	185	188	245	
AFT RING	48	48	48	2219 Design
AFT DOME MEMBRANE, WELDS	137	139	181	
SLOSH RING BAFFLE	12	13	13	Density Ratio
VORTEX BAFFLE, SCREEN	9	9	10	Density Ratio
TANK SUPPORT STRUTS, FTGS	32	32	32	
Mass Ratio	1.00	1.01	1.25	

Figure 1-3.2.1-5. Typical Cryotank Comparison

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SB-1.5 Inert Mass Breakdown

- Based on equivalent inert weights (Core=1.0, Descent tanks= 0.45, TLI tanks = 0.16)

Legend:
- ▨ Structure and Mechanisms
- ▨ Tankage - Main
- ▨ Protection
- ▨ Propulsion
- ▥ Power
- ■ Avionics
- ▨ ECLSS, Personnel Prov.
- □ Weight Growth Margin
- ■ Crew, Consumables
- ▨ Non-Cargo Items - Residuals

Percentages: 28.58%, 6.77%, 13.67%, 13.00%, 4.95%, 5.54%, 6.41%, 11.84%, 5.01%, 4.23%
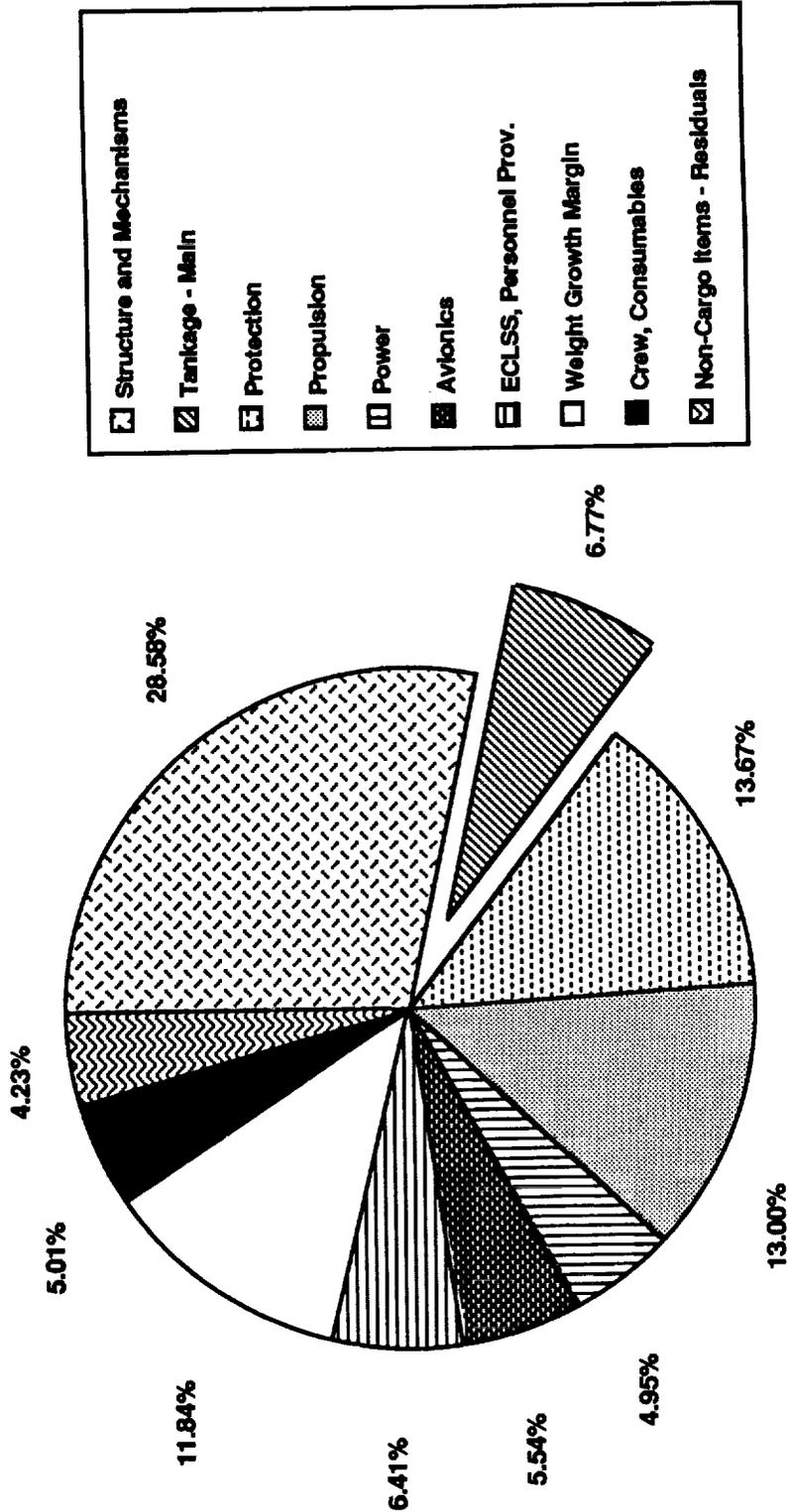
A 25% increase in tankage mass would result in only a 1.7% increase in vehicle equivalent inert mass.

Figure 1-3.2.1-6. Mass Impact of Tank Material on Vehicle Sizing

The Boeing Company logo

foam and MLI combination for tanks that are launched full of propellant. Double aluminized Kapton (DAK) was chosen as the radiation shield material for two reasons: (1) Kapton meets NASA flammability requirements, whereas alternative materials like Mylar do not, and (2) aluminized shields have thermal properties comparable to goldized shields, but they are less expensive. For the core vehicle tanks, we selected 2 inches of DAK as the baseline insulation. However, we did not optimize this thickness. The optimum thickness would be chosen to maximize payload delivery to the lunar surface by trading insulation mass and boiloff losses Optimization thickness would depend on the propellant, the duration of storage, and the tank geometry. For example, Figure 1-3.2.1-1 shows percent boiloff losses as a function of insulation thickness for a 6 month lunar stay for an early vehicle configuration.

For the droptanks, we selected foam and MLI over plain MLI based in part on the results of an earlier study that indicated that foam and MLI generally resulted in more payload to orbit for low-thrust orbit transfer vehicles. The better performance is due to the fact that a layer of closed-cell foam on the outside of the tank provides enough insulating capability to raise the outer temperature of the foam high enough so that nitrogen purge gas can be used rather than helium. The nitrogen gas has a lower thermal conductivity and results in less overall boiloff during the ground hold phase. Using nitrogen also improves commonality with other purge systems. This trade is very sensitive to both ground hold time and storage time in LEO, and thus may have to be revisited when the vehicle design is more refined.

Tank Pressure. The propellant boiloff created when heat leaks into the propellant system creates two interrelated problems in the tank. The first problem is the increased pressure buildup in the tank from the vaporized propellant. To control this pressure the options we considered were refrigeration and venting with a thermodynamic vent system (TVS). Refrigeration has the advantage that it results in no boiloff of the propellant, thus reducing tank size and propellant mass. However, refrigeration of cryogenic propellants requires substantial amounts of power, especially at the low efficiencies inherent in refrigerating at cryogenic temperatures. The net result is that the total system mass, which includes the mass of the propellant, tanks,

refrigerator, and power system is greater for storage times less than about 1 to 2 years.

Propellant Gauging. For the space-based scenario, it is critical that the amount of propellant in each of the various tanks be known with certainty before a mission phase is started. For example, after propellant is transferred into the core vehicle while in LEO, we must be able to verify that enough propellant was transferred into the core vehicle to later enable the vehicle to perform the ascent and TEI burns. We looked at two options: (1) a pressure-volume-temperature (PVT) gauge currently under development and (2) distributed point sensors. Distributed point sensors require that all the liquid be settled at one end of the tank and that the liquid-vapor interface be relatively flat. This may not be the case in low gravity, so a settling thrust would have to be applied to get an accurate reading of the propellant quantity using point sensors. The PVT gauge will, in principle, give an accurate reading of the amount of propellant regardless of the liquid orientation, and thus will save the propellant associated with settling thrusts.

Liquid Acquisition. We considered using a start basket to trap liquid propellant at the outlet and ensure that subcooled liquid is provided to the engines at engine start. However, Pratt & Whitney has already demonstrated in prototype testing that their engines can run in "tankhead idle" mode. In this mode the engines can take vapor, liquid, or a two-phase mixture and operate at a low thrust sufficient to settle the remaining liquid over the tank outlet. Thus, we decided that a start basket would add extra weight and was not necessary.

We also considered using a screened-channel liquid acquisition device (LAD) to withdraw liquid from the drop tanks into the core vehicle tanks during the propellant transfer operation in LEO. However, it turned out that the weight was prohibitive and that it was more efficient to transfer during the TLI burn while the propellant was settled by the vehicle acceleration.

1-3.2.2   Propulsion

Reference Vehicle for Propulsion Analyses. A two-stage vehicle was identified by the 90-day study as appropriate for the lunar phase of the Human

Exploration program. The LTV provides propulsion for transfer from low Earth orbit to lunar orbit and return. The LEV provides transportation for crew and cargo from lunar orbit to the lunar surface and return to lunar orbit. These vehicles were selected as a convenient reference for trade studies to define propulsion system characteristics needed to accomplish Human Exploration program objectives. Earth-Moon transportation requirements affecting propulsion for the lunar vehicles include the following:

1. Man rated.
2. Cryogenic (LOX/LH2) propellants.
3. Vehicle may be maintained, mated, and serviced at Space Station Freedom (space based) or return the crew capsule to the Earth (ground based).
4. Reusable and capable of five complete missions without major servicing (space based) or partially reusable (crew capsule) with expendable boost modules (ground based).
5. Capable of rendezvous, docking, and payload and propellant transfer in lunar orbit.

The selected LTV configuration has two sets of propellant droptanks, one of which was dropped after the TLI burn and the second set dropped after the lunar landing. Lunar ascent and TEI are accomplished with the core module, including tankage, and the main propulsion system, which is returned to LEO with the space-based system. With the ground-based configuration, all but the crew capsule is staged and expended; the crew cab reenters and lands on the Earth.

Man Rating. Man rating is discussed in detail by the JSC-23211 document "Guidelines for Man Rating Space Systems." Fundamentally man rating a system incorporates those design features necessary to accommodate human participants. This implies the capability to safely conduct manned operations including safe recovery from any credible emergency situation. Man rating has been interpreted as requiring that all critical systems (such as the propulsion system) must be two failure tolerant. Stated another way, the propulsion system must provide for safe return of the crew to LEO from any part of the lunar mission. An independent crew escape system to return from the lunar surface

does not appear to be a practical solution for early missions because of the large ΔV (more than 3,000 m/s) and vehicle mass requirements.

An "engine workshop" was organized by MSFC to address new engine design criteria issues, which included man rating. The Boeing position on man rating is based on application of the "two failure tolerant" requirement to the propulsion system instead of to the engine as a separate item. This approach relieves the engine from the two failure tolerant requirement for internal components such as turbopumps. Engine designs are simplified by this approach, but redundant engines become necessary for man rating. Any engine failure that may occur must be isolated to a single engine and not propagate to the redundant engines or other critical systems. Isolating failures to a single engine may be accomplished with shielding between engines by initially designing the engines to contain any credible failures or by designing engines to shut down passively prior to the onset of catastrophic failure.

Space Basing. Propulsion system design for space basing requires data defining the effects of space environments (e.g., vacuum, thermal, radiation, lunar dust, and atomic oxygen) on each of the systems components and materials. The availability of the data needed for long-term space exposure of cryogenic components and materials is not known. A comprehensive list of components and materials should be prepared and the adequacy of available data determined as soon as practical. Requirements for additional materials and components tests need to be established and testing initiated to obtain data supporting designs within the STV schedule. These could be incorporated in future LDEF missions and experiments.

Simplified logistics is an important design goal to support space basing. Engines and other systems should eliminate any requirement for special valve actuation, purge, or other purposes.

Operations. Engine-related maintenance and checkout operations at the Space Station will incur crew costs now estimated at $123,000 (1989) per hour for two EVA personnel and one IVA observer. The high costs emphasize the need for highly reliable systems that will require little or no maintenance over the life of the vehicle. The reliability of the functional hardware must be

supported by comprehensive instrumentation to verify the status and confirm that reliability has not been degraded over the life of the vehicle. Redundant instrumentation with additional verification by cross-referencing related measurements will be required to ensure that health of the hardware is correctly diagnosed.

System and Subsystem Interface Architecture. Interfaces between the engines and vehicle systems should be designed to minimize the number of electrical and fluid connections to engine change out if required. All connections should provide verification of integrity without manual inspection. Fluid connections should be designed to the amount of fluid spilled during connection and disconnection.

Health Monitoring. The health monitoring logic diagram, Figure 1-3.2.2-1, shows the flow of data and commands across the engine vehicle interface. The vehicle system is planned to perform the main diagnostic tasks of the complete system including engines. Characteristic response data for all of the engine sensors will be stored by the data system and will include results of all firings subsequent to the acceptance test. These data will be evaluated on a real-time basis to determine if engine operating commands should be modified. Significant outputs of the health management system will be caution and warning type data sent to the flight controls computer for action at the discretion of the pilot. All commands to the engines will be through the flight controls computer. Engine controllers will have direct authority only for solely internal engine commands or particular emergency conditions that will have been negotiated by vehicle and engine company design teams.
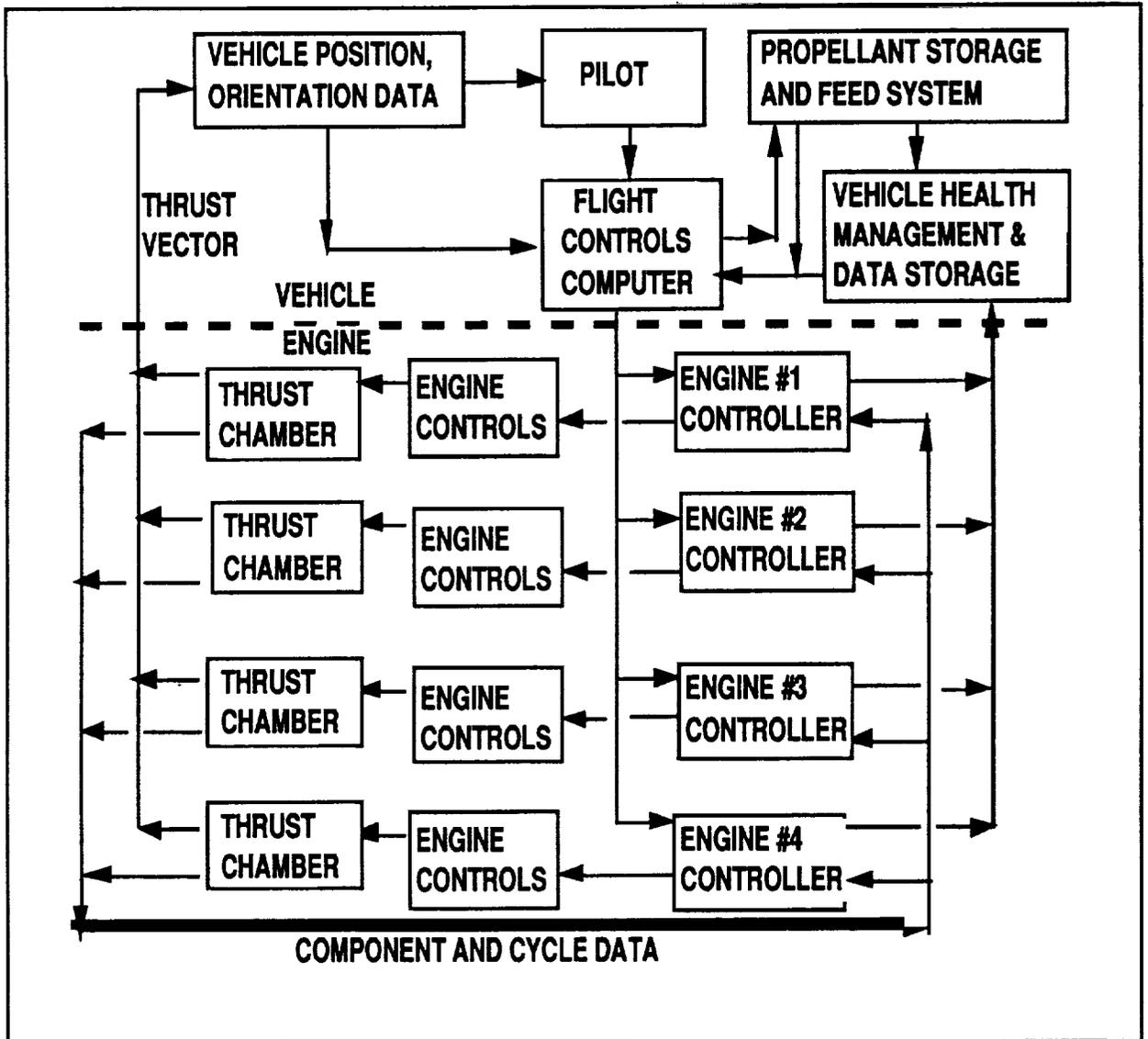
Figure 1-3.2.2-1. Health Monitoring Logic Diagram

Data required for propulsion system health monitoring will depend on actual system and engine designs. A generic set of data, as might be appropriate for expander cycle engines with typical cryogenic propellant storage, is shown in Figure 1-3.2.2-2.

- DATA PROVIDED BY THE VEHICLE

  - PROPELLANTS
    - QUANTITIES REMAINING
    - INTERFACE PRESSURES
    - INTERFACE TEMPERATURES

  - COMMANDS
    - THRUST
    - MIXTURE RATIO

  - ENGINES HISTORICAL RECORD CHARACTERIZATION

- DATA PROVIDED BY THE ENGINE

  - COMPONENTS
    - VIBRATION
    - ROTATIONAL SPEED
    - TEMPERATURES
    - STATUS (VALVES OPEN/CLOSED)

  - THERMODYNAMIC CYCLE
    - MIXTURE RATIO
    - FLOW RATES
    - PRESSURES
    - TEMPERATURES

Figure 1-3.2.2-2. Health Monitoring Required Data

1-3.2.2.1 Pratt & Whitney Subcontract

A subcontract was awarded to Pratt & Whitney to accomplish three tasks for RL10 derivative engines:

1. Task 1 - Reusability, reliability and health monitoring.
2. Task 2 - Space basing and maintainability.
3. Task 3 - Interface requirements and performance.

Results from Task 1 health monitoring show that the RL10 displays graceful failure modes (the current configuration has more than 3,800 tests over 25 years without catastrophic failure) and does not require the level of instrumentation being considered for an advanced booster engines. A schematic for a full throttling configuration of the RL10 is shown in Figure 1-

3.2.2.1-1. Engine malfunctions show up in the chamber pressured (Pc), turbopump speed (rpm) and/or vibrations. Valve positions are required for reference Pc and rpm levels and for available control electric power and/or pneumatic power-level measurements for control capability. The oxidizer control, fuel control, and cavitating venturi are electromechanical control valves that set mixture ratio and thrust and isolate fuel system instability at low thrust levels. The other valves shown are On/Off valves currently pneumatically actuated that might be converted to electromechanical actuation systems. Parameters that should be monitored include the following:

1. Turbopump speed.
2. Valve positions.
3. Vibrations.
4. Bearing acoustic emission.
5. Chamber pressure.
6. Power supply.
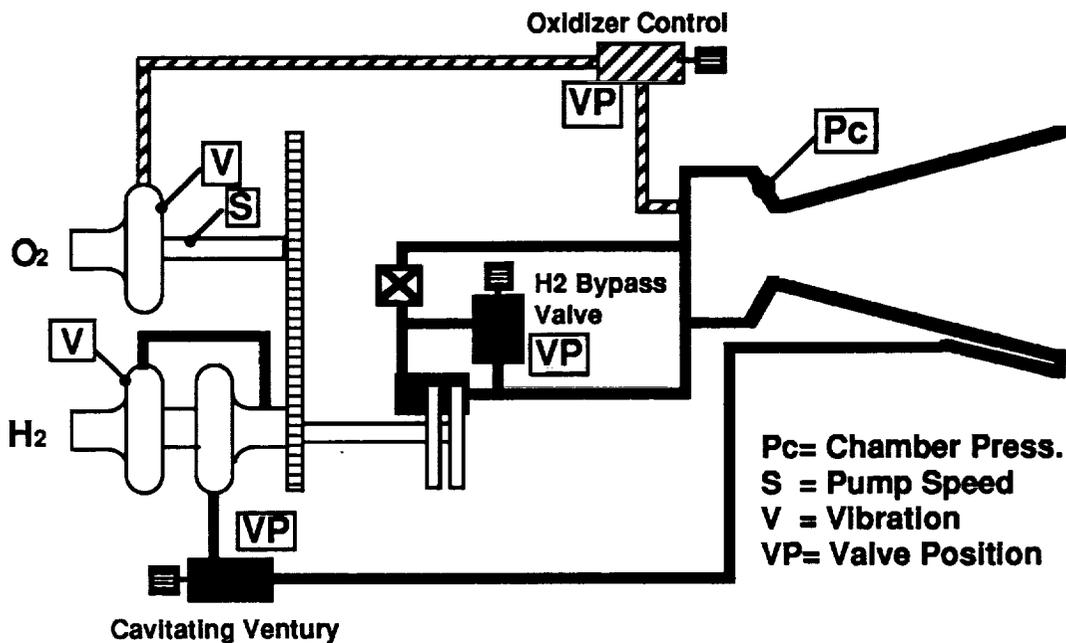
Figure 1-3.2.2.1-1. RL10 Schematic

Figure 1-3.2.2.1-2 lists malfunction modes that have been identified for the RL10 and indicated anticipated effects. Figure 1-3.2.2-1-3 provides quantitative

response characteristics of chamber pressure and rpm to changes in efficiency and possible fluid leaks. These data are essentially influence coefficients that could be used aid identification of deteriorating component performance and potential failures. An example of rpm data that could be used to identify a potential bearing failure is shown by Figure 1-3.2.2.1-4. This speed tailoff characteristic is clearly abnormal although the test run was completed with apparent problems. Posttest inspection revealed the bearing problem and review of the data revealed the unusual data trend.

MALFUNCTION MODES			INDICATED EFFECTS
LEAKS			Change in Pc, RPM
TURBOMACHINERY	- Bearings		Fast Speed Tailoff at Cutoff Vibraton Change in Pc, RPM
	- Seals		Change in Pc, RPM
	- Performance		Change in Pc, RPM
CONTROL VALVES	-Turbine Bypass	- OPEN / - CLOSED	Low Pc, Low RPM / High Pc, High RPM
	-Oxidizer Flow	- OPEN / - CLOSED	High Pc, High RPM / No Pc
	-Cavitating Venturi	- OPEN / - CLOSED	Fuel Instability at Low Thrust / No Pc
ON-OFF VALVES	-LOX Tank Press.		No Effect
	- Fuel Tank Press		No Effect
	- LOX Inlete	-OPEN / -CLOSED	No Effect / No Pc
	- Fuel Inlet	- OPEN / - CLOSED	No Effect / No Pc
	- Fues Shutoff	- OPEN / - CLOSED	Slow Shutdown / No Pc

Figure 1-3.2.2.1-2. RL10 Malfunction Modes

	ΔPc	ΔRPM
-2.6% Fuel Turbopump Efficiency	-2%	-1%
-3.4% Fuel Turbopump Head	+2%	+2%
-11.6% Lox Turbopump Efficiency	-2%	-1%
-3.8% Lox Turbopump Head	-2%	-0.3%
7.7% (2.9 lb/sec) Lox Leak	-2%	-0.7%
2.6% (0.16 lb/sec) Fuel Leak D.S. of Turbine	+2%	-1.6%
2.0% (0.13 lb/sec) Fuel Leak U.S. of Turbine	-2%	-1.1%

Figure 1-3.2.2.1-3. RL10 Malfunction Effects on Pc and Turbopump Speed
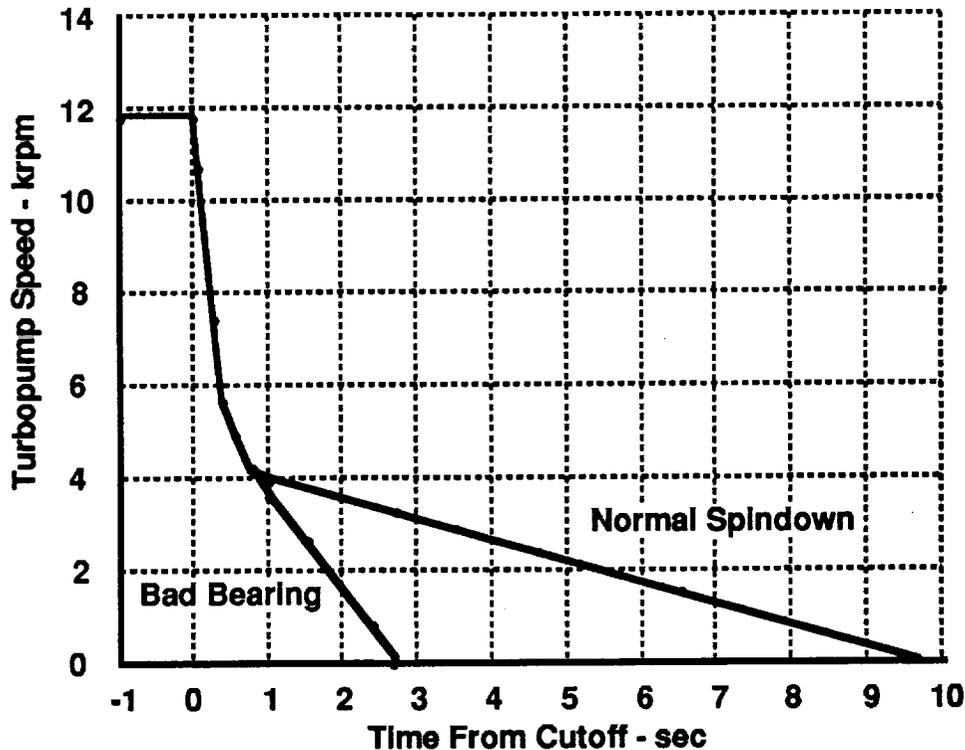
Figure 1-3.2.2.1-4. RL10 Turbopump Speed Tailoff Characteristics

A preliminary instrumentation selection (Figure 1-3.2.2.1-5) has been made to support health monitoring of a space-based STV and minimize maintenance and checkout requirements. Goals of the instrumentation selection were 5 years without calibration, robust design, redundancy, and self-generating or low-power requirement.

Performance. Studies of advanced space engines have been conducted for several years to develop components and determine performance capabilities. As a result of these studies a chamber pressure of 1,500 psia appears achievable with low risk. The benefits of higher chamber pressure than available from the RL10 (400 to 600 psia) are illustrated in Figure 1-3.2.2.1-6. This figure was developed to identify engine constraints related to launch vehicle size. The same trends and benefits of high chamber pressure are applicable to any other area or size constraint. A 1,500-psia chamber pressure has been assumed for all propulsion trades that have been conducted.

BOEING

	Self Generating	Low Power	Temperature Sensitivity	Radiation Tolerant	Vacuum Tolerant	Long-term Stability	Electronics Required	Direct Computer Readout	No. Wires	Accuracy			
Speed													
o Magnetic core	X	None	None	X	X	X	L		2	H	X	?	
Pressure													
o quartz frequency		X	L	X	X	X	H		?	H	?	15 oz	
o Pierozoresistive		M	M	X	X	X	M		?	M	?	15 oz	
o Quartz capacitive		?	M	X	X	M	L		4	L	M	L	
o Bellows/bourdon			?	?	X	X	MH		?	?	?	?	
tube with wire contact encoder		None	L-M	?	X	X	None	X	8+	L-M	MH	?	X
Position													
o Potentiometers		X	L	X	X	X	L		2	L-M	MH	?	X
o Resolver encoder		?	None	X	X	X	M		?	M	X	?	
o Contace encoder		None	None	X	X	X	None	X	8+	M	X		
Vibration	X	X	M	?	X	X	?		?	M	X	?	X
		X	M	?	X	X	H		?	?	?		X

Figure 1-3.2.2.1-5. Instrumentation Selection

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D180-32040-2

EFFECTS OF THRUST AND CHAMBER PRESSURE ON SPECIFIC IMPULSE, NOZZLE AREA LIMITED TO 35% OF 25 FOOT DIAMETER LAUNCH VEHICLE SHROUD
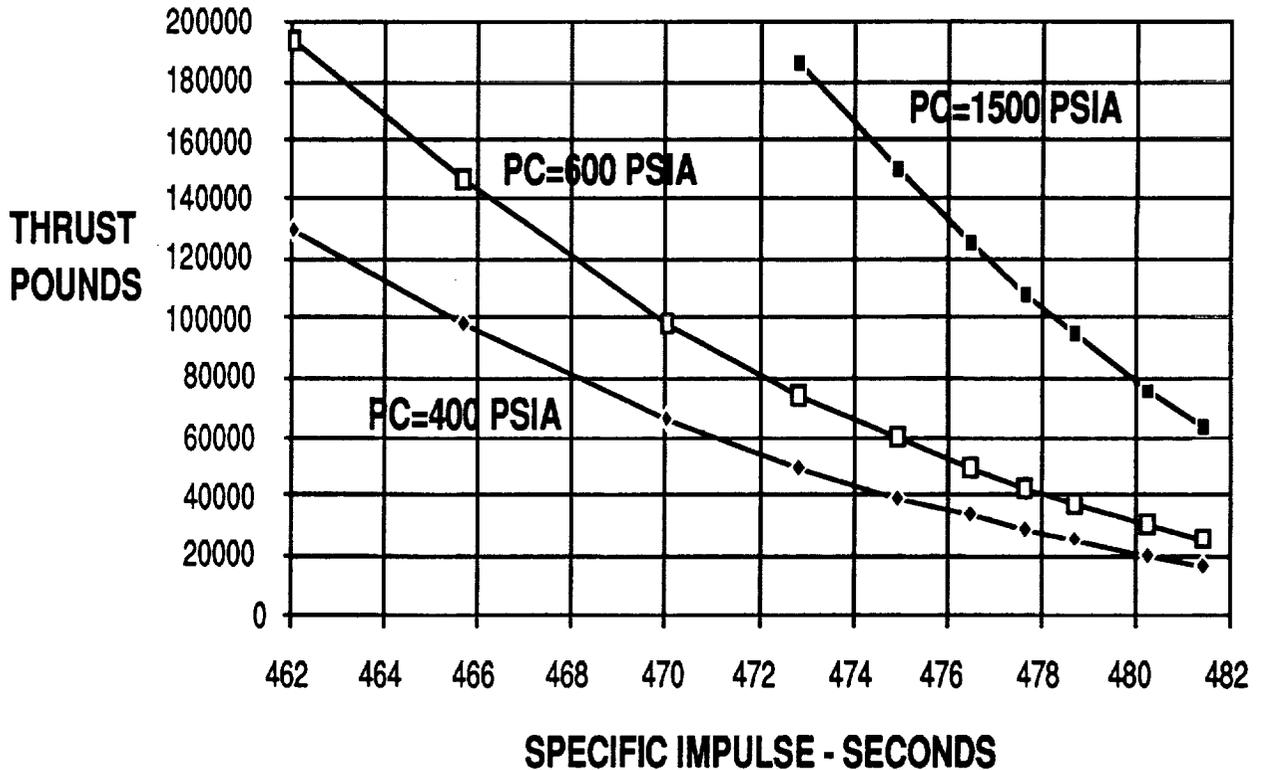
Figure 1-3.2.2.1-6. Thrust, Chamber Pressure, and Area Relationship

1-3.2.2.2 LEV Thrust Requirements for Lunar Landing

Lunar landing is the most critical phase of the lunar mission for the main engines. For this reason lunar landing requirements for engine thrust were taken as the engine sizing requirement. Actual engine thrust levels for landing the four-engine vehicles depends on cargo mass as shown in Figure 1-3.2.2.2-1. Differences in the 90-day study and reference vehicle reflect differences in configuration and inert weight scaling. The curves at 75% hover thrust represent the lowest thrust required based on comparison with Apollo flight experience. If the reference vehicle is ultimately selected for development, an engine design thrust of 66,700N (15,000 lbf) capable of throttling to 20% of design thrust would

provide landing capability with zero cargo. The zero cargo landing capability may be desirable to provide flexibility including rescue missions if required. Shutting down opposing engines will reduce throttling requirements. A 20,000-lbf design thrust engine with throttling to 10% would be adequate for landing the the 90-day study vehicle mass with the required 13 metric tons of cargo mass.
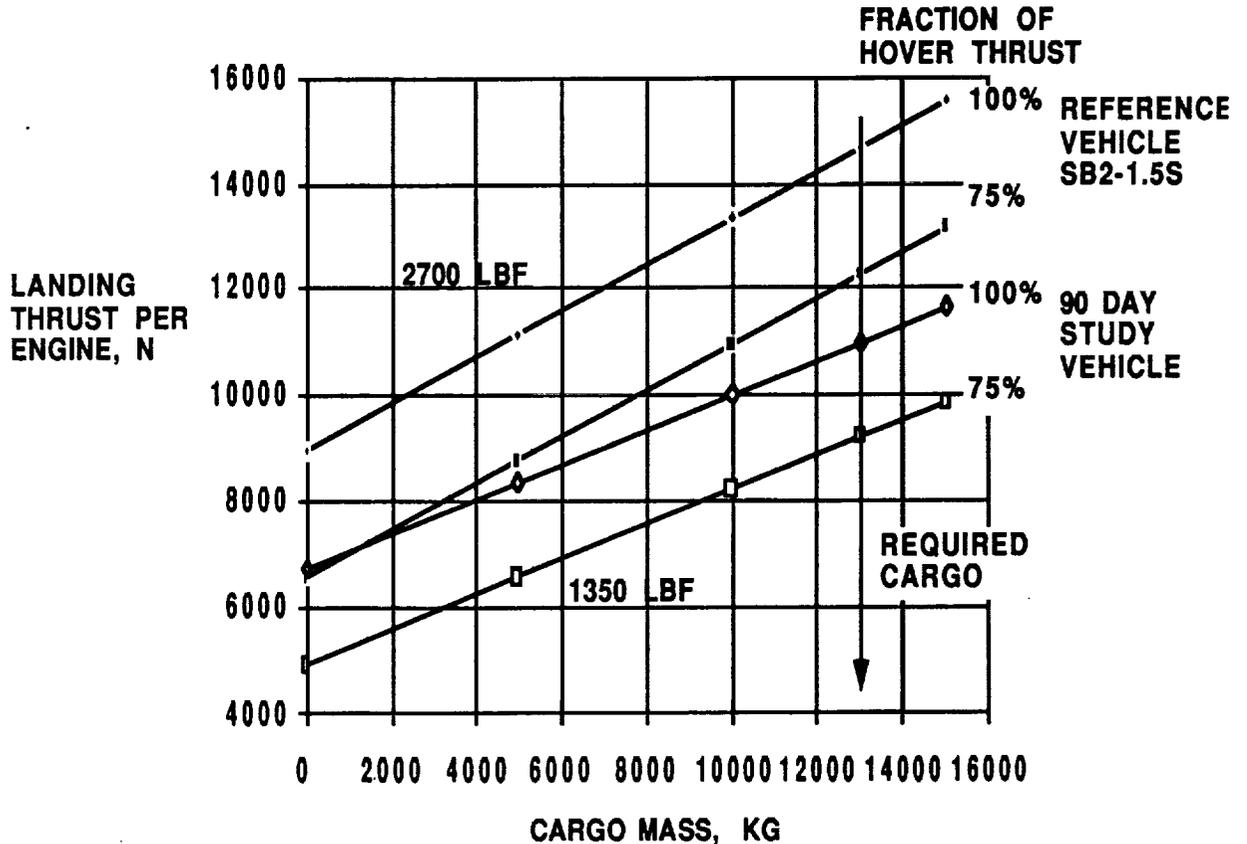
Figure 1-3.2.2.2-1. Lunar Landing Required Thrust

The LEV engines are arranged in a pattern as illustrated in Figure 1-3.2.2.2-2. The vehicle center of mass varies from about 3.0m to 4.9m forward of the engine gimbal plane. The nozzle area ratio of the 15,000-lbf engine determines the exit diameter as shown in the figure. A minimum separation of 15 cm between the nozzles was used with the exit diameter to determine the engine offset from the vehicle axis, and each engine pair provides off setting moments about the center of mass, minimizing high gimbal angles. Cosine losses reduce

the axial thrust and delivered specific impulse as shown if each engine thrust is directed through the center of mass. This operation strategy will reduce these vector losses by aligning each engines thrust vector with the vehicle axis at all times except when operating at low altitudes. Evaluation of this strategy will require detail studies of the vehicle controllability under conditions of a sudden engine failure. The effects engine out control requirements on gimbal power and/or reaction control system sizing must be evaluated during the control study.
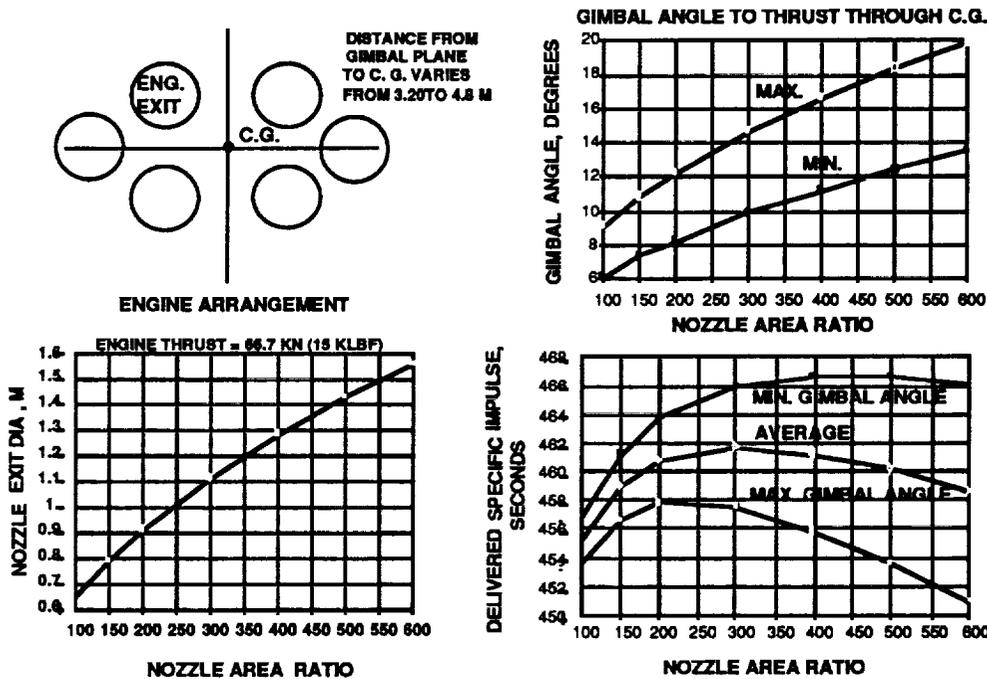
Figure 1-3.2.2.2-2. Gimbal Angle Effects on Specific Impulse

1-3.2.2.3   LEV Number of Engines and Thrust for Abort

The "worst case" abort condition for the LEV is at the end of descent with the vehicle at essentially zero velocity, which is the same as abort from the lunar surface (Figure 1-3.2.2.3-1). Three and four engine vehicles with engine thrusts of 15,000 and 20,000 lbf were evaluated for abort capability by comparing normal ascent propellant requirements with abort propellant required with two engines failed. The abort propellant required with the payload dropped was less than the normal ascent propellant for four engine vehicles with two engines

operating and two failed at either the 15,000- or 20,000-lbf thrust level. No additional reserve propellant was required for abort of the four engine vehicles. Three engine vehicles with two engines failed and only one operating did require additional reserve propellant for abort. Three engine vehicles were eliminated from further consideration because of the requirements for abort propellant reserves.

·NOMIMAL ASCENT LIFT OFF MASS, 16803 KG

·ABORT AT END OF DESCENT, PAYLOAD DROPPED, MASS IS 16430 KG

CONDITION	THRUST	THRUST/WT	ΔVG LOSS	ΔV TOTAL	PROP*
ASCENT					
4 ENG. 20K	80,000	2.164	0.6 M/S	1809 M/S	5501 KG
4 ENG. 15K	60,000	1.623	3	1811	5508
3 ENG. 20K	60,000	1.623	3	1811	5508
3 3NG. 15K	45,000	1.217	1 2	1820	5530
ABORT					
2 ENG. 20K	40,000	1.107	1 8	1826	5421
2 ENG. 15K	30,000	0.83	4 4	1852	5484
1 ENG. 20K	20,000	0.55	111	1919	5643
1 ENG. 15K	15,000	0.41	176	1984	5795
				*ISP=465	

Figure 1-3.2.2.3-1. Abort Performance From the Lunar Surface

1-3.2.2.4 LTV Engine Optimization for TLI Firing

The TLI burns are the main source of gravity losses that affect engine selection for the LTV. Figure 1-3.2.2.4-1 shows the gravity losses used for this study for both one- and two-burn transfer orbit strategies. A one-burn transfer orbit is preferred to minimize the time spent in high-radiation regions.
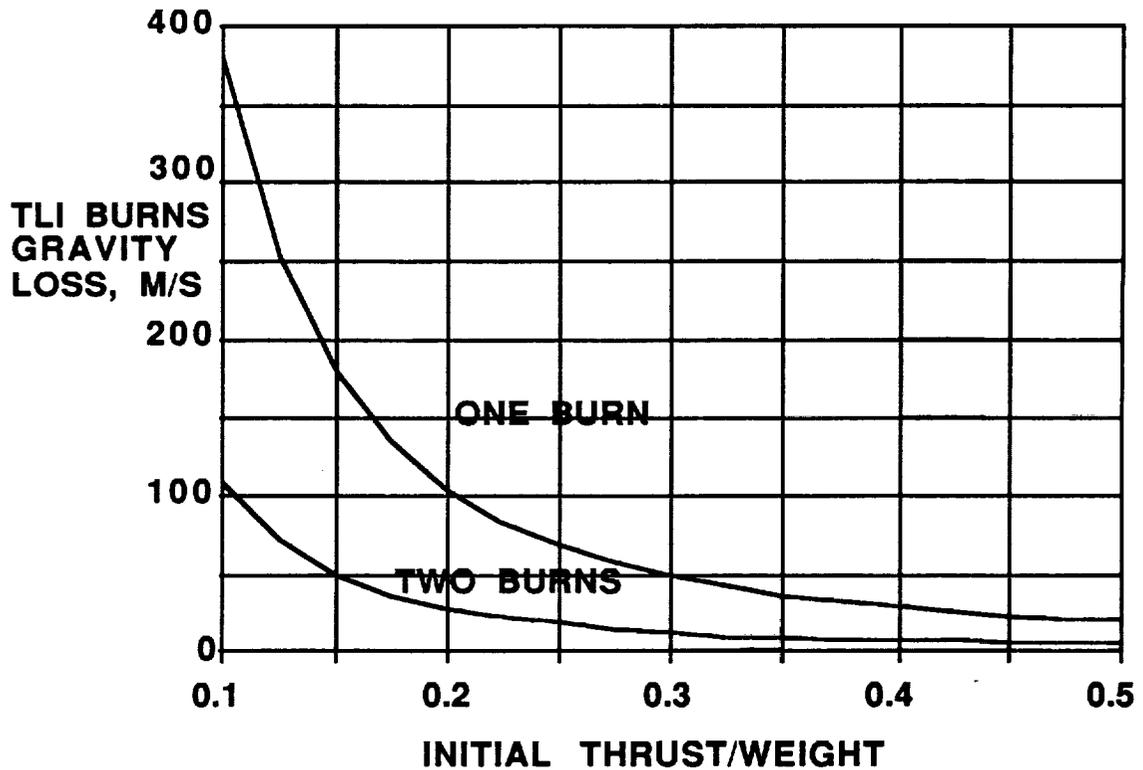
Figure 1-3.2.2.4-1. Gravity Losses for TLI Burn

Performance data for advanced space engines have been developed by Pratt & Whitney, Aerojet, and Rocketdyne. Examination of these data during earlier orbit transfer vehicle (OTV) studies found significant differences in performance predictions of the different manufacturers. The manufacturers data were used to construct Boeing-estimated specific impulse performance and engine weight characteristics (Figures 1-3.2.2.4-2 and 1-3.2.2.4-3) that were used for engine sizing.

SPECIFIC IMPULSE FOR SPACE TRANSPORTATION VEHICLES
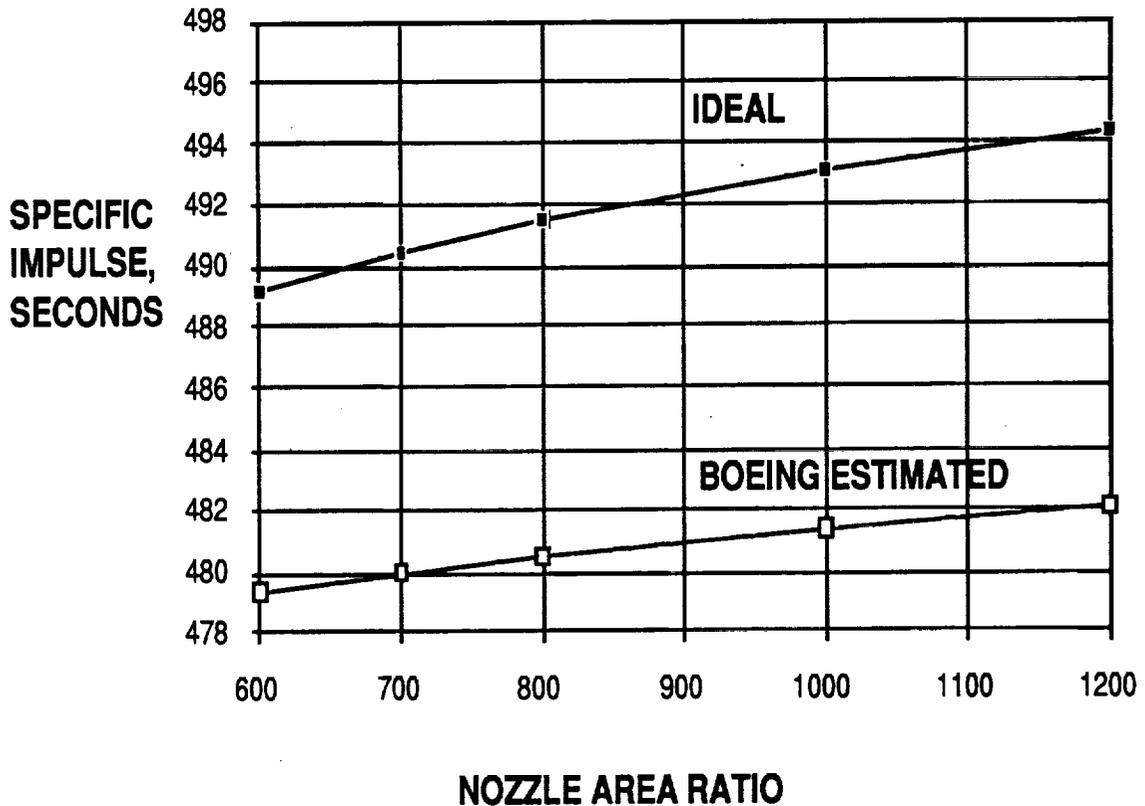
Figure 1-3.2.2.4-2. Boeing-Estimated Advanced Engine Isp Performance

Development costs for an advanced space engine will be on the order of $1 billion. Using the same engine (with the possible exception of the nozzle) for the LEV and LTV can therefore potentially save $1 billion in overall program costs. The LEV engines will require a low area ratio (200 to 300) nozzle for landing. A higher area ratio nozzle for the LTV engines can improve the vehicle performance because of the higher specific impulse available with the larger nozzle. The burnout mass after TLI and LOI burns was used as a criterion to select nozzle area ratio for the LTV engines. The burnout mass net of engines for an initial mass of 170,000 kg in LEO for configurations with five and seven engines of 15,000-lbf thrust each is shown in Figures 1-3.2.2.4-4 and 1-3.2.2.4-5. A nozzle area ratio of 600 appears to be a good choice based on the burnout mass net of engine mass with a 30% installation weight penalty.

Trades conducted to determine the optimum number of engines for the LTV were based on a groundrule that the mission could be completed after one engine failure occurring at any time during the mission. The end of mission burnout mass net of engines and reserve propellants required for one engine operation was the criterion used to determine the optimum number of engines. Figure 1-3.2.2.4-6 shows results of this trade for 15,000-lbf engines with an initial mass of 170,000 kg and a one burn lunar transfer injection burn mission. The burnout mass was near maximum with seven engines and was only slightly lower for a six-engine configuration. Significantly lower performance is obtained with five- and four-engine configurations.

ADVANCED ENGINE MASS
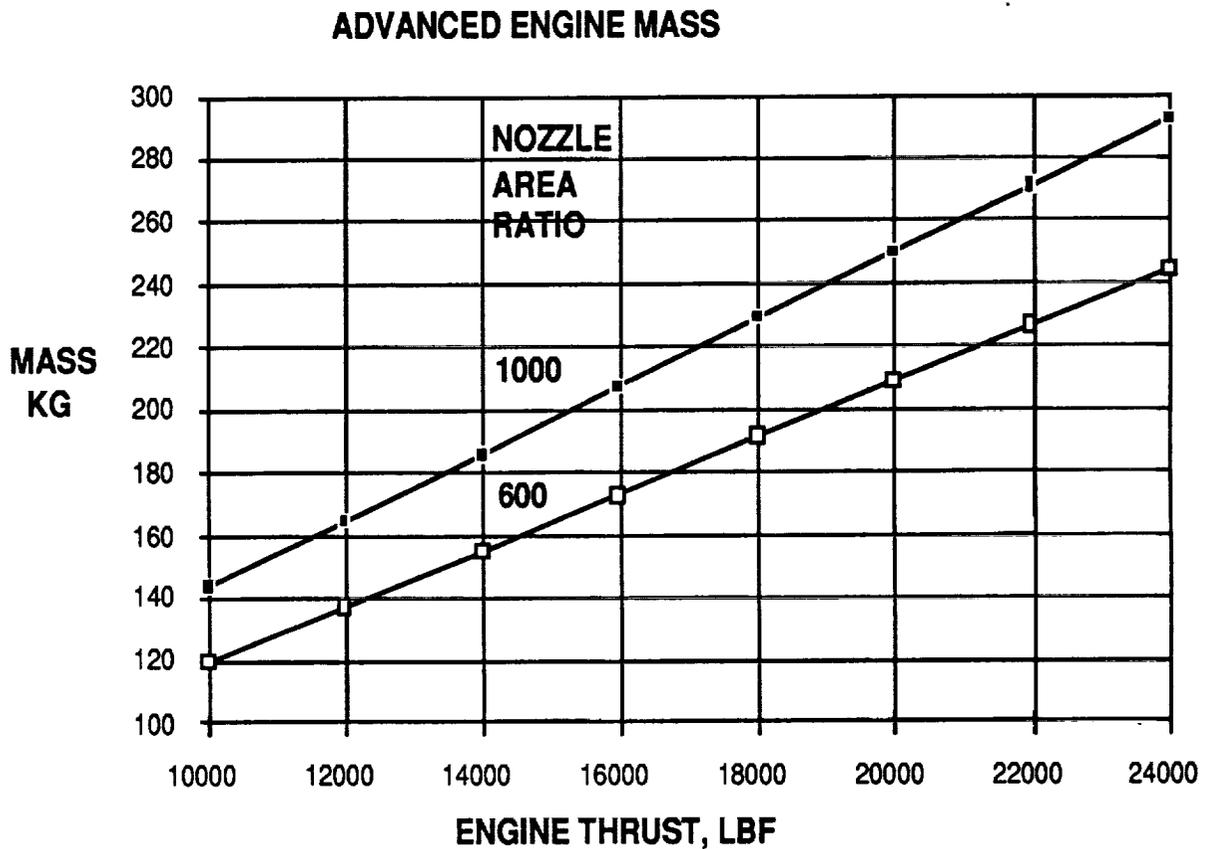
Figure 1-3.2.2.4-3. Boeing-Estimated Advanced Engine Mass

BURNOUT MASS NET OF ENGINES
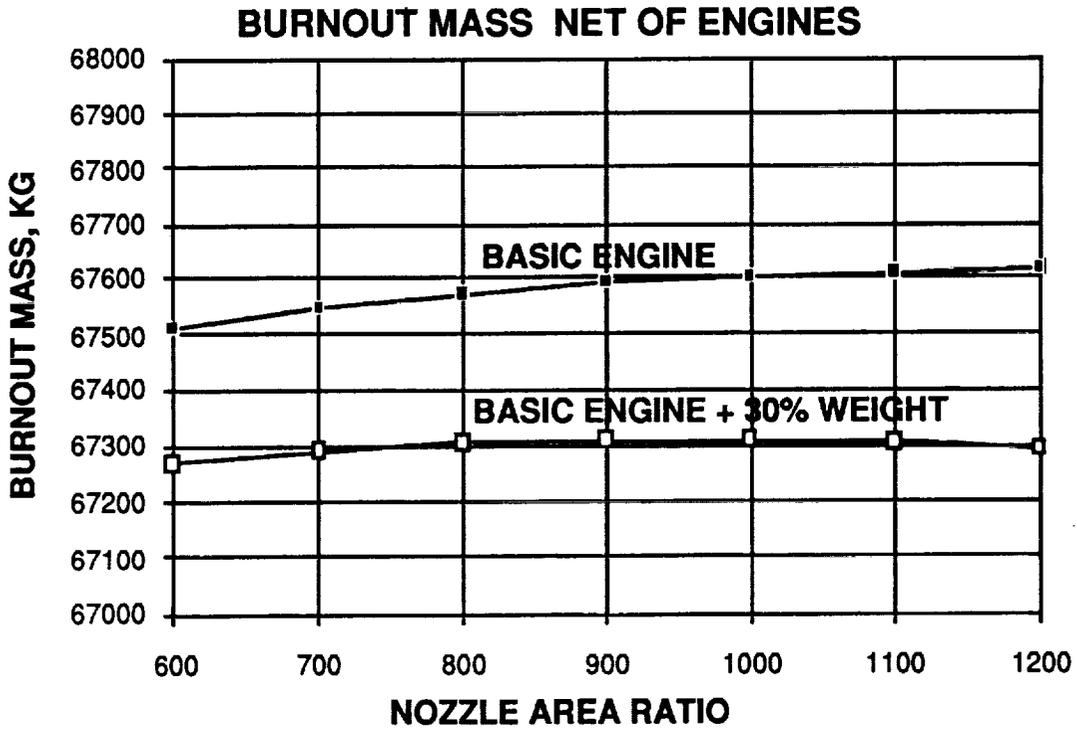
Figure 1-3.2.2.4-4. Nozzle Trade for TLI+LOI Burns, Five Engines at 15,000 lbf

NOZZLE TRADE FOR TLI+LOI BURNS, 7 ENGINES AT 15,000 LBF THRUST EACH, BOEING ESTIMATED ENGINE DATA, BURNOUT MASS NET OF ENGINES
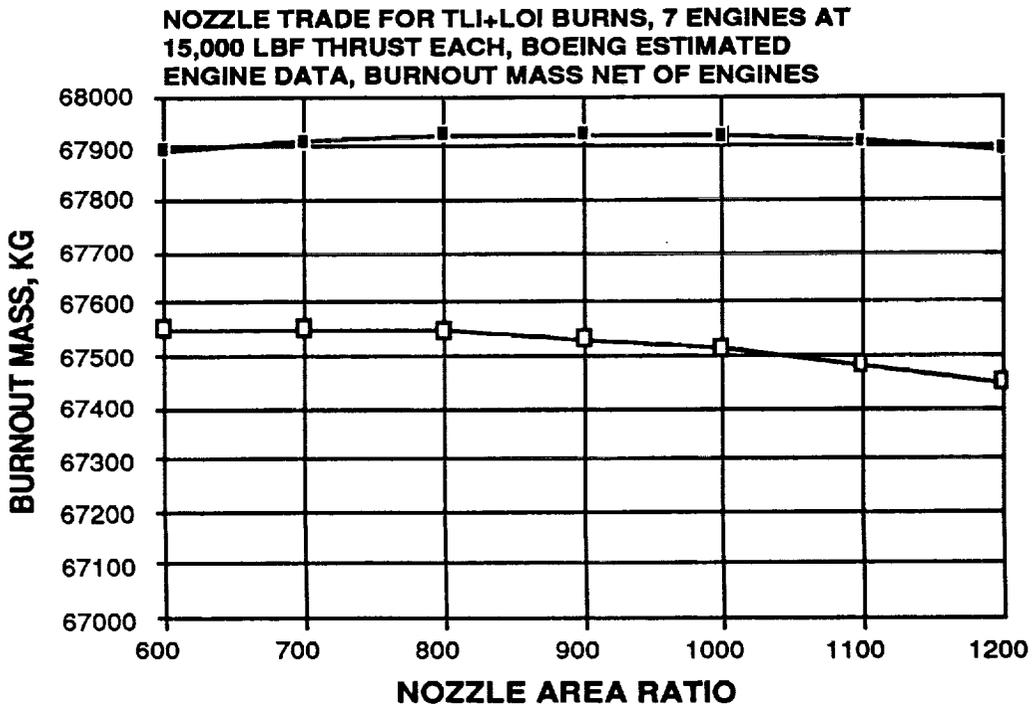
Figure 1-3.2.2.4-5. Nozzle Trade for TLI+LOI Burns, Seven Engines

•ENGINE THRUST, 66723 N (15,000 LBF)
•INSTALLATION WEIGHT PENALTY 30%
•INITIAL MASS IN LEO 170,000 KG
•BOEING ENGINE PERFORMANCE & WEIGHT

**MISSION END MASS NET
OF ENGINES & RESERVES**
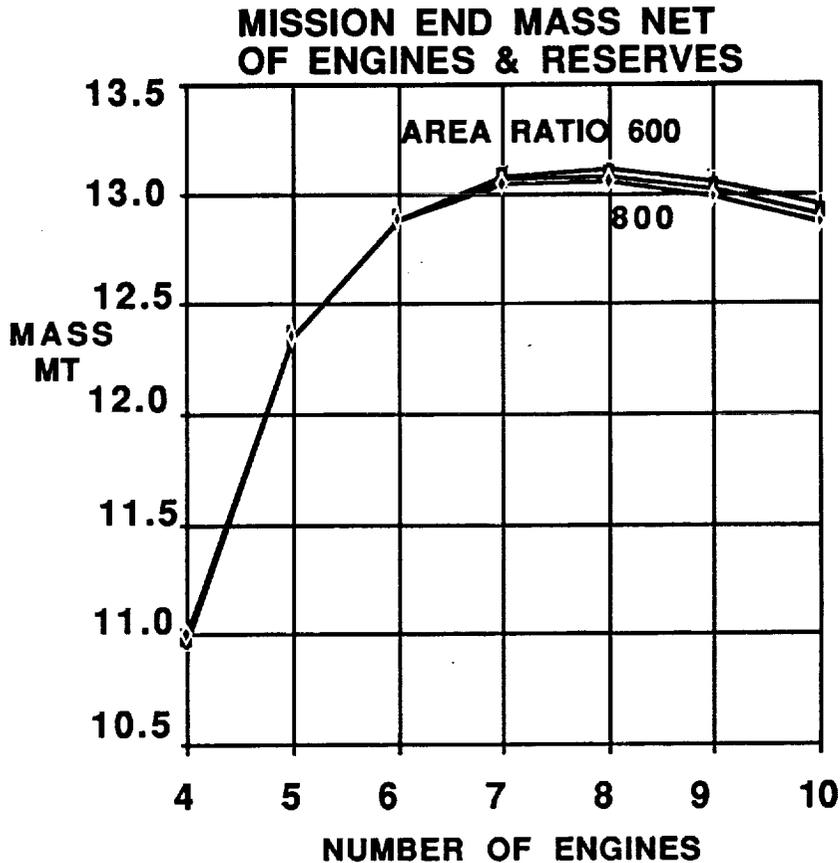
Figure 1-3.2.2.4-6. Mission Burnout Mass Nozzle Trade

The Boeing-estimated engine specific impulse (Isp) performance used for these trades is a conservative estimate of the gains available from higher area ratio nozzles. A 98% equilibrium Isp performance characteristic (Figure 1-3.2.2.4-7) was used to repeat engine and nozzle trades and validate the number of engines and nozzle area ratios selected. Figure 1-3.2.2.4-8 shows the results of the equilibrium performance analysis. These data show a small increase in burnout mass (net of engines and reserves) for the area ratio 1,000 nozzle in comparison with the area ratio 600 nozzle. The small increase of about 100-kg

burnout mass is not adequate to justify the larger envelope of the high area ratio nozzle.

Engine thrust level is a factor in selecting the number of engines for the LTV. Figure 1-3.2.2.4-9 summarizes trade study results for TLI with one or two burns for engine thrust levels of 12,000 to 20,000 lbf. Six engines at 20,000 lbf or seven engines at 15,000 lbf each are near optimum for the one-burn TLI mission. Four engines at 20,000 lbf or five engines at 15,000 lbf are optimum for the two-burn mission.
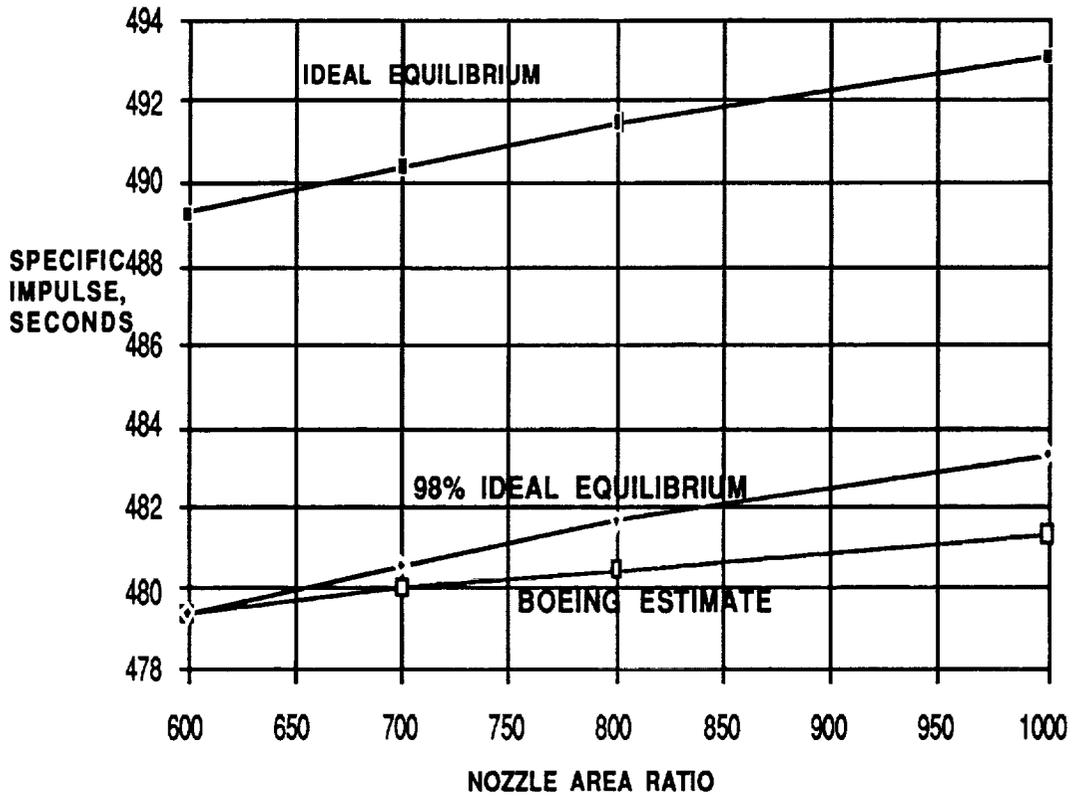
Figure 1-3.2.2.4-7. Equilibrium Specific Impulse Area Ratio Trend

BOEING

- ENGINE THRUST, 66723 N   (15,000 LBF)
- INSTALLATION WEIGHT PENALTY 30%
- INITIAL MASS IN LEO 170,000 KG
- BOEING ENGINE PERFORMANCE & WEIGHT
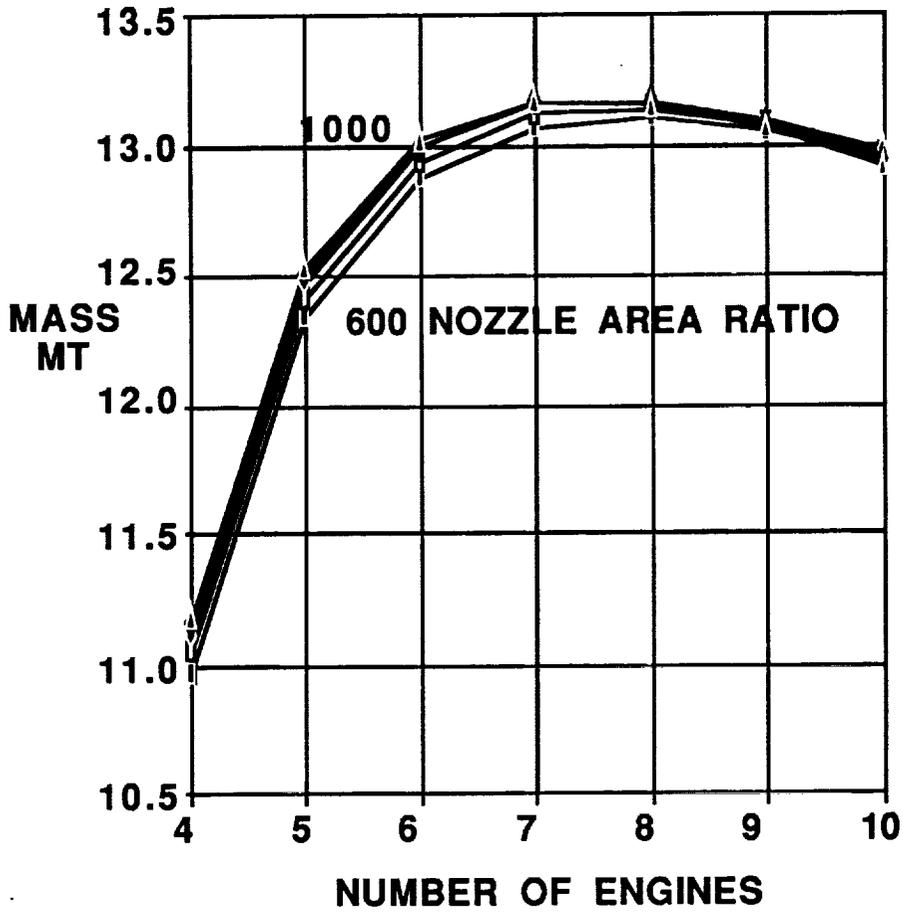
Figure 1-3.2.2.4-8. Mission Burnout Mass With Ideal Area Versus Isp Trend

- BURNOUT MASS NET OF ENGINES & RESERVES
- INITIAL MASS IN LEO 170,000 KG
- NOZZLE AREA RATIO 600
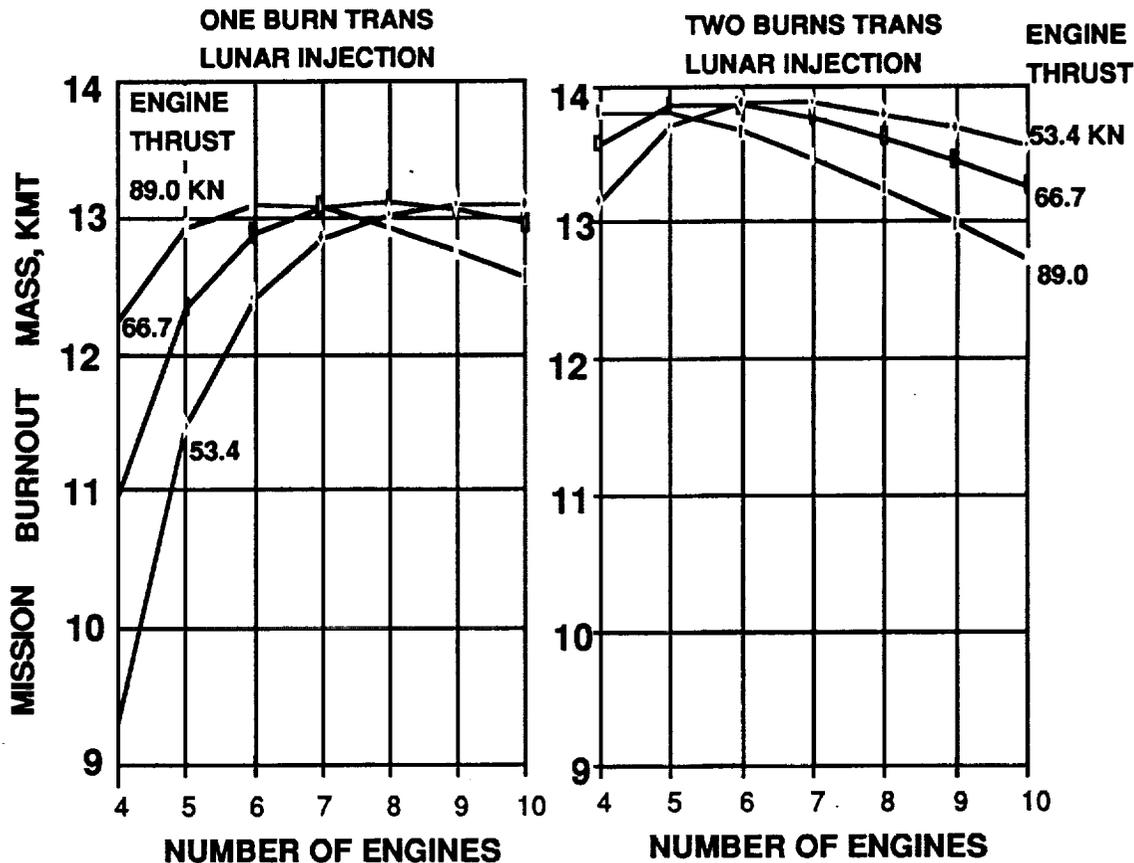
Figure 1-3.2.2.4-9. Number of Engines for TLI

1-3.2.2.5 Engine Applicability for Evolution to Mars Landing

Manned Mars missions are being considered as a successor to the currently planned lunar missions. It is highly desirable that a mature engine with proven reliability be available and be used for the first Mars mission landing vehicle. The engine planned to be used for lunar landing is a logical candidate for the Mars landing vehicle.

The large investment required for the Mars mission implies that the Mars landing mission must use systems capable of completing the mission with at least two failures including engines failures. Thrust requirements for the Mars

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landing with vehicles having four to seven engines are based on currently available vehicle mass data as listed in Figure 1-3.2.2.5-1. The engines must be capable of providing thrust over the range between that required for hover with two engines failed and the thrust of 76% hover with all engines operating. For example, engines for a five-engine vehicle must have a maximum thrust of at least 15,618 lbf and must be throttleable to a minimum of 7,028 lbf each. Similarity a six-engine vehicle must have engines with a thrust range of 5,857 lbf to 11,714 lbf . These thrust levels are within the 15,000- to 20,000-lbf range found to be suitable for the lunar landing.

The Mars ascent stage mass at liftoff is estimated at 22,336 kg or 49,139 lbm. A six-engine configuration with engine thrusts in the range of 15,000 to 20,000 lbf should be adequate for this liftoff mass without excessive gravity losses.

- •Mars landing engine thrust requirement is based on:
  - •Land successfully with two engines failed
  - •Total mass landed is 53,168 kg
  - •Mars gravitational acceleration is 3.92 meters per second square
  - •Total thrust for landing is 156,315-208,420 newtons (75%-100% hover)

Engine thrust for hover with two engines failed

Engines Installed	Engines Operating	Thrust per engine	
		Newtons	Pounds
4	2	104,210	23,427
5	3	69,473	15,618
6	4	52,105	11,714
7	5	41,684	9,371

Engine thrust for 75% hover, all engines operating

Engines Installed	Engines Operating	Thrust per engine	
		Newtons	Pounds
4	4	39,079	8,785
5	5	31,263	7,028
6	6	26,053	5,857
7	7	22,331	5,020

Figure 1-3.2.2.5-1. Engine Thrust Required for Mars Landing

Engine Design Margins. Design margins for the STV engine should be higher than normally used for unmanned vehicles that have no reusability requirements. Increased design margins should provide the increased reliability and longer life needed for the human exploration program. Trade studies to determine the optimum engine life and required design margins should be conducted by the engine manufacturers.

Engine Configuration and Characteristics. The STV engine is expected to be space based with a primary mission to support the human exploration program for several years. The STV engine will also be required to provide propulsion capability for a variety of commercial and military missions. High reliability is essential to achieve a man-rated vehicle capable of efficient operation in a space-based mode. Design for maintainability in space is also a major consideration in efficient operation of the propulsion system.

Propellant Storage and Delivery Systems. The space-based STV is intended to perform five missions over a 5-year period. Designs to achieve this goal must have minimal maintenance requirements and be based on proven components with high reliability. The TLI and LD propellant feed system (Figure 1-3.2.2.5-2) has 14 dual-seat valves, including disconnects, which are required to complete the mission with any single valve failure and satisfy man-rating requirements for safe return with two failures. Similarly, the LEV has 60 dual-seat feed system valves and disconnects (Figure 1-3.2.2.5-3) to satisfy man-rating requirements.
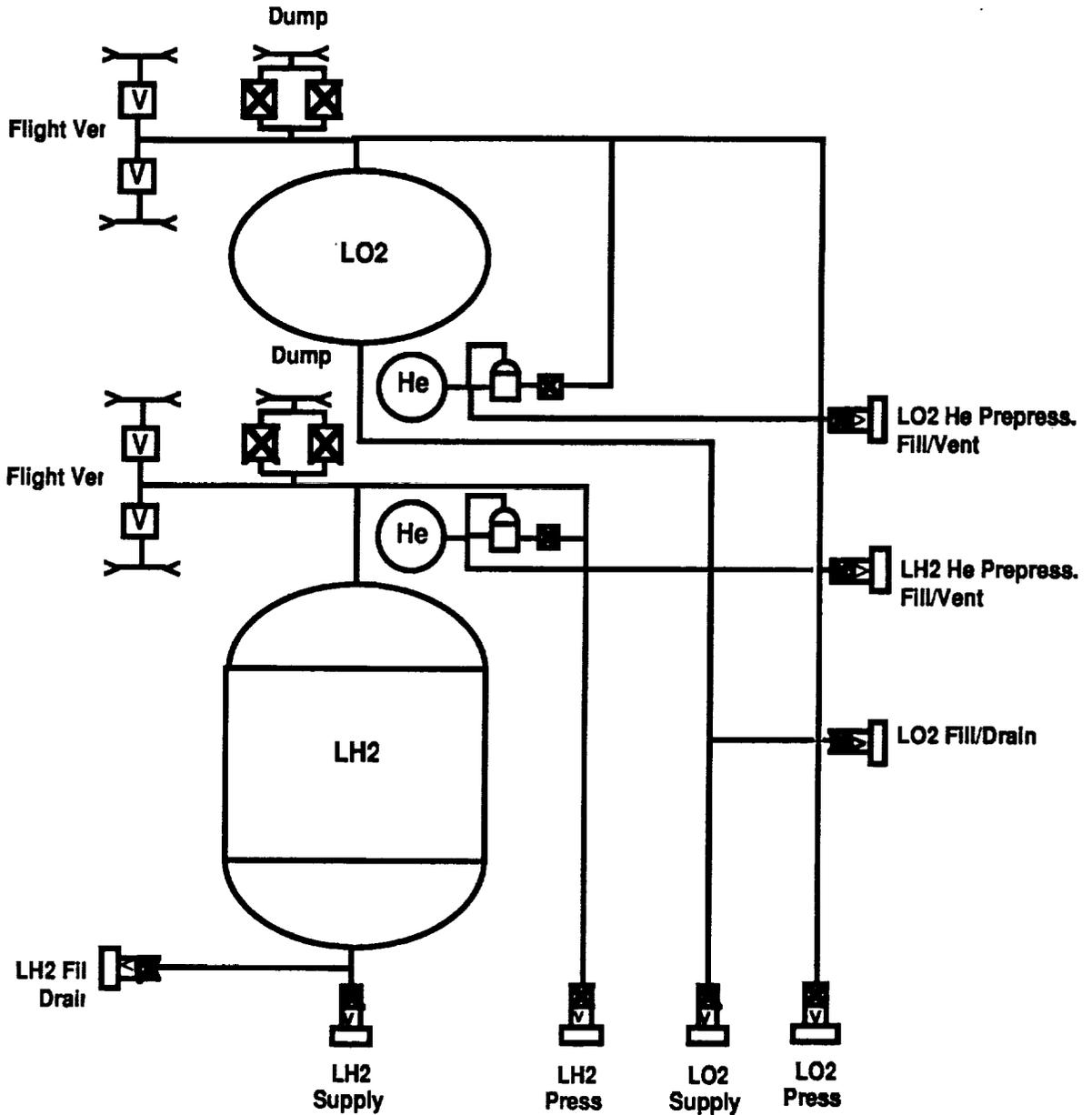
Figure 1-3.2.2.5-2. LTV Propellant Feed System Schematic

The large number of valves in the propellant feed systems causes concerns about reliability and maintainability. Some of the valves may not be readily accessible when all tanks are installed and loaded unless the system design is forced to consider accessibility at an early phase. The reliability of cryogenic valves is not known to a high degree of accuracy. A failure rate of 236 failures per million cycles was estimated from RL10 historical data.
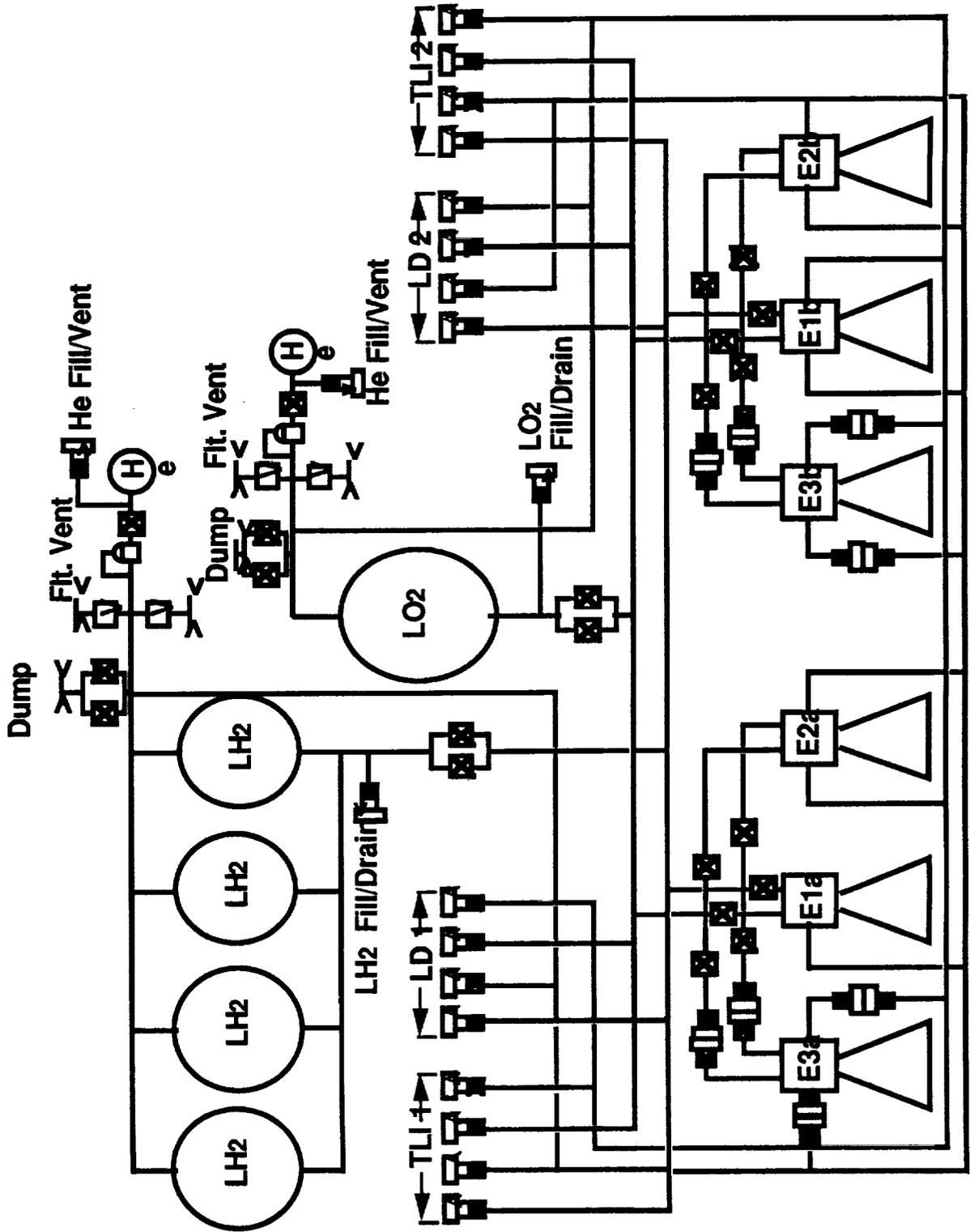
Figure 1-3.2.2.5-3. LEV Propellant System Schematic

The probability of a valve failure in the STV propellant feed system was estimated as a function of the number of valve cycles using the RL10 data with the results shown in Figure 1-3.2.2.5-4. Each complete mission will result in 5 to 10 cycles per valve; therefore, the probability is about 50% that one failure will occur after no more than two missions. It is important to note that the feed system probably includes no more than one-third of the total valves in the propellant system.

- •VALVE RELIABILITY BASED ON RL10 INLET VALVES
  - •TOTAL 1470 FIRINGS WITH NO FAILURES THROUGH MAY, 1988
  - •COMBINED FUEL AND OXIDIZER VALVES DUE TO SIMILAR DESIGN
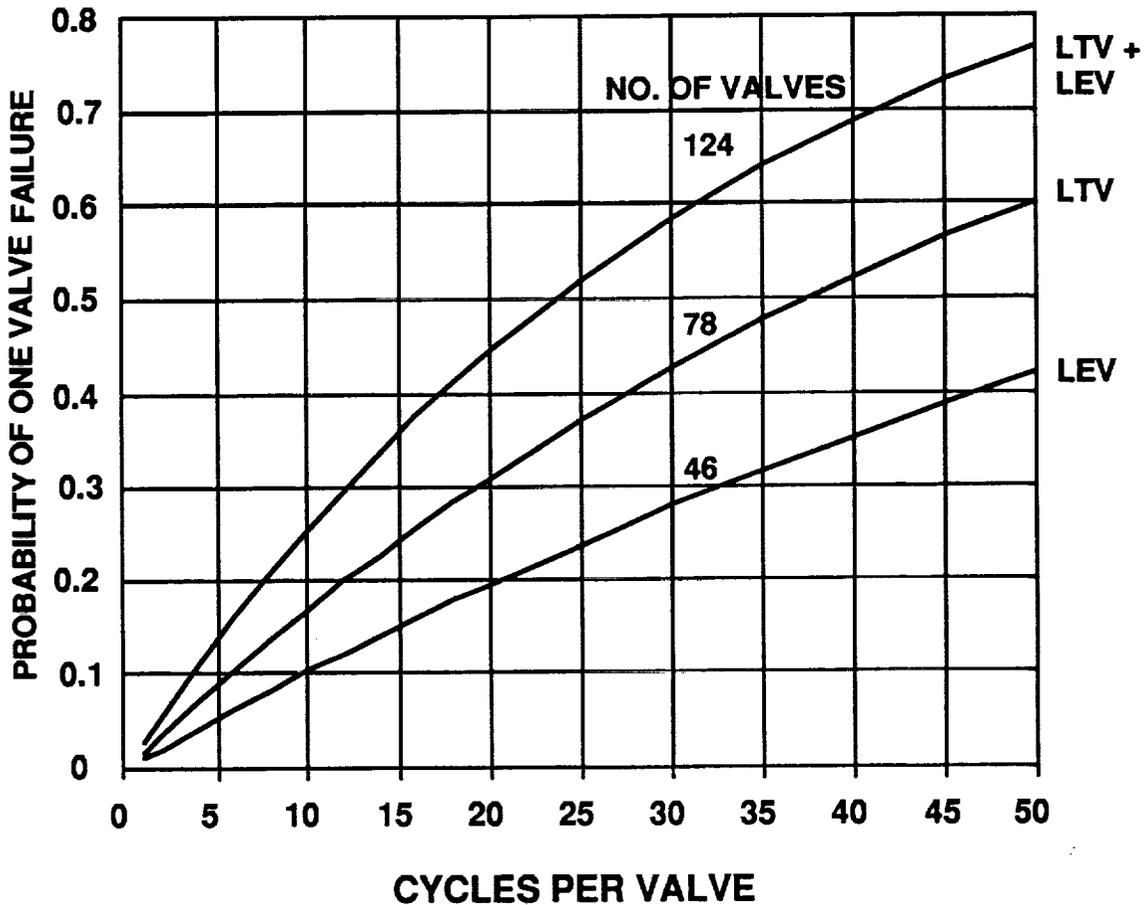
Figure 1-3.2.2.5-4. STV Propellant Feed System Valve Failure Rate

Figure 1-3.2.2.5-5 presents results of a preliminary study to determine optimum engine interface pressure requirements based on pump weights and gaseous residuals in the tanks. Increasing the oxygen NPSH requirement above 15 kPa

(~2 psi) will cause tank gaseous residual mass increases in the large droptanks more than the engine pump weight reduction for the four-engine configuration. The increased pump weight for the six-engine configuration shows some benefit from increasing the NPSH requirement to about 25 kPa with consideration of only the large droptank residuals resulting from the TLI burn. This result clearly depends on the vehicle tank configuration and mission burn sequence. If the same engines are used for additional major burns, the weight advantage will shift to the zero NPSH requirement. It should be noted that the NPSH trade is based on full-power operation only. Pumped idle engine operation with zero NPSH is required for tank pressurization to eliminate any requirement for helium on the vehicle.

• NET MASS CHANGE INCLUDES GAS RESIDUALS IN TWO LARGE
  DROP TANKS AND CHANGES IN ENGINE PUMPS WEIGHT WITH NPSP.
• ENGINE THRUST 66723 N (15,000 LBF) EACH
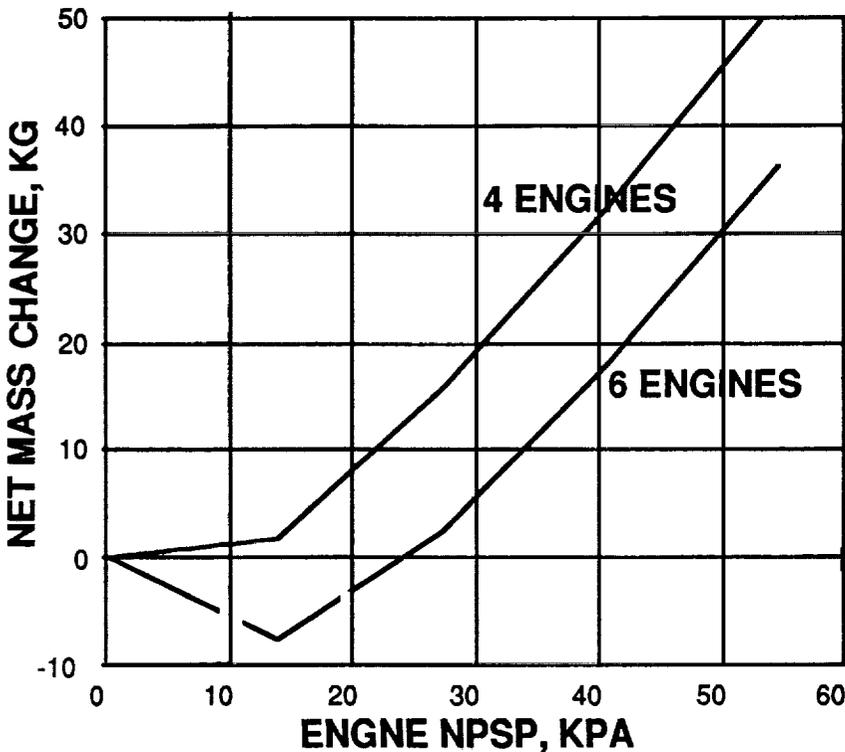
Figure 1-3.2.2.5-5. Engine Weight as Effected By Pump Inlet Pressure

MPS Component Characteristics. The technology issues involved with the MPS control components are mostly of a design and demonstration nature with no enabling new technology envisioned at this time. Component designs, as conceived, are essentially state of the art and appear to be solvable through normal development processes. Top-level requirements for major MPS components are listed as follows:

1. Ducting.
   a. Engine gimbal motion absorbed by vehicle ducts (pumps fixed to engine).
   b. All cryogenic lines insulated (vacuum or inert gas filled jackets to be traded).
   c. All LO2 and LH2 pump inlet lines built on common tooling to the greatest extent possible.
   d. Minimum bellows spring rate.
2. Staging disconnects.
   a. New designs are to be investigated.
   b. Positive sealing at staging.
   c. Redundant sealing.
   d. Simple, high reliability.
   e. Low actuation forces and separation reactions.
   f. Couplable by suited astronaut for space-based system.
   g. Meet STV response and leakage requirements.
3. Valve and controls.
   a. Dual seal ball valves with redundant cavity relief valves.
   b. Commonality between applications (e.g., fill/drain, vent, and prevalve).
   c. Dual-valve actuation, EMA versus pneumatic to be traded (no hydraulics).
   d. Integral lockup relief on series cryogenic valves.
4. Hazardous gas detection.
   a. Integrated into VHMS for continuous monitoring on ground and in flight.
   b. The advanced hazardous gas detection system and the rocket engine leak detection mass spectrometer, under development at KSC, are currently of interest.
   c. Evaluating other promising concepts.

1-3.2.3   Reaction Control System Definition

1-3.2.3.1   System Requirements

The reaction control system (RCS) is used for vehicle orientation and control for all phases of the mission. To attain total redundancy and minimize cost, the system must be as simple as possible and use as many common components as possible and preferably a common thrust level for all maneuvers. To further simplify the system and demonstrate consistent performance, common propellants with the MPS should be used.

To meet mission requirements the RCS response requirements, shown in Figure 1-3.2.3.1-1, were developed to ensure adequate control authority for each phase.

Maneuver	Roll ($^\circ$/sec^2)	Pitch ($^\circ$/sec^2)	Yaw ($^\circ$/sec^2)
Trans Lunar Injection	0.05-0.10	0.05-0.10	0.05-0.10
Lunar Deorbit	0.1-0.3	0.1-0.3	0.1-0.3
Lunar Landing	-	5.0-10.0*	5.0-10.0*
Trans Earth Injecton	0.1-2.0	0.1-2.0	0.1-2.0
Earth Reentry	3.0-5.0	1.0-2.0	1.0-2.0

* Apollo pilot rating requirements met by engine gimbaling.

Figure 1-3.2.3.1-1. RCS Response Requirements

1-3.2.3.2   System Definition

Two vehicle configurations were considered: the ground based and space based. To maintain commonality of propellants on the vehicles, gaseous oxygen and hydrogen have been selected for the RCS. Both the oxygen and

hydrogen will be stored as supercritical gas in high-pressure bottles. These bottles could be provided full from the ground or filled with liquid at low pressure prior to launch with the possibility of being refilled during flight from the main propulsion system tanks. Total redundancy could not be achieved with identical systems for both configurations because of differences in moment of inertia, center-of-gravity locations, and available locations for mounting thrusters caused by fundamental differences in operation and resulting configurations.

The space-based RCS configuration is shown in Figure 1-3.2.3.2-1. For this configuration, center-of-mass locations vary between 8.52m to 5.60m along the vehicle axis from the center of RCS thrust. Tankage dropped following TLI burn and on the lunar surface limit available interface areas. Maximizing available moment arms to reach control authority goals dictated selection of the bottom surface of the core module, near the MPS engines as the best for mounting the RCS thrusters. Calculated mass moments of inertia and center-of-mass locations are shown for each segment of the mission.

Mode	Inertia (Kg*m^2)
TLIR	4.8E6
TLIP	7.1E6
TLIY	3.6E6
LLOR	2.3E6
LLOP	2.0E6
LLOY	8.7E5
TEIR	2.6E5
TEIP	2.7E5
TEIY	2.8E5
R'ENTR	2.0E5
R'ENTP	2.0E5
R'ENTY	2.1E5
LLP	7.4E5
LLY	7.1E5

TLI Sta 7.57
R'Ent Sta 5.64
TEI Sta 5.35
LL Sts 4.75

Gimbal Plane Sta 0

¢ RCS Thrust Sta -0.85

Cargo

Type 1 RCS Module
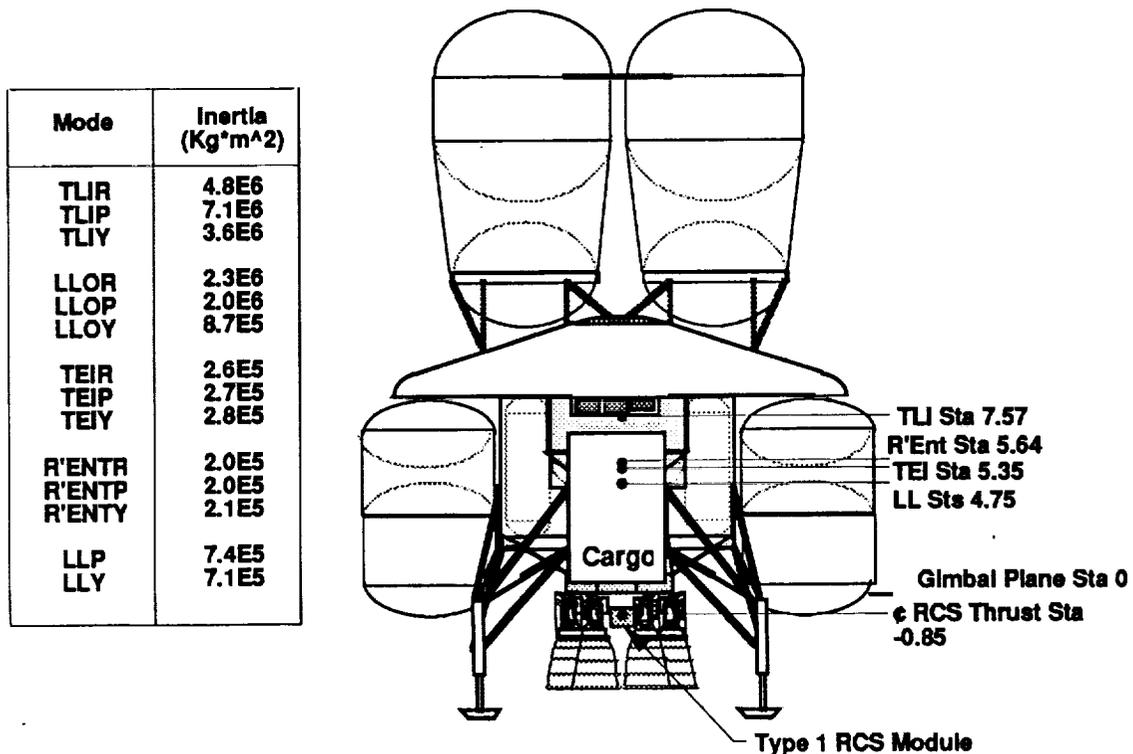
Figure 1-3.2.3.2-1. Space-Based Vehicle Mass Properties and RCS Location

Thruster arrangement for the space-based configuration is shown in Figure 1-3.2.3.2-2. Two clusters of eight engines are mounted to the lower core module surface, providing complete redundancy of operation.

The schematic arrangement for the space-based vehicle RCS is shown in Figure 1-3.2.3.2-3. Redundant O_2 and H_2 tanks provided with electric resistance heaters supply a manifold connecting two thruster clusters and three fuel cells, which provide power for vehicle functions as well as maintaining the propellant quality. The thruster clusters are mounted on the vehicle as discussed.
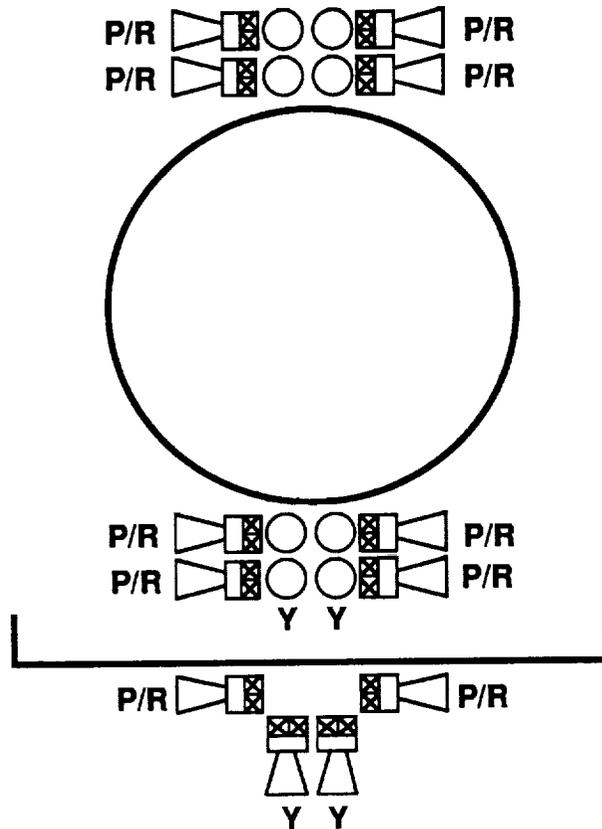
Figure 1-3.2.3.2-2. Space-Based RCS Thruster Arrangement
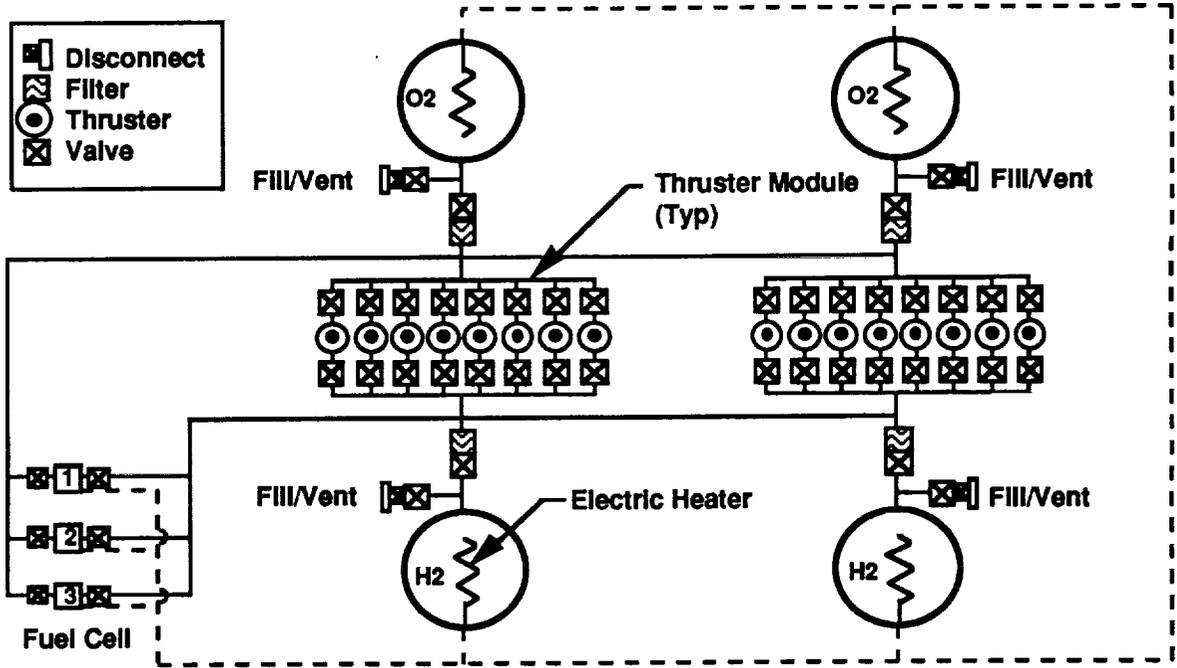
Figure 1-3.2.3.2-3. Space-Based Vehicle RCS Schematic

Using the data from Figures 1-3.2.3.2-1 and 1-3.2.3.2-2, minimum and maximum thrust requirements were calculated for each mission condition, as shown in Figure 1-3.2.3.2-4. From these data, a thrust level was selected that best fit the thrust requirements; that is, a value larger than the minimum offering total redundancy and within the maximum where possible (because the minimum values are critical, exceeding the maximum by as much as 50% will not cause problems meeting mission requirements).

Mode	Thrust (a min)	Conditions	Thrust (a max)
TLIR	578n		1156n
TLIP	736		1472
TLIY	433		867
LLOR	554		1661
LLOP	563		1689
LLOY	244		735
TEIR	63		1252
TEIP	84		1683
TEIY	87		1348
R'ENTR	1444	← Select 160lbf Thrusters (712n ea.)	2407
R'ENTP	537		1079
R'ENTY	506		1011
	Gimbal (a min)		
LLP	6.8°		
LLY	6.5°		

Figure 1-3.2.3.2-4. Space-Based Vehicle RCS Thrust Requirements and Selection

The ground-based RCS configuration is shown in Figure 1-3.2.3.2-5 along with calculated center-of-mass locations and mass moments of inertia for each mission segment. Operation of this vehicle through lunar landing is the same as the space-based vehicle. Following lunar liftoff and Earth orbit insertion, however, the core propulsion system is staged off the crew reentry vehicle, which continues back for Earth landing, with RCS functions still required through the Earth reentry phase. This results in the need for two RCS systems: one for ascent to the Moon and another for return to the Earth.

The large base structure offers an ideal location for the outbound RCS thrusters, providing large moment arms for roll, pitch, and yaw control. The core stage requires a different approach. Since the core MPS separates for coast and reentry, the RCS must be mounted on the reentry capsule. Fortunately, the centers of mass are suitably located for major control operations, requiring relatively low thrust to attain the required control authority.
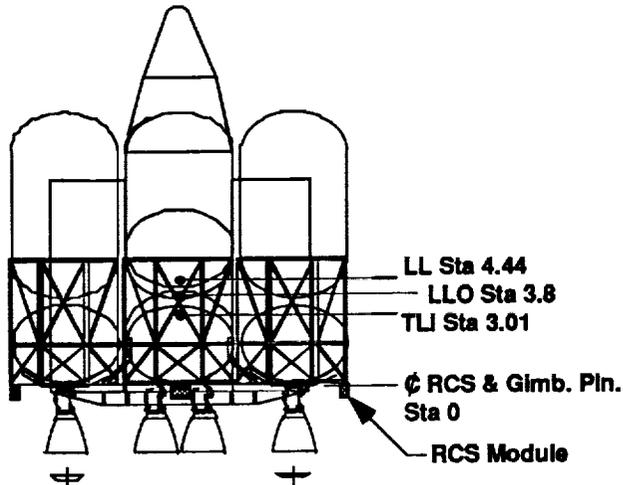
Outbound to Moon

Mode	Inertia (Kg *m^2)
TLIR	4.2E6
TLIP	4.7E6
TLIY	3.7E6
LLOR	1.2E6
LLOP	2.2E6
LLOY	1.5E6
TEIR	3.0E4
TEIP	5.2E5
TEIY	5.2E5
R'ENTR	2.1E4
R'ENTP	1.7E4
R'ENTY	2.1E4
LLP	1.4E6
LLY	1.3E6
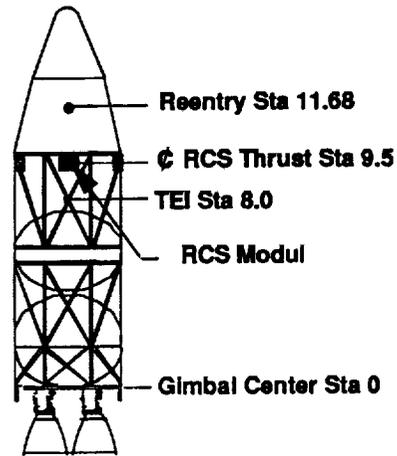
Return to Earth

Figure 1-3.2.3.2-5. Ground-Based Vehicle Mass Properties and RCS Locations

Thruster arrangement for the ground-based configuration is shown in Figure 1-3.2.3.2-6. Two clusters of six engines are mounted on the yaw axis and two modules of four engines are mounted on the pitch axis. This provides total redundancy and a single fault tolerant system, while providing the use of multiple engines to perform maneuvers requiring larger than nominal single-engine thrust. Both the outbound and the return systems use this arrangement.

The schematic arrangement for the ground-based vehicle RCSs is shown in Figure 1-3.2.3.2-7. Essentially it is identical to the system shown in Figure 1-3.2.3.2-3 except for appropriate revision to the thruster modules.
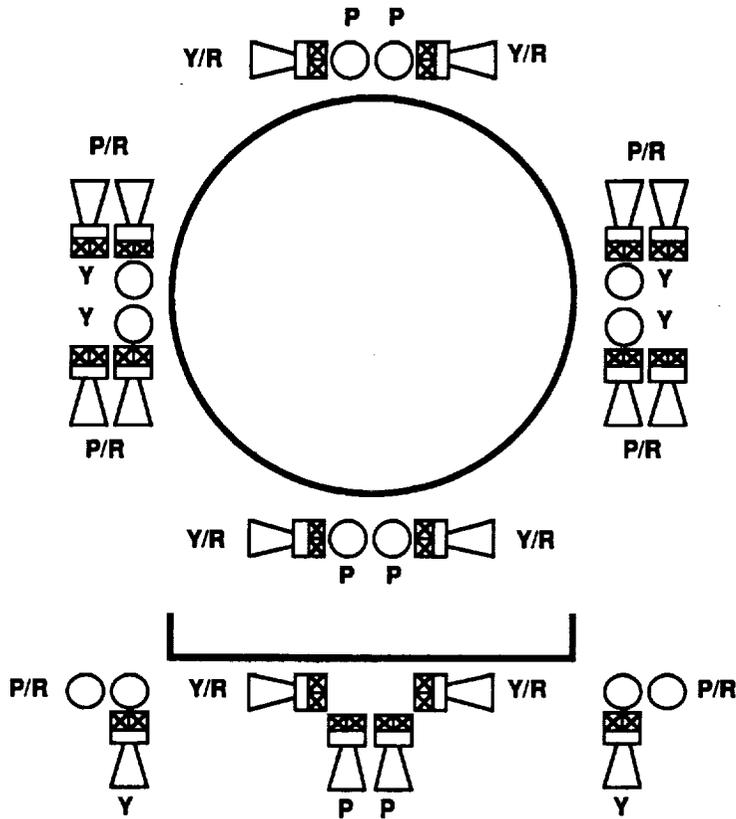
Figure 1-3.2.3.2-6. Ground-Based Vehicle RCS Thruster Arrangement
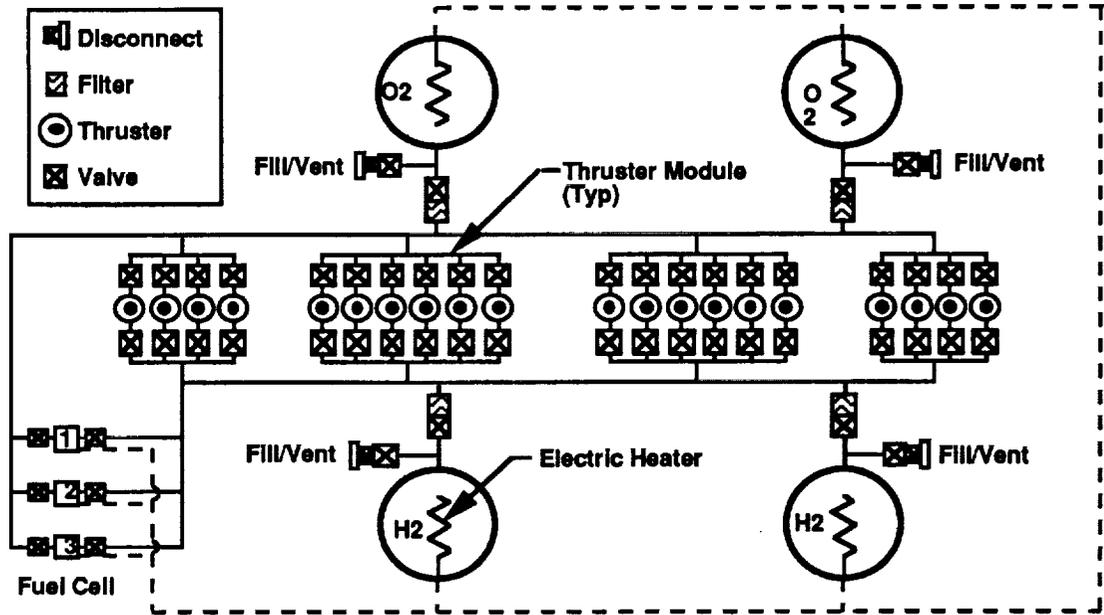
Figure 1-3.2.3.2-7. Ground-Based Vehicle RCS Schematic

Using the data from Figures 1-3.2.3.2-1 and 1-3.2.3.2-5, minimum and maximum thrust requirements were calculated for each mission condition, as shown in Figure 1-3.2.3.2-8. From these data, a thrust level was selected to the same criteria as the space-based vehicle. Because of the broader range of mass moments of inertia calculated for the core stage during the Earth bound and reentry portions of the mission, some maneuvers use fewer thrusters than are available to prevent grossly overpowering the requirements.

Mode	Thrust (a min)	Conditions	Thrust (a max)
TLIR	555n	← Select 150lbf Thrusters (668n ea.)	1111n
TLIP	621		1367
TLIY	489		978
LLOR	317		952
LLOP	582		3031
LLOY	397		1190
TEIR	19	2 Roll Thrusters Only	388
TEIP	336	4P or 2P + 2P/R Thrusters to Make Min. Rqmt.	1008
TEIY	336	4Y or 2Y + 2Y/R Thrusters to make Min. Rqmt.	1008
R'ENTR	407	← Select 25lbf Thrusters (110n ea)	679
R'ENTP	135		270
R'ENTY	136		271
	Gimbal (a min)		
LLP	9.5°		
LLY	8.8°		

Figure 1-3.2.3.2-8. Ground-Based Vehicle RCS Thrust Requirements and Selection

1-3.2.3.3   RCS Component Status

With the exception of the thrusters, all components required for the RCSs have been qualified in a usable form for space flight. Oxygen and hydrogen bottles will have to be scaled to meet the capacity of the STV systems and requalified; however, the technology is in place to build the system.

In support of the early Space Station design effort, Boeing and MSFC developed an oxygen/hydrogen thruster testbed and conducted extensive testing on several thruster configurations. These thrusters were built by both Aerojet and Rocketdyne and were of 25 lbf (110N). Two potential problem areas were considered very important: verification of combustion stability of the gas-gas propellants and developing a long-life, reliable igniter for oxygen/hydrogen use.

Combustion stability was demonstrated by firing thrusters with several different injector conceptual designs, all of which yielded satisfactory results. A reliable igniter was developed, solving the second problem. The pulse tube ignition

system ignites the propellant mixture with an acoustic energy pulse using no electrical power or moving parts. The design appears to be scaleable to meet any of the derived STV requirements.

1-3.2.3.4 Electrical Power System

Electrical power is a flight critical subsystem during all phases of flight. The selected electrical power system (EPS) must provide adequate, reliable power, even in contingency (abort) operations.

Within the near-term TAD goals, there are sufficient EPS technologies available to meet the requirements for a STV. The options for power sources include a variety of hardware that is already man rated and flight qualified. Batteries offer a simple, reliable energy source, although the system weight is significant. Solar photovoltaic cell offer a lightweight power source for orbital operations, unfortunately, the highest power loads occur during ascent and descent when the solar array would be useless. Fuel cells and auxiliary power units (APU) can provide high power levels at reasonable system weights, but they are more mechanically complex. The mission duration and power load profile will ultimately determine the selected EPS concept.

The STV requires between 3 and 8 kW for the duration of the mission. Using Figure 1-3.2.3.4-1 to consider the options on a weight basis indicates that fuel cells provide the lowest weight option for the ≈5-kW average load for the up to 30-day mission duration.

Other advantages of the use of a O2/H2 fuel cell include being able to use the byproduct water to provide crew drinking water during the duration of the lunar mission. This means that the vehicle will require less consumables because, essentially, the O2/H2 is used twice, once as reactants for power generation and once as crew water. A second advantage comes from the fact that the STV has LOX/LH2 main engines and GOX/GH2 RCS thrusters. Because the fuel cells will be able to use propellant grade reactants, fuel from the main tanks can be pulled into the supercritical supply tanks for the fuel cells. This synergism will reduce operational complexity by reducing the number of fluids in the system.

POWER SYSTEM APPLICABILITY

System/Subsystem Concepts
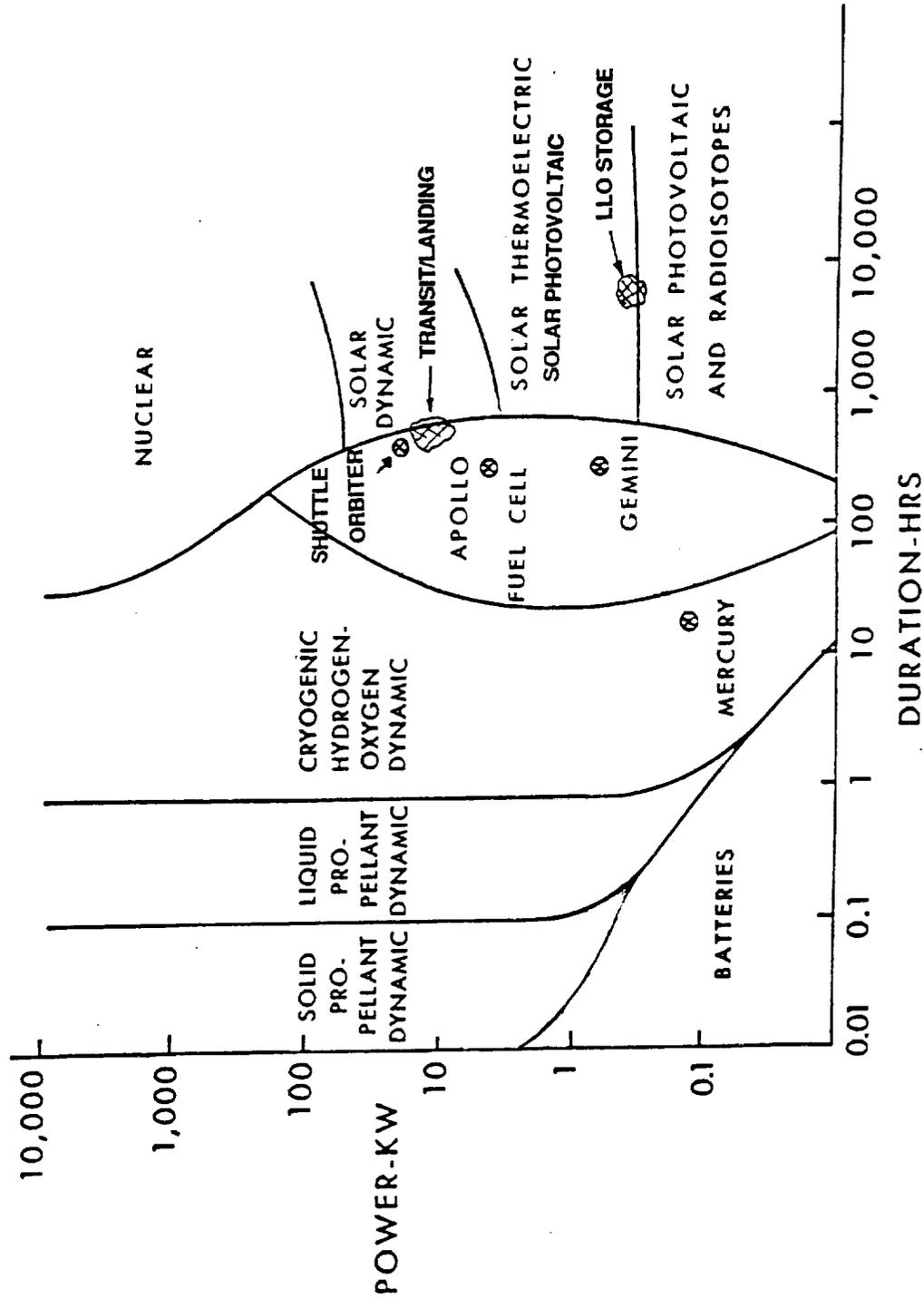
Figure 1-3.2.3.4-1. *Power System Applicability*

If the fuel cells are sized for the short duration, high-peak loads will occur whenever the main engines TVC is operating. There are two options that were considered during this study as the method to handle these peak loads. These options are batteries (either rechargeable or expendable) or an APU.

The batteries considered were either nickel hydrogen for the rechargeable case or lithium thionyl chloride for the expendable case. The disadvantage of the one-use batteries was that although they had a significantly higher power density than the nickel hydrogen batteries, they introduced the additional operational complexity of having to replace the battery. These batteries, once they have been used, have a life of 1 year. This means that the batteries would either have to be replaced after each mission or if the mission was for any reason extended beyond the current 6-month stay time, this type of battery would have to be replaced. The added operational complexity of having to replace the batteries after every flight was also a battery decision factor. For these reasons, the rechargeable battery case was chosen.

The APU alternative would use a percentage of the engine hydrogen mass flow to drive a turbine and subsequently a power generator. This option promises a lighter weight system and requires further exploration.

1-3.2.4   Avionics System

Flight Element Definition Study. During the architecture trades, the flight elements that make up the configurations required some definition of the avionics in those elements. During this definition of the different flight elements, it also became apparent that two distinct options for the avionics configuration were created by the desire to maintain a common vehicle configuration for both the piloted and unpiloted (i.e., cargo) configuration (Figure 1-3.2.4-1). These flight elements (the crew cab, propulsive stages, droptanks, and the aerobrake), when all the variations are considered, gives a total of 31 distinct flight elements. These 31 flight elements were combined to provide vehicles that were capable of flying the required 94-orbit options being explored in the architecture studies.

AVIONICS LOCATION
AVIONICS SUBSYSTEMS

STV

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AVIONICS OPTION / MISSION	OPTION 1 P/A MODULE TYPE	OPTION 2 AVIONICS IN CAB
CARGO Flt 0, 1, 4, 8	- BASIC AVIONICS IN LANDER/ TRANSFER STAGE - autonomous rendezvous & landing - autonomous power control	- add pallet in place of cab - add autonomous ops
PILOTED Flt 2, 3, 5-7, 9-24	- add MMI, C&T, and ECLSS for crew - Overridable full auton- omous ops for landing/ rendezvous	- BASIC AVIONICS IN CAB - partial autonomy in selected mission phases

Figure 1-3.2.4-1.   Avionics Location

STV PITCH

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Once the flight element options were defined, it was also necessary to define three different cases for the vehicle types. The three cases are defined by the number of propulsion stages and are shown in Figure 1-3.2.4-2. The impact of the common vehicle that can fly both piloted and unpiloted are shown in the case of the one propulsive stage vehicle in Figure 1-3.2.4-3.

Avionics Location Options. Expanding the analysis for the avionics locations to the one propulsion module case to the two and three propulsion stage case yields a matrix of flight elements that shows which of those elements contain avionics (Figure 1-3.2.4-4). To more clearly define what the avionics suites required for each element contained, each element must be considered separately.

The avionics must provide certain functions throughout the vehicle (Figure 1-3.2.4-4). If the vehicle is configured using Option 1 (smart propulsion stage and minimal crew module avionics), the avionics splitout for each element is shown in Figures 1-3.2.4-5 through 1-3.2.4-8. Combining these elements into the space-based configuration generates an avionics configuration, which is shown in Figures 1-3.2.4-9 and 1-3.2.4-10.

If, however, the vehicle is configured using the Option 2 avionics configuration (dumb lander and smart crew cab), the flight elements and their avionics elements are as shown in Figures 1-3.2.4-11 through 1-3.2.4-15. This combination of elements leads to a avionics configuration as shown in Figures 1-3.2.4-16, 1-3.2.4-17, and 1-3.2.4-18.

Functional Allocation and Partition. Each identified function was considered for its level of criticality and complexity. To support team costing exercise, a hardware utilization list was identified with estimates on cost and weight based on prior programs and technology projections. The partitioning of avionics application software is shown in Figures 1-3.2.4-19 and 1-3.2.4-20.

STV Design Issues. The STV avionics design process will be iterative from early concepts pre-phase A, phase B demonstrations, and phase C/D strawman. Fundamental to this process are top design issues that must be addressed throughout each phase. Figure 1-3.2.4-21 illustrates these issues,

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STV SPACE BASED FLIGHT ELEMENT CASES

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AVIONICS SUBSYSTEMS

Flight Element	Case 1 — single propulsion stage (single cab)	Case 1 (dual cabs)	Case 2 — two propulsion stages (single cab)	Case 2 (dual cabs)	Case 3 — three propulsion stages (single cab)	Case 3 (dual cabs)
TLI stage	CT $_{LL}$	CT ^{LO}	PT r ^{LO} e	PT r ^{LO}	PT e	PT e
Crew Cabs		CE $_{LL}$	CT $_{LL}$	CT ^{LO}	CT $_{LL}$	CT ^{LO}
				CE $_{LL}$		CE $_{LL}$
TEI stage					PT r ^{LO}	PT r ^{LO}
TLI, TEI, or decent/ascent stage	PL r $_{LL}$	PL r $_{LL}$	PL r $_{LL}$	PL r $_{LL}$	PL r $_{LL}$	PL r $_{LL}$

SB2-1.5S

Similar to HEI 90 day study

LL: Lunar Lander
LO: Lunar Orbiter
e: expendable
r: reusable

Figure 1-3.2.4-2.   Space-Based Flight Element Cases

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CASE 1 OVERVIEW
AVIONICS SUBSYSTEMS

OPTION 1

4 Missions

Cargo

P/A Module

• Autonomy Avionics
• Simple Cargo I/F

21 Missions

Cargo   Crew Cab   Cargo

P/A Module
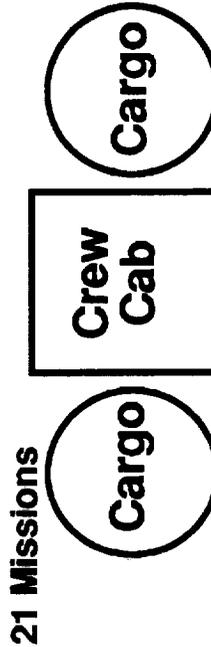
• Autonomy + Crew Avionics
• I/F Requires Reliable Overrides &
  High Data Rates
• Partitioning of Functions

OPTION 2

21 Missions

Cargo   Crew Cab Avionics   Cargo

Propulsion Platform
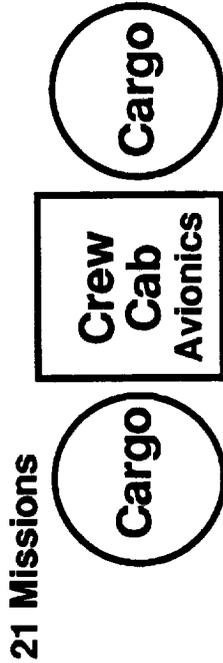
• Reduced Autonomy
• Common Avionics

4 Missions

Cargo

Avionics Pallet
Propulsion Platform
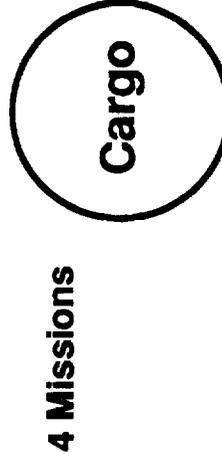
• Autonomy Avionics
• I/F Remains the Same
  for Cargo Mission

Figure 1-3.2.4-3. Case 1 Overview

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Figure 1-3.2.4.4. Avionics Partitioning

Avionics Trade Location

* AP for Cargo Mission

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Avionics Functions

AVIONICS SUBSYSTEMS

MI	Man-machine Interface (D&C, workstations, helmet, VA/VS,...)
MM	Mission Management
HM	Vehicle Heath Management/Monitor
NI	Navigation (inertial)
NR	Relative Navigation -Rendevous sensor
NL	-Landing sensor
ND	-Docking sensor
GC	Guidance and Control
PC	Propulsion Control
FC	Critical Fluids Control
AC	Attitude Control System
CT	Communication and Tracking
EP	Electrical Power, Control & Distribution
EC	Environmental Control and ECLSS
SM	Structures & Mechanisms (safety critical doors, gears,...)
PS	Payload Services & Accommodations
MU	Mission Unique (RMS, telerobotic,...)

Figure 1-3.2.4-5.   Avionics Functions

CREW MODULES (SINGLE CAB)

AVIONICS SUBSYSTEMS

AV Characteristic	Option 1, Cat 1	Option 1, Cat 2	Option 1, Cat 3
Crew Modules		Mid Functionality Redundancy Space Based 1-2 KW	

Option 1, Cat 2:
HM MI ND
4 CT
Xfer / Excursion Cab
SB2-1.5S

Option 1, Cat 3:
SM ND
AC
NR NI EC
MI CT EP
HM GC MM
R   R
1 CB
Ballistic Return
Xfer / Excursion Cab
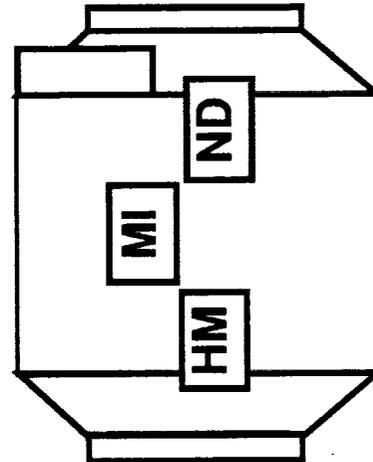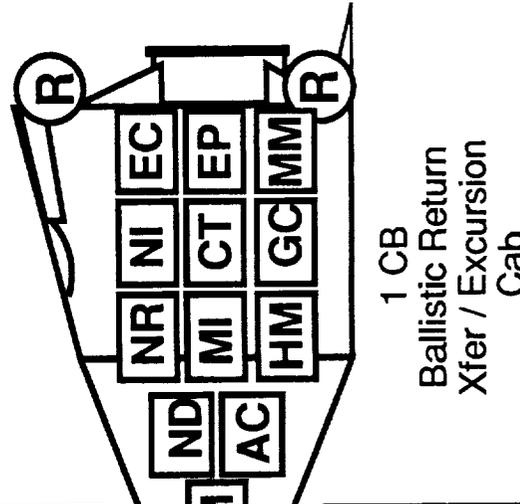
CT - Communication and Tracking
HM - Vehicle Health Management/Monitor
MI - Man-machine interface
ND - Relative Navigation -Docking sensor
NR - Relative Navigation -Rendevous sensor

Figure 1-3.2.4-6. *Crew Modules (Single Cab)*

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LANDER STAGES
AVIONICS SUBSYSTEMS

AV Characteristic	Option 1, Cat 1	Option 1, Cat 2	Option 1, Cat 3
Lander Stages	Low Functionality Redundancy 1-2 KW	High Functionality Redundancy 1-3 Mo Storage 4-6 KW	High Functionality Reusable Avionics Redundancy 1-3 Mo Storage 4-6 KW

5 PLD Expended on Lunar Surface

6 PL Manned Expended after TEI

7 PL, 8 PL Manned Reusable

SB2-1.5S

AC - Attitude Control System
CT - Communication and Tracking
EC - Environmental Control and ECLSS
EP - Electrical Power, Control & Distribution
FC - Critical Fluids Control
GC - Guidance and Control
HM - Vehicle Health Management/Monitor
MM - Mission Management
NI - Navigation (Inertial)
NL - Relative Navigation -Landing sensor
NR - Relative Navigation -Rendevous sensor
PC - Propulsion Control
PS - Payload Services
R - Reaction Control System
SM - Structures & Mechanisms (safety critical doors, gears,...)

Figure 1-3.2.4-7. Lander Stages

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BOEING

AEROBRAKES
AVIONICS SUBSYSTEMS

AV Characteristic	Option1, Cat 1	Option1, Cat 2	Option1, Cat 3
	Low Functionality	Mid Functionality 1-2 KW	High Functionality Redundancy 1-2 KW

STV

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Aerobrakes

Option1, Cat 1:
- 1A Manned — HM
- 3A Unmanned
- 2A Manned — HM SM
- 4A Unmanned

Option1, Cat 2:
- 6A Unmanned — AC CT HM / ND GC — R

Option1, Cat 3:
- 5A Manned — AC CT HM / ND GC — R R

SB2-1.5S

AC - Attitude Control System
CT - Communication and Tracking
GC - Guidance and Control
HM - Vehicle Health Management/Monitor
ND - Relative Navigation -Docking sensor
R - Reaction Control System

Figure 1-3.2.4-8. Aerobrakes

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DROP TANKS
AVIONICS SUBSYSTEMS

AV Characteristic	Option 1, Cat 1	Option 1, Cat 2	Option 1, Cat 3
Drop Tanks	Low Functionality Short Term Storage	Mid Functionality Long Term Storage	High Functionality Long Term Storage 1-2 KW

AC - Attitude Control System
CT - Communication and Tracking
FC - Critical Fluids Control
GC - Guidance and Control
HM - Vehicle Health Management/Monitor
ND - Relative Navigation -Docking sensor
R - Reaction Control System

Figure 1-3.2.4-9. Droptanks

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SB2-1.5S CONFIGURATION OPTION 1

AVIONICS SUBSYSTEMS

Figure 1-3.2.4-10. SB2-1.5S Configuration Option 1

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STV
CARGO CONFIGURATION OPTION 1
AVIONICS SUBSYSTEMS

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SSF Departure Configuration (1)

Coast Flight Configuration to Moon (2)

Lunar Lander (3b)

1T (Tank D1)

HM	FC

8PL (Prop 1+Tank C2)

CT			
EP	NI	GC	EC	MM	SM
AC	PS	FC	PC	HM	NL
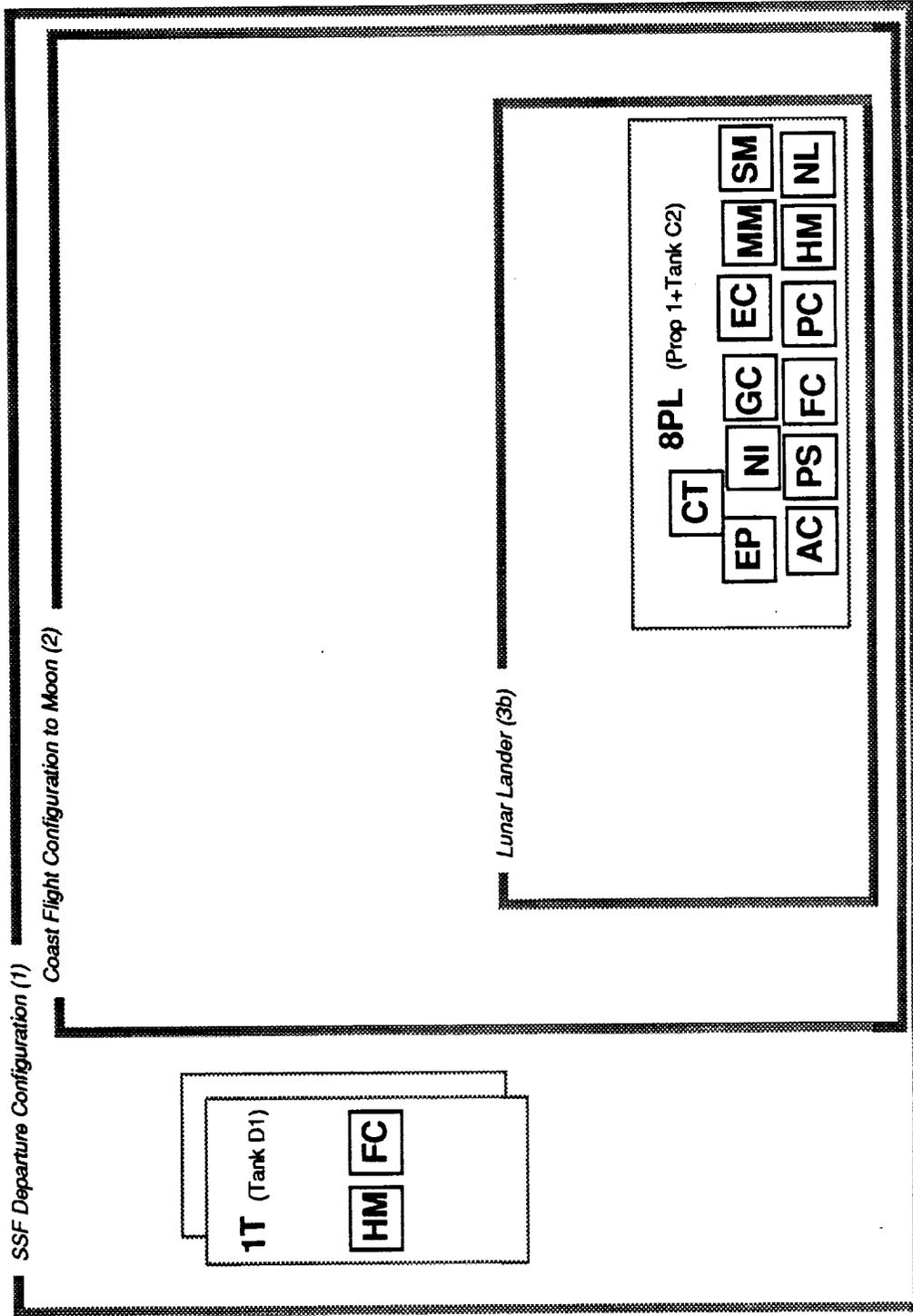
Figure 1-3.2.4-11. Cargo Configuration Option 1

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CREW MODULES (SINGLE CAB)

AVIONICS SUBSYSTEMS

	Option 2, Cat 1	Option 2, Cat 2	Option 2, Cat 3
AV Charcteristic		High Functionality Redundancy Space Based 3-5 KW	High Functionality Redundancy Ground Based 4-6 KW
Crew Modules			
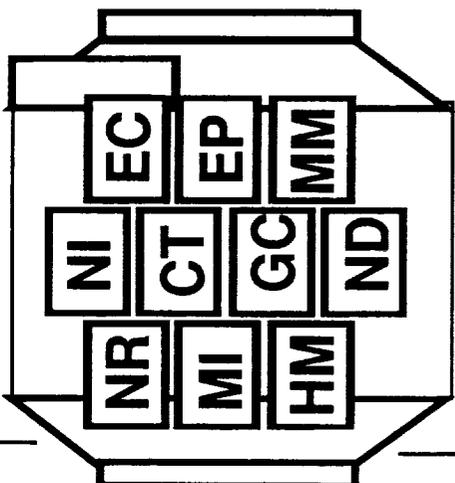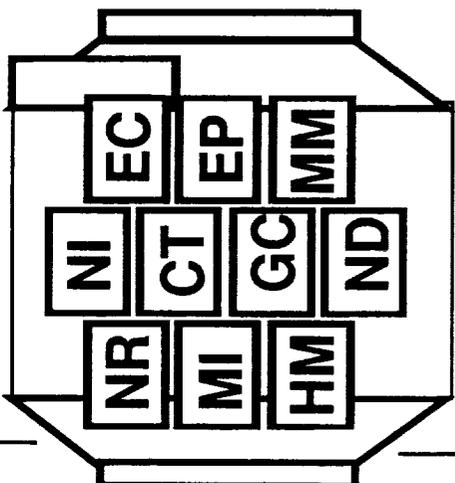
4 CT
Xfer / Excursion Cab

SB2-1.5S
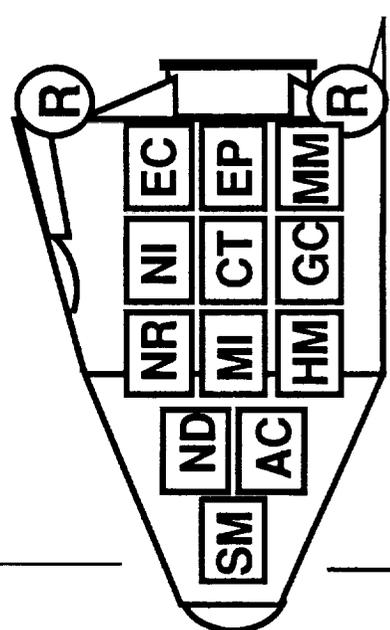
1 CB
Ballistic Return
Xfer / Excursion Cab

AC - Attitude Control System
CT - Communication and Tracking
EC - Environmental Control and ECLSS
EP - Electrical Power, Control & Distribution
GC - Guidance and Control
HM - Vehicle Health Management/Monitor
MI - Man-machine Interface
MM - Mission Management
ND - Relative Navigation -Docking sensor
NI - Navigation (Inertial)
NR - Relative Navigation -Rendevous sensor
R - Reaction Control System
SM - Structures & Mechanisms (safety critical doors, gears,...)

Figure 1-3.2.4-12. *Crew Modules (Single Cab)*

STV PITCH
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LANDER STAGES
AVIONICS SUBSYSTEMS

BOEING

STV

MSFC-*BOEING*

AV Charcteristic	Option 2, Cat 1	Option 2, Cat 2	Option 2, Cat 3
	Low Functionality Longe Term Storage 1-2 KW	High Functionality Redundancy 1-3 Mo Storage 2-3 KW	High Functionality Reusable Avionics Redundancy 1-3 Mo Storage 2-3 KW

Option 2, Cat 1 — 5 PLD Unmanned Expended on Lunar Surface

Option 2, Cat 2 / 6 PL Manned Expended after TEI

Option 2, Cat 3 — 7 PL, 8 PL Manned Reusable

SB2-1.5S

Lander Stages

AC - Attitude Control System
EP - Electrical Power, Control & Distribution
FC - Critical Fluids Control
HM - Vehicle Health Management/Monitor
NL - Relative Navioation -Landing sensor
PC - Propulsion Control
PS - Payload Services
R - Reaction Control System
SM - Structures & Mechanisms (safety critical doors, gears,...)

Figure 1-3.2.4-13. Land Stages

STV PITCH
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Figure 1-3.2.4-14. Aerobrakes

STV
MSFC-*BOEING*

BOEING

AEROBRAKES
AVIONICS SUBSYSTEMS

AV Charcteristic	Option 2, Cat 1	Option 2, Cat 2	Option 2, Cat 3
	Low Functionality	Mid Functionality 1-2 KW	High Functionality Redundancy 1-2 KW

Option 2, Cat 3

HM
GC
AC
CT
ND

5A
Manned

SB2-1.5S

Option 2, Cat 2

HM
GC
AC
CT
ND

6A
Unmanned

Option 2, Cat 1

HM

1A   3A
Manned   Unmanned

HM SM

2A   4A
Manned   Unmanned

Aerobrakes

AC - Attitude Control System
CT - Communication and Tracking
GC - Guidance and Control
HM - Vehicle Health Management/Monitor
ND - Relative Navigation –Docking sensor
R - Reaction Control System
SM - Structures & Mechanisms (safety critical doors, gears,...)

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Figure 1-3.2.4-15. Droptanks

DROP TANKS
AVIONICS SUBSYSTEMS

BOEING

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AV Charcteristic	Option2, Cat 1	Option2, Cat 2	Option2, Cat 3
Drop Tanks	Low Functionality Short Term Storage	Mid Functionality Long Term Storage	HighFunctionality Long Term Storage 1-2 KW

AC - Attitude Control System
CT - Communication and Tracking
FC - Critical Fluids Control
GC - Guidance and Control
HM - Vehicle Health Management/Monitor
ND - Relative Navigation -Docking sensor
R - Reaction Control System

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AVIONICS PALLET
AVIONICS SUBSYSTEMS

AV Charcteristic	Option 2, Cat 1	Option 2, Cat 2	Option 2, Cat 3
			High Functionality Fully Autonomous Land Redundancy 4-6 KW

Avionics Pallet

Option 2, Cat 3

NR CT EC
NI GC EP
HM ND MM
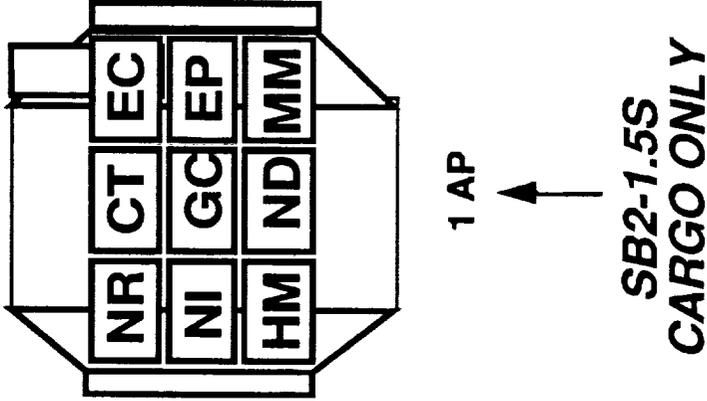
1 AP

SB2-1.5S CARGO ONLY

CT - Communication and Tracking
EC - Environmental Control and ECLSS
EP - Electrical Power, Control & Distribution
GC - Guidance and Control
HM - Vehicle Health Management/Monitor
MM - Mission Management
ND - Relative Navigation -Docking sensor
NI - Navigation (Inertial)
NR - Relative Navigation -Rendevous sensor

Figure 1-3.2.4-16. Avionics Pallet

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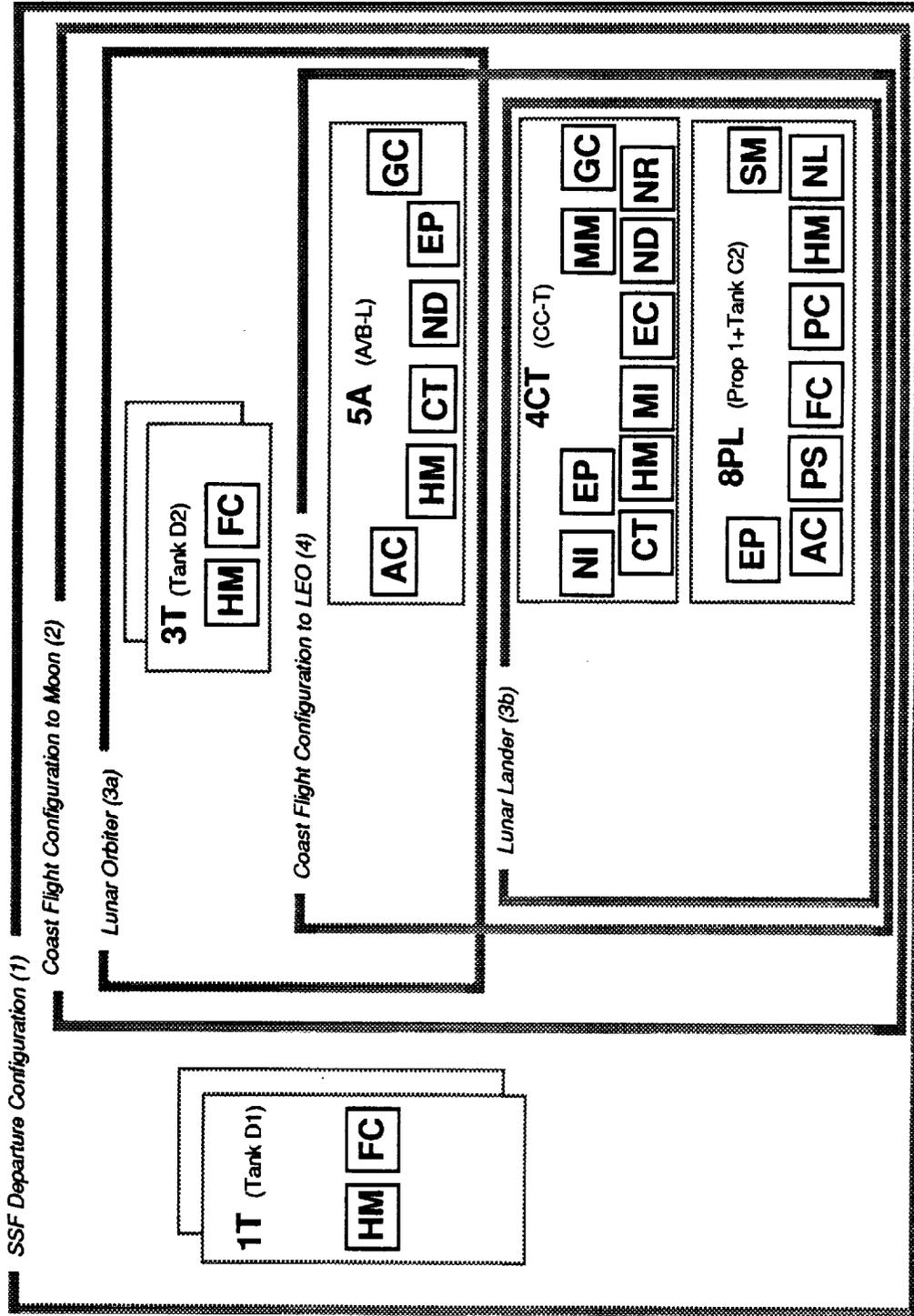
Figure 1-3.2.4-17.   SB2-1.5S Configuration Option 2

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CARGO CONFIGURATION OPTION 2
AVIONICS SUBSYSTEMS

SSF Departure Configuration (1)

Coast Flight Configuration to Moon (2)

1T (Tank D1)

HM	FC

Lunar Lander (3b)

1AP (CC-T)

NI	EP	HM	MM	ND	EC	GC
CT					NR	

8PL (Prop 1+Tank C2)

AC	PS	EP	FC	PC	HM	SM
						NL
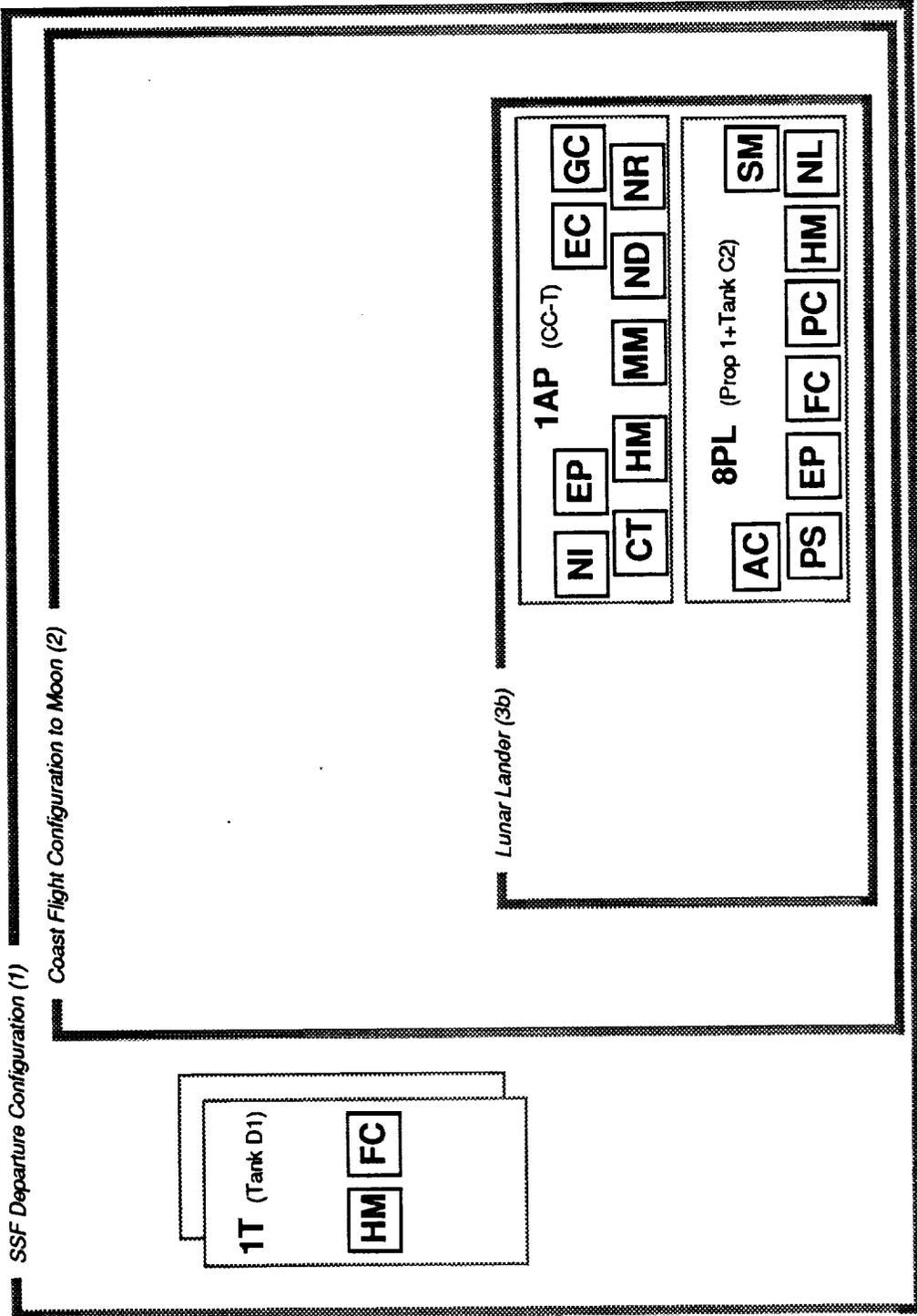
Figure 1-3.2.4-18. Cargo Configuration Option 2

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STV HARDWARE/SOFTWARE PARTITIONING

Avionics Subsystems

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	Function	Criticality	Hardware Complexity (L,M,H)	Software Complexity (L,M,H)
System Management	MI Man-Machine I/F	F/S	M D&C, work station, helmet/gloves voice actuation/synthesis	H RealTime Decisions (AI/ Expert Systems)
	MM Mission Management	-	L DMS*	H — Damage Assessment
	HM Health Management	-	L/H DMS, recorders, sensors	H Failure Analysis (AI)
Navigation	NI Navigation - Inertial	F	H IMU, Platform, Chronometer, Optical sensors	H Orbit determination with observables
	NR - Rendezvous	F	H Radar, Pedestal, signal processor	H Orbit determination with local observables
	NL - Landing	F	H Sensors, Laser, Radar, signal processor	H Acquisition + tracking, Autonomous ops.
	ND - Dock	F	M Laser, Targets	M Autonomous operations
Guidance & Control	GC Guidance + Control	F	M/H DMS	L/H Docking, landing, aerocapture
	AC Attitude Control	F	M Controllers	L/M
	PC Prop Control	S	M Driver (RJ, OMS), DMS, sensors	M
	FC Fluid Control	S	M Valve Drivers, DMS, sensors	L
Comm	CT Communication & Tracking	-	M Transponders, switches, controllers Transceivers, antennas	M autonomous acquisition
Power	EP Electrical Power, control, distribution	F	M Sources, distribution and controllers	M autonomous operation
Vehicle	EC Environmental Control/ECLSS	-	L Consumable monitoring, sensors,	L
	SM Structures & Mechanisms	S	L controllers DMS, controllers, sensors	M
Mission	PS P/L Services	M	L DMS	L/M Science Analysis + Replanning
	MU Mission Unique	M	M Robotics/EVA control	?

F= Flight Critical, S= Safety Critical, M= Mission Critical

*DMS processors, data buses, memory, mux/demux I/O

Figure 1-3.2.4-19. Hardware and Software Partitioning

STV

FLIGHT SOFTWARE
Avionics Subsystems

Source Lines of Code

The STV software will not have overlays. Many of the software functions will be used for more than one phase of operation.

Category	SB2-1.5S SLOC (K)		Combined total
	Piloted	Cargo	
Operating System	15	15	15
Mission Management	8	16	16
Vehicle Utility	1	1	1
Guidance	18	27	27
Navigation	8	8	8
Flight Control	18	12	18
Controls & Displays	25	0	25
Communication & Telemetry	3	6	6
Environment Control	1	1	1
Payload Services	2	2	2
Vehicle Health Management/ Monitor	30	45	45
	129	133	164

All estimates are for Ada (with restrictions); data bases and growth not included. Uncertainty on SLOC is +50% -20% based upon multiple factors including software development methodology.

Figure 1-3.2.4-20. Flight Software

STV Avionics Top Level Design Issues

Avionics Subsystem Splinter

STV

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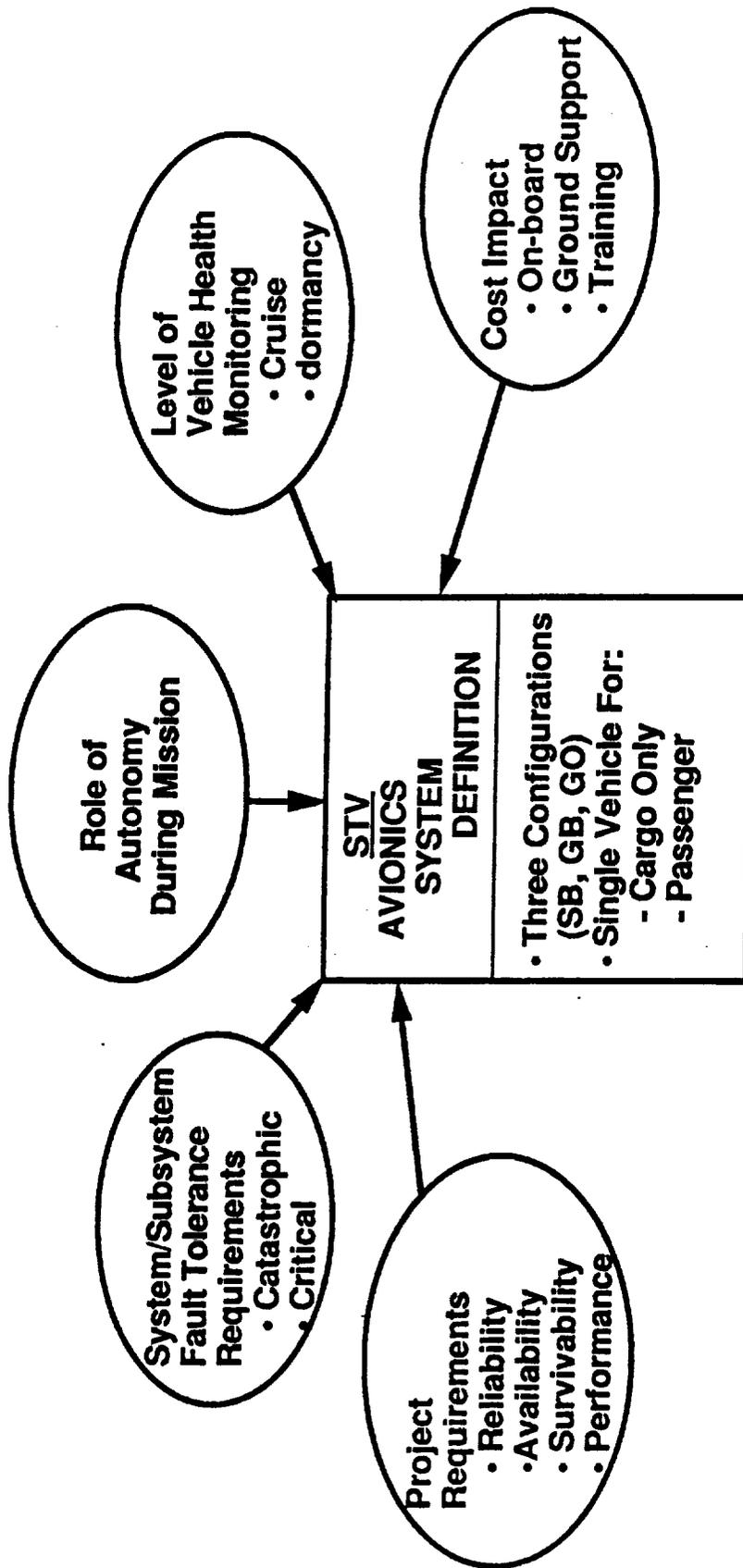
Figure 1-3.2.4-21. STV Avionics Top-Level Design Issues

which are interrelated. Autonomy, health management/monitoring and fault tolerance share common ground among sensors, data networks, and criticality. Design requirements and implementation relate to cost. Figure 1-3.2.4-22 scales low onboard hardware vehicle cost (but, high operational cost) to highly autonomous system. Where the lunar vehicle avionics cost will be depends on judicious choice and interrelation between the top design issues.

Key Technologies. Figure 1-3.2.4-23 compares avionics technologies with driving requirements. The technologies are divided into three elements of lowest (device) network devices that connect components together and highest (subsystem) network devices.

1-3.2.5   Thermal Rejection

Excess heat, produced by both equipment and metabolic activity, is removed by the environmental control and life support system (ECLSS) and rejected to the surrounding environment. Thermal-rejection strategies must account for various phases of the mission, which occur in different surroundings. These phases vary between the configurations but include pre-launch, ascent, operations at the SSF, in-transit operation, lunar surface operation, aeromaneuver, descent, and post-landing. Reducing the amount energy consumed on the vehicle directly reduces the heat load that needs to be removed. In particular, the use of highly integrated avionics can significantly decrease power consumption and hence thermal output. For the technology availability level assumed for this study, both STV configurations were assumed to require ≈ 10 kW of heat rejection.

During the pre-launch phase, the ground-based configuration will require external thermal control, which would be provided at the launch site or servicing facility. The ground support equipment (GSE) would provide coolant (most likely cold Freon) through flyaway disconnects to a heat exchanger in the ECLSS loop (Figure 1-3.2.5-1). This conserves STV expendables usage while waiting in a powered-up configuration for an indefinite launch hold. This strategy also eliminates the possibility for unfavorable interactions between radiated heat or vented vapors and the launch vehicle or launch facilities.

Installed Hardware Cost* Spectrum

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Avionics Subsystem Splinter

"Astro Wagon" (Requirement Scrub)
- Piloting Skills
- Ground Support/Updates
- Man-Rated
- Reusable

"SB Reference (Space Based)
- Modular
- Autonomous G & C
- Minimum Crew Interaction
- Vehicle Health Monitoring
- Autonomous Test and Checkout
- Man-Rated
- Reusable

"Deconditioned Crew" (No Interaction)
- Multi-Mission
- Highly Modular
- Adaptable
- Fully Autonomous Operations
- Self Healing
- Fly With Faults

On-Board Hardware Cost

$

$$$
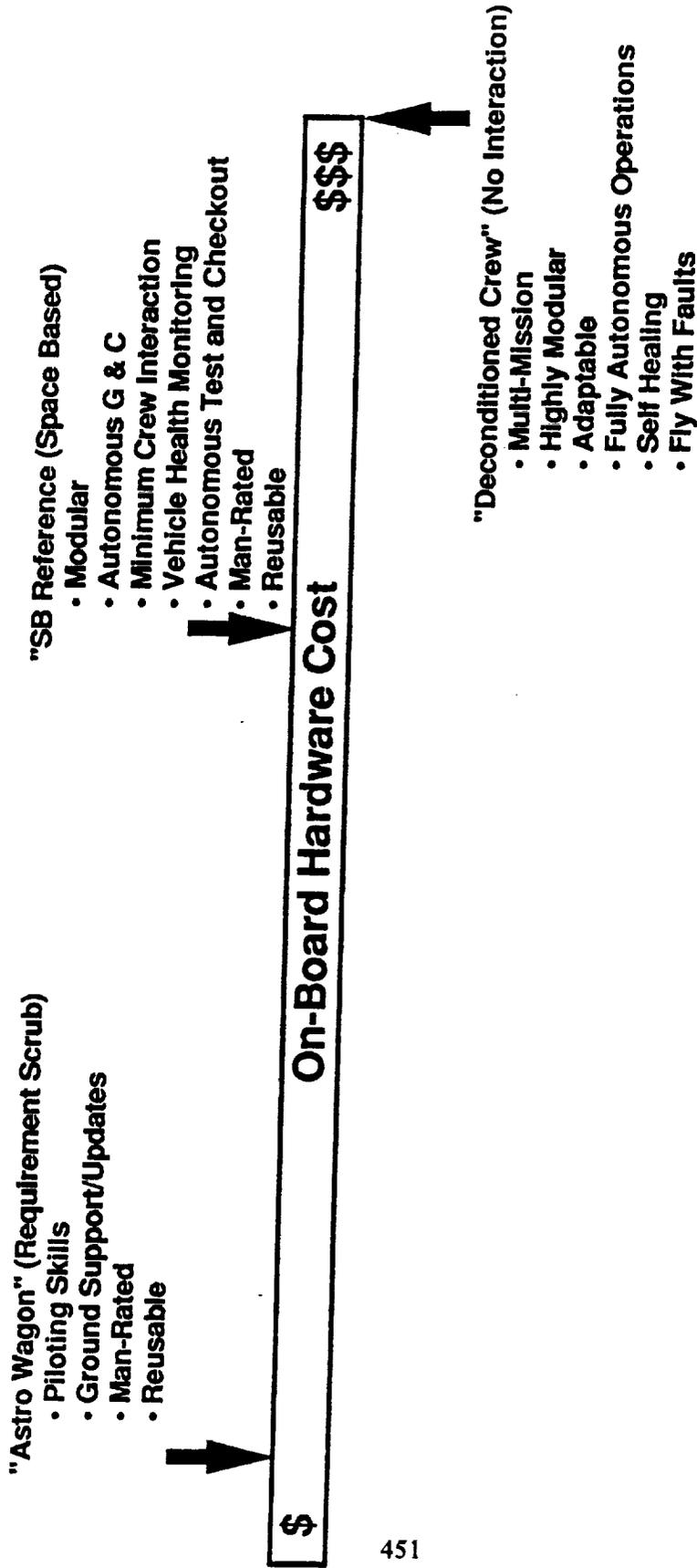
* NOTE: Does NOT include attendant costs such as, Ground Operations, Space Station Freedom
Operations, Crew Training, Mission Planning, Software Development.

Figure 1-3.2.4-22. Installed Hardware Cost Spectrum

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STV
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KEY TECHNOLOGIES
Avionics Subsystem Splinter

AVIONICS TECHNOLOGY AREAS & LEVELS	Reduce Mission Operation Costs	Reduce Ground/SSF Operation Costs	Self-Inspection/ Diagnostics	Increased Number of Transducers	Transducer Data Rate	Robust Flight Control	High Performance Processing	Module Level Fault Detection	Minimize Avionics Implementation Cost
DEVICE									
1 Application Specific Integrated Circuits			X				X	X	X
2 Fiber Optic Sensors			X				X	X	X
3 Neural Networks			X	X			X	X	X
4 Navigation Instruments						X			X
NETWORK									
5 Digital Data Buses				X	X	X	X		
6 Sensor Networks			X	X	X	X			X
7 Standard Interfaces		X	X	X					X
SUBSYSTEM									
8 Autonomous Navigation Subsystems	X	X				X			
9 Autonomous Guidance and Control	X					X			X
10 Vehicle Health Monitoring	X	X	X	X					
11 Expert Systems	X	X	X	X				X	
12 Fault Tolerant Avionics & Avioptics	X	X	X					X	
13 Communication and Tracking	X	X						X	

Figure 1-3.2.4-23. Key Technologies
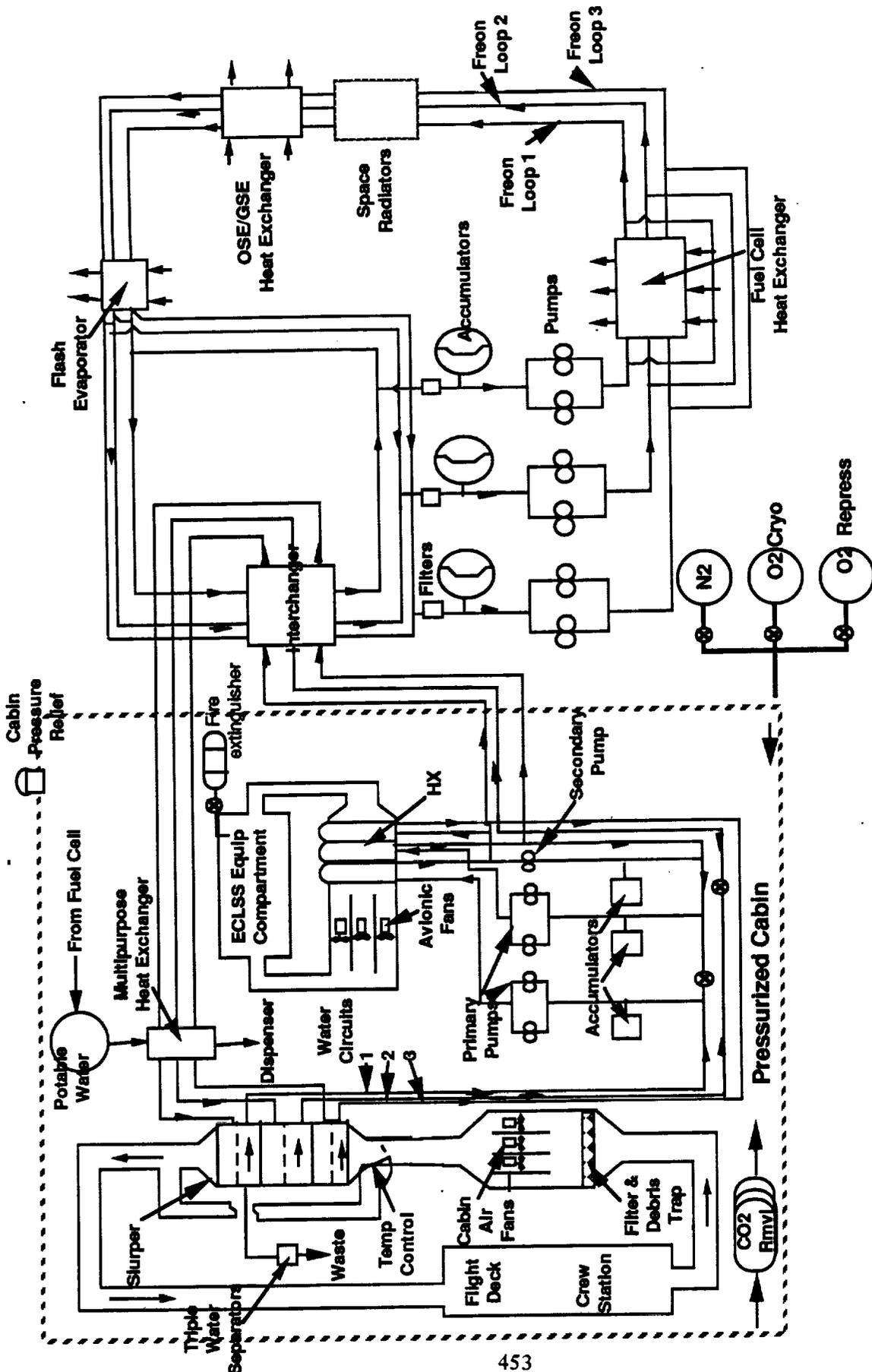
Figure 1-3.2.5-1. Baseline ECLSS Configuration

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which are interrelated. Autonomy, health management/monitoring and fault tolerance share common ground among sensors, data networks, and criticality. Design requirements and implementation relate to cost. Figure 1-3.2.4-15 scales low onboard hardware vehicle cost (but, high operational cost) to highly autonomous system. Where the lunar vehicle avionics cost will be depends on judicious choice and interrelation between the top design issues.

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During the pre-launch phase, the ground-based configuration will require external thermal control, which would be provided at the launch site or servicing facility. The ground support equipment (GSE) would provide coolant (most likely cold Freon) through flyaway disconnects to a heat exchanger in the ECLSS loop (Figure 1-3.2.5-1). This conserves STV expendables usage while waiting in a powered-up configuration for an indefinite launch hold. This strategy also eliminates the possibility for unfavorable interactions between radiated heat or vented vapors and the launch vehicle or launch facilities.

As the ground-based vehicle is launched and ascends to orbit (a period of time lasting up to a few hours), the vehicle is subjected to aerodynamic and aerothermal forces that prevent the use of some heat-rejection concepts. Deployable devices, for example, would be unacceptably heavy if designed to be robust enough to tolerate dynamic pressure loads. Passive thermal control, or heatsinks, could be used and are close to the present state of the art. Analysis shows that a reasonable passive concept could provide thermal control for a period of time much less than the length of the mission; if such a system were used to its capacity, some other form of thermal rejection would still be required to cover the rest of the mission.

Another type of device, a flash evaporator, has been used successfully on the space shuttle. Previous water evaporator experience consists of wick-feed boilers of the type used on Mercury, Gemini, and the Apollo command module and porous plate type sublimators used on the lunar module, Apollo space suits, and the Saturn V. All these devices, while meeting reliability expectations, had response, heat load range, and life limitations that led to the shuttle-type flash evaporator development. Flash evaporation involves spraying water on the walls of a chamber heated by the coolant loop. The chamber is maintained at a saturation pressure low enough for the water to evaporate at a temperature below the desired coolant loop outlet temperature. The generated steam is vented overboard through a sonic nozzle. Water is the preferred fluid for several reasons. First, water has the best latent heat of vaporization per weight per volume of any candidate fluid and therefore minimizes the weight and volume penalty on the vehicle. Secondly, by drawing excess water from the fuel cells (a byproduct of power production), a synergistic reduction in total vehicle mass is realized. Thirdly, water is non-toxic and is relatively benign when vented to space in the vicinity of adjacent spacecraft.

The selected ascent thermal control uses a water flash evaporator. To reach the necessary low-operating pressure, the vehicle must be above 140,000 feet. During a typical boosted trajectory, it takes about 2 minutes to reach this height and sufficient thermal inertia is assumed to passively control the thermal environment until the flash evaporator can be activated.

For both vehicles, while operation in a vacuum environment (i.e., in space and on the lunar surface), several options for heat rejection are available. A flash evaporator would function adequately, but the additional consumable weight (water) becomes considerable for longer missions. Also, the outgassed steam, although benign compared to other fluids, can negatively affect other spacecraft. In fact, current SSF operating rules would probably not permit this venting while in the vicinity of the station.

One of the significant sources of heat is the heat given off by the vehicle's avionics. On several missions (e.g., cargo missions) the STV will fly without a crew on board. In this configuration the vehicle will also fly without a crew module and the mission avionics are essentially unchanged. This would done to keep from developing separate avionics suites for the piloted and unpiloted missions.

If the avionics are aircooled in the piloted configuration, these avionics will have to be aircooled on the unpiloted missions. This will introduce the added complexity on the unpiloted missions of a pressurized avionics pallet and some means of cooling this pallet.

If the avionics are vacuum qualified and can be radiatively cooled to avoid carrying the pressurized pallet on the unpiloted vehicle, view factors for those avionics become a integration constraint. In the piloted mission, these integration constraints of giving the avionics reasonable view factors conflict with the requirement of protecting those same avionics for the reentry environment.

The other category of thermal control schemes radiate waste heat to the low temperature of black space. There have been many vehicles that have used radiators, from simple conductive cooling fins to deployable panels (such as the STS orbiter). Radiator designs are relatively simple, reliable, and robust. To maximize performance, a high-reflectivity high-emittance coating is required (such as white paint). Using a radiator introduces integration complexities that the boilers avoid. The space-based configuration has the aerobrake area to use as a radiator. Although the aerobrake itself cannot be used as a radiator (the tiles on the forward face of the aerobrake are much to efficient an insulator) an

expandable radiator could be placed atop it (and jettisoned prior to the aeropass) or a reusable radiator could be extended from behind the aerobrake. In this case, before the aeropass, the radiator would be retracted behind the brake. In the event of a failure of the retraction mechanism, the radiator would be jettisoned. On the ground-based vehicle, the limited size of the return vehicle greatly limits the size of a reusable radiator. Expendable radiator concepts (discarded at reentry) would alleviate these safety concerns and negate the vehicle size and landing weight issues associated with the large STV radiator size. In addition to metallic panel-type radiators, one could use an inflatable device using ECLSS air as a working fluid. Such a system promises to be extremely low cost, have low stowed volume, and be very lightweight but is as yet an unproven concept though worthy of exploration.

The descent phase for the ground-based vehicle is similar to the ascent phase in that aerodynamic and aerothermal forces dominate external surface design. A significant difference though, is the time spent in the atmosphere (up to 0.5 hours on descent). Water flash evaporators will not function below about 140,000 feet, and heat loads are large for passive systems of reasonable size. On the space shuttle, an ammonia boiler is used to provide cooling for the last 10 minutes of flight and for about 15 minutes after landing until GSE can be connected. Ammonia, while having a latent heat of evaporation only half that of water, is the next most efficient coolant by weight and volume. Alternative fluids have been explored but are either inefficient and require large storage volumes or are environmentally hazardous to release (such as chloroflourines). Why not use the same ammonia boiler for ascent? Ammonia is toxic and can be stored sealed until the end of the mission to minimize potential hazards. This ammonia flash evaporator system has been selected for the descent phase.

In the post-landing phase, there is still a requirement to reject waste heat. Some subsystems (e.g., communications and ECLSS) may be kept on for hours. Additionally, depending on the vehicle's thermal protection system concept, a significant amount of heat has been absorbed on reentry and will reradiate after landing, even if all systems are shut down. The capability for the structure and secondary structures to safely absorb this heating without auxiliary GSE remains to be determined.

1-3.2.6   Human Habitation

The crew compartments of the ground-based and space-based configurations are two very different configurations. The most obvious difference is the outer mold line. The reason for this is that, unlike the space-based vehicle, the crew module returns to the ground. This ground return dictates that the outer mold line of the vehicle is capable of reentry and hypersonic flight. Once this reentry configuration is determined, the crew compartment and internal arrangement are fit into that shape. Obviously this is the iterative process that must continue until the outsides is larger than the inside.

Another major difference between the two configurations is caused by the difference in the mission duration. The main difference in mission duration is due to the difference in the abort scenarios. For the ground-based vehicles, during an abort, the vehicle has no constraints placed on its Earth return because it can land just about anywhere. This is not true for the space-based vehicle. In the space-based vehicle, the abort trajectory has the additional complexity of having to return to the Space Station. As previously discussed, after the STV leaves the Space Station the orbit of the station will precess away from the orbit of the STV. Should the mission be aborted at this point, the vehicle is unable to return to the Space Station. In such an event, the crew has two options that they can pursue.

The first option is for the vehicle to continue on the free-return trajectory, bypass the Moon, and return to Earth. Upon reaching the Earth, the STV would aerobrake and go into a circular orbit at the Space Station altitude. Having achieved a stable orbit, the STV now needs to be rescued, either by a vehicle from Earth (e.g., a shuttle or PLS) or from the Space Station (e.g., a space tug) with the latter being more likely.

The second option for the space-based vehicle in an abort is to not bypass the Moon. In this option, the crew would go into a lunar orbit and wait. Wait for the Space Station's orbit to precess back into a position where the STV can then rendezvous.

Both of these options significantly increase the amount of time the crew will be in the crew module. This will impact the amount of consumables required and the habitable volume of the crew compartment .

Crew Compartment. The first impact that the mission duration has on the STV crew compartment is shown in Figure 1-3.2.6-1. Using the historical examples and the guidelines of NASA-STD-3000, a crew volume can be estimated with consideration given to whether the volume will be luxurious, optimal, or confining.

It is important to remember that while considering the layout of the crew compartment, the orientation of the "g" vector must also be considered. It is this acceleration that gives an "up" and "down" to a compartment. In the STV space-based configuration, it becomes obvious that the g force on the Moon is 180 degrees from the g forces that will be experienced by the crew during reentry. Setting up two crew stations (one to control the vehicle during landing and one to control the vehicle during aeromaneuver) to handle this change in orientation is unwieldy and awkward.

The ground-based vehicle has the added complexity of being launched from the Earth. This, in the worst case, adds a third crew orientation to handle this acceleration and if not handled properly could introduce a third crew station to control the vehicle during launch.

ECLSS. During the ECLSS, subsystem analysis trades and comparisons were performed to determine if different ECLSS approaches were warranted for the three STV configurations and to determine the level of loop closure in the ECLSS. As part of this analysis four different types of crew module were considered during this phase (Figure 1-3.2.6-2).

Shown in Figure 1-3.2.6-3 the configurations considered in the architecture phase of the study all had mission durations of greater than 150 days (with the exception of the ground launch expendable missions). It is important to remember that the crew is not aboard the vehicle during this entire period. Although the actual time that the crew is aboard is shown in Figure 1-3.2.6-4, the vehicle may require some ECLSS during the entire mission time to provide
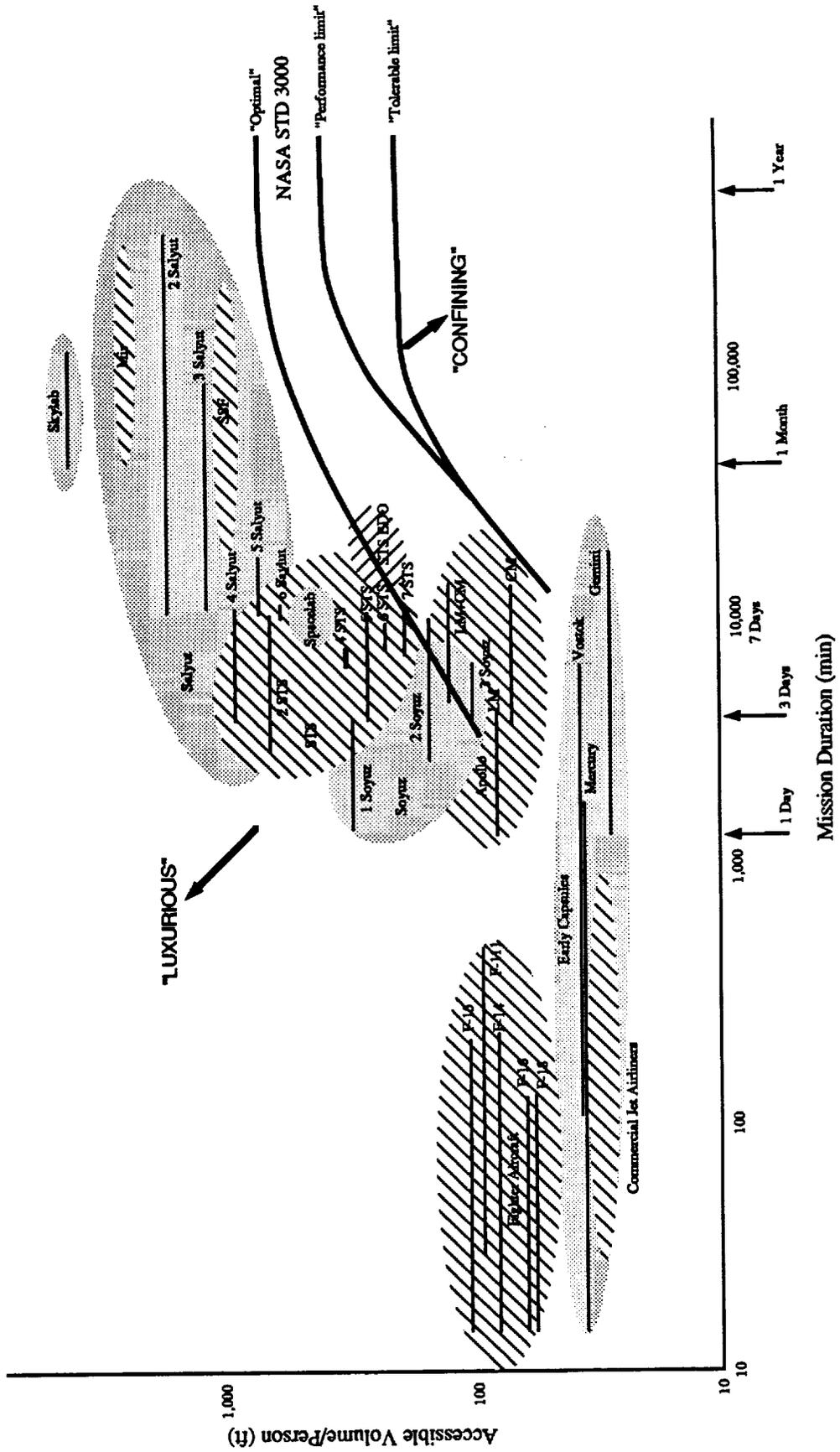
Figure 1-3.2.6-1. Available Volume Per Person Versus Mission Duration

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CREW MODULE SUBTYPES

System/Subsystem Concepts

- Transfer cabs
  - 4 person crew
  - Module travels from Earth to LLO

- Excursion cabs
  - 4 person crew
  - Module descends from LLO to Lunar surface and back

- Transfer/excursion cabs
  - One module performs both Earth to LLO and LLO to lunar surface

- Hybrid cabs
  - Full cab travels from Earth to LLO
  - Partial cab descends from LLO to lunar surface

Figure 1-3.2.6-2. Crew Module Subtypes

STV

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TOTAL MISSION DURATION
System/Subsystem Concepts
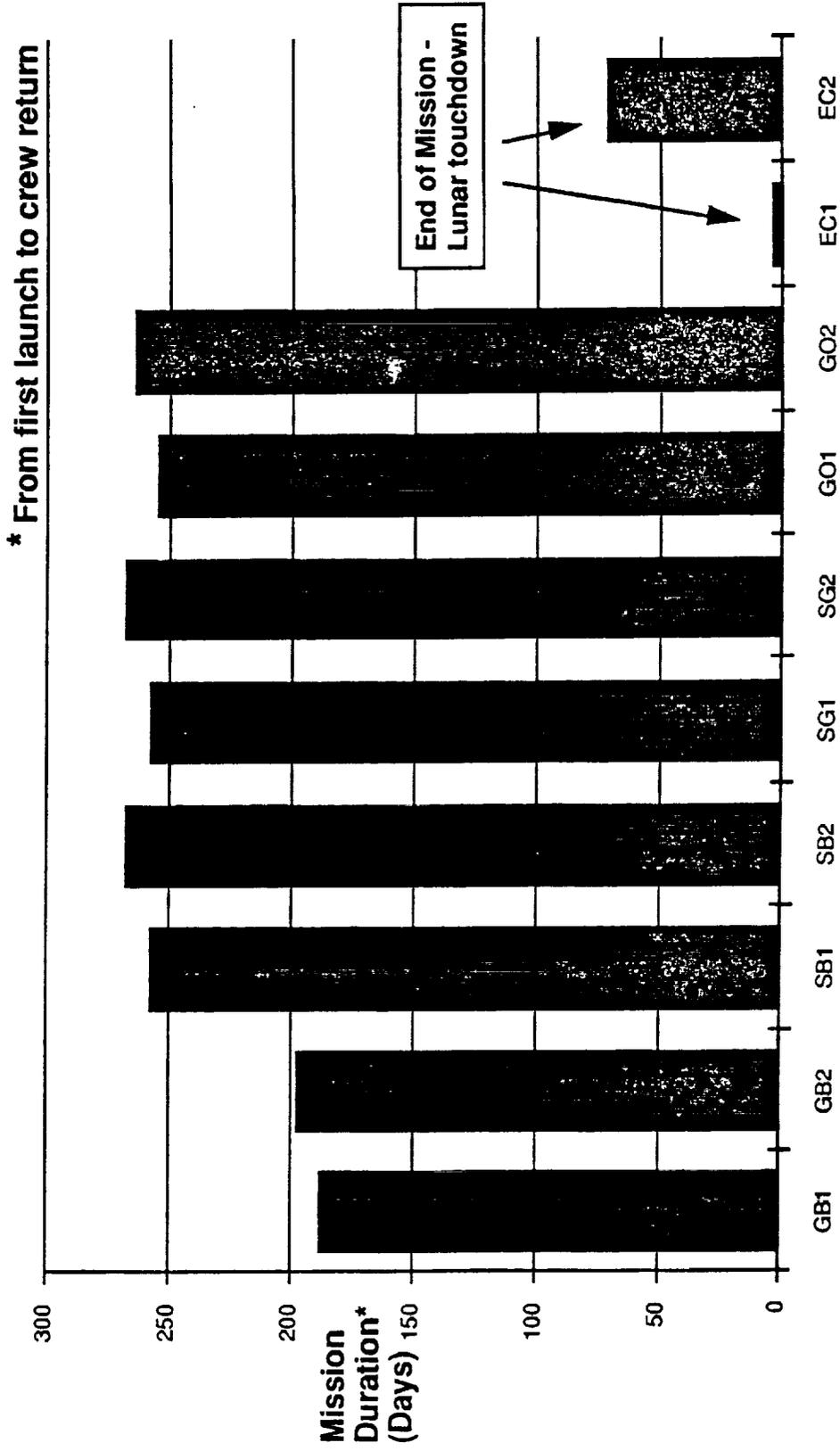
* From first launch to crew return

Figure 1-3.2.6-3. Total Mission Duration

CREW MODULE SUPPORT DURATION

System/Subsystem Concepts

A - Single or hybrid mode excursion module
B - Dual or hybrid mode transit module
C - Dual mode excursion module

Assumes crew remains in crew module during on orbit assembly

Additional time for abort

Crew Module Stay Time (Days)
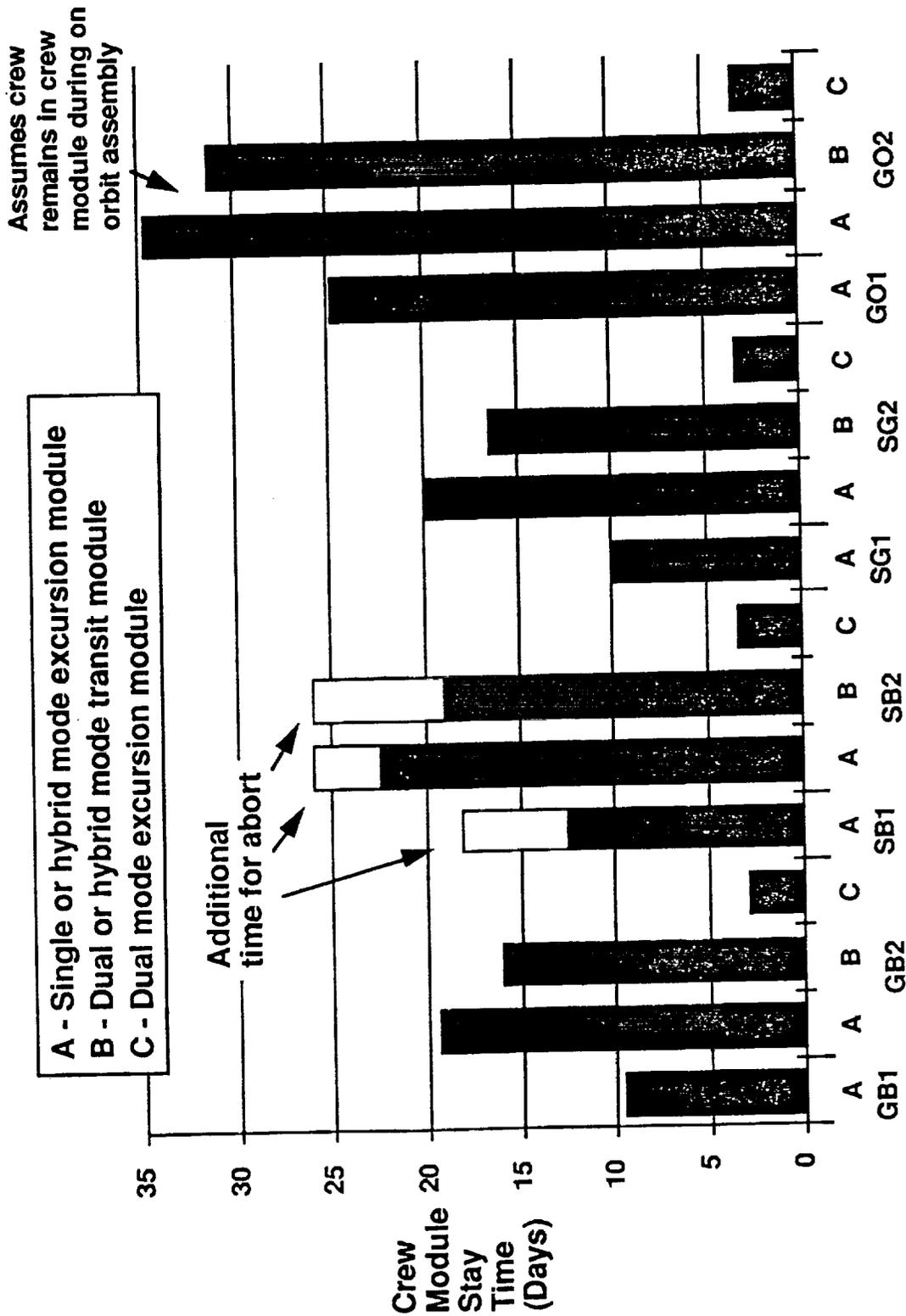
Figure 1-3.2.6-4. Crew Module Support Duration

subsystem (such as avionics) cooling. As an alternative to this lunar habitat, the configuration impacts of no lunar surface support were considered. These impacts are shown in Figure 1-3.2.6-5.

STV Configuration Impact on ECLSS Design Drivers. The first task in the determination of the life support system design for the STV was to look at the three principle STV configurations (ground based and single launch, space based, and ground based and assembled in LEO) to see what were the basic ECLSS design drivers and if there were appreciable design driver differences among the three configurations. Figure 1-3.2.6-6 summarizes the findings, and a more detailed discussion follows.

The principle design drivers for the selection of a life support system approach are crew size and stay time. Crew size determines the mass, power, and volume of the equipment. Stay time determines the amount of consumables, expendables, and spares that must be taken along to keep both the crew and system operating efficiently. A combination of crew size and stay time (in person-days) can be used to determine when it becomes advantageous, logistically, to use regenerative versus open-loop technology in the life support system. There are, of course, other design factors such as safety, reliability, and maintainability, but these tend to affect equipment selection and layout after the degree of closure has already been determined.

According to Figure 1-3.2.6-4, all three STV configurations have the same crew size (four) and close to the same mission times (5.8 to 6.4 days). This similarity would indicate a common life support system approach. A common approach is also supported by the requirement for five reuses in each configuration and the same number of crew modules (one). There are differences, however, in the storage and refurbishment locations (i.e., space based versus ground based) that drive the reliability and maintainability approaches to equipment selection and layout.

ECLSS Approach Comparison - Open Versus Closed Loop. The determination of the degree of loop closure in the ECLSS was made on the basis of ECLSS weight. Figure 1-3.2.6-7 shows the affect of ECLSS loop closure on system weight based on increasing mission time. This comparison is

STV
Space Transfer Vehicle

Payload Performance Without PSS Support
Payload/PSS Interfaces
MSFC- *BOEING*

ASSUMPTIONS :

- NO LUNAR SERVICES AVAILABLE (PWR, HAB, ETC.)
- BASELINE LEV SYSTEMS (NO CHANGE IN SYSTEMS OPERATIONS)
- NO VOLUME INCREASE FOR CREW
- POWER SYSTEM : SOLAR ARRAYS 14 DAYS, FUEL CELLS 14 DAYS
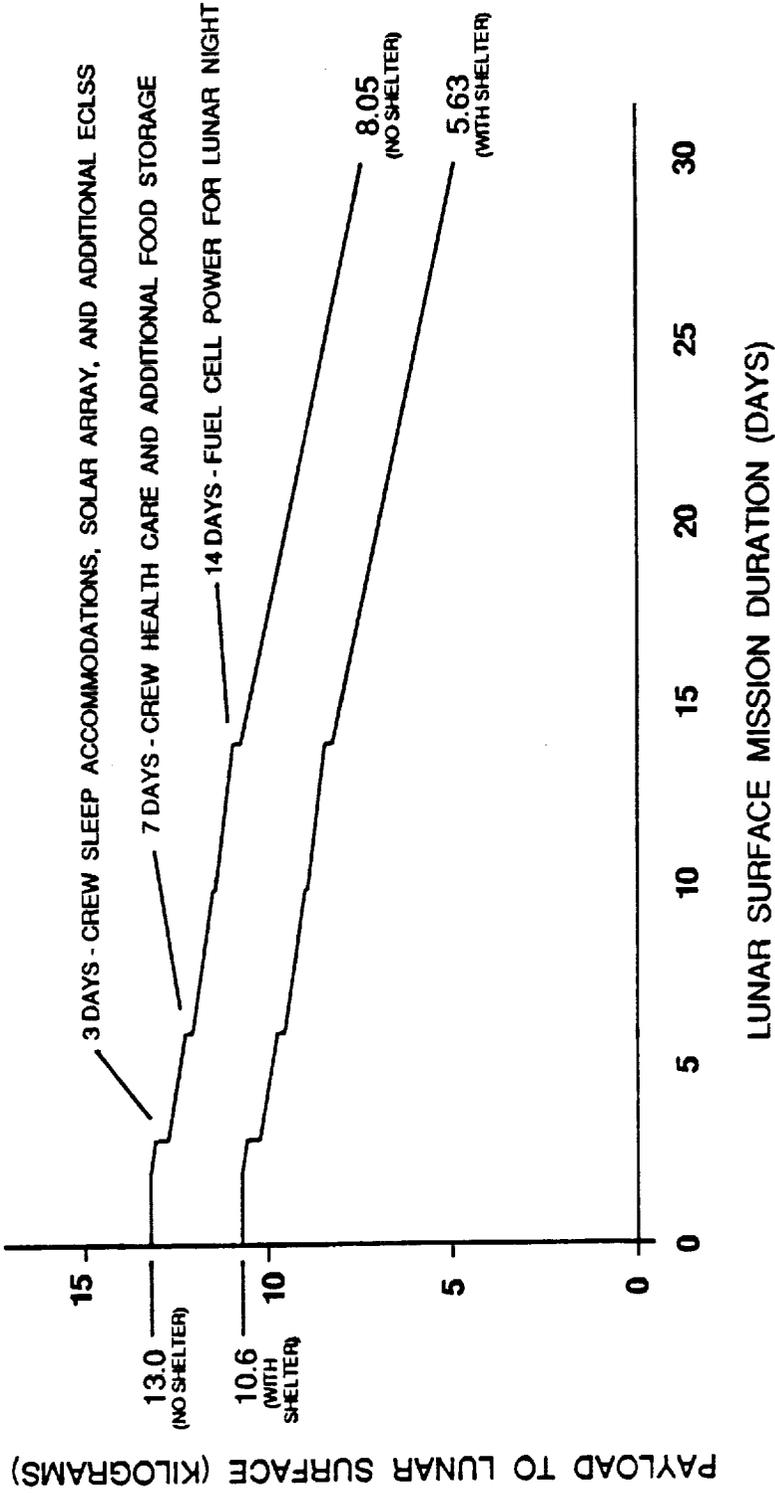
Figure 1-3.2.6-5. Payload Performance Without PSS Support

STV Configuration	Servicing Base	Reuse	Crew Modules	Idle Periods Place	Time	Avionics Location	Crew Size	Mission Time
GB-1 Ground Based Single launch	KSC	5 (5 yr) except tanks	1	Base w/o sprt	28-d	external	4	5.8-d (up to 48 h support on surface)
				Base w/sprt	150-d			
SB-1 Space Based (baseline)	SSF	5 (5 yr)	1	LEO storage	52-d	external	4	6.4-d (up to 48 h support on surface)
				LEO prep	15.3-d			
				Base w/o sprt	28-d			
				Base w/sprt	150-d			
GO-1 Ground based - LEO assembly	KSC	5 (5 yr) except tanks	1	LEO storage	52-d	external	4	5.8-d (up to 48 h support on surface)
				LEO prep	15.3-d			
				Base w/o sprt	28-d			
				Base w/sprt	150-d			

Figure 1-3.2.6-6. STV Configuration Impact on ECLSS Design Drivers

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Figure 1-3.2.6-7. ECLSS Loop Closure Comparison
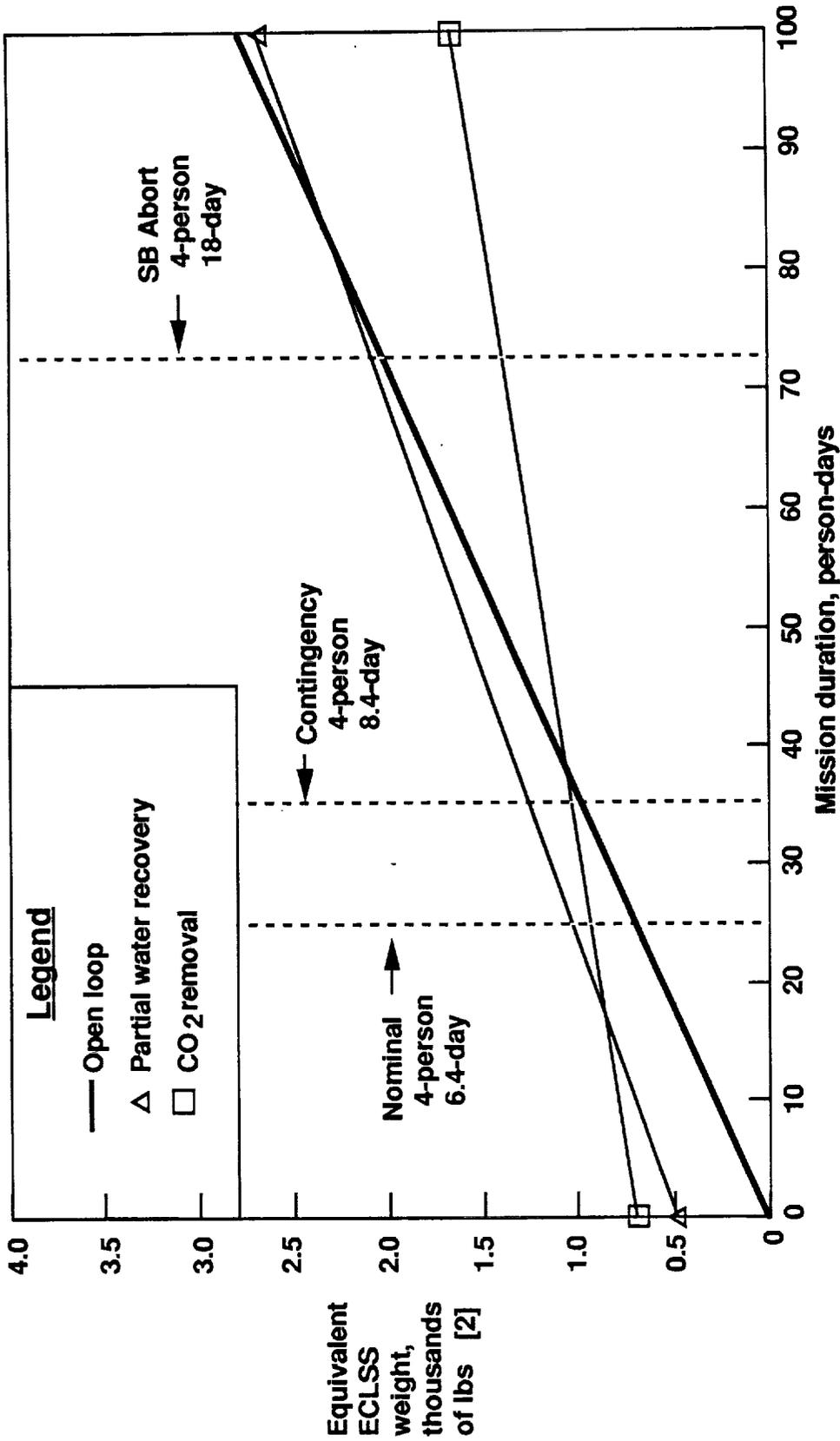
[1] Quattrone, P.D., "Extended Mission Life Support Systems", June, 1981

[2] Single hardware weight plus spares and expendables. No hardware redundancy included.

STV Rpt_Fig. 6 2/5/91

based on single hardware weight plus required spares and expendables plus consumables. No hardware redundancy is included.

It can be seen from this figure that both the nominal and contingency STV missions fall within the realm of the open-loop approach. Only the contingency abort mission makes regenerative carbon dioxide removal technology seem favorable. The majority of the weight difference between the open-loop and the CO_2 removal systems is the extra consumables (i.e., the oxygen consumed by the crew). Remember, however, that in this worst case abort, the vehicle has not made a lunar descent and ascent. The crew can tap into this excess propellant to provide both the propellant for the fuel cells giving the crew the power necessary for the longer mission and providing any additional breathing gases for the longer mission. In addition to this, when STV required fault tolerance is considered in the system closure comparison, an open-loop system using LiOH for carbon dioxide absorption is lighter and smaller, consumes less energy, dissipates less heat, and is more reliable than a regenerable four bed molecular sieve system with triple redundancy (i.e., three pieces of equipment).

Partial water recovery (i.e., recovery of cabin humidity condensate for drinking) only breaks even against an open-loop system when regenerative carbon dioxide removal is already in place. Another factor tending to deselect the water recovery option is the ready availability of adequate drinking water from the STV vehicle fuel cell power system.

Fire Detection and Suppression. Fire poses a significant hazard in the confined space of the pressurized compartments of a STV. Careful selection of materials, insulation, and isolation will reduce the risk of fire damage. In the event of even the smallest fire or smoldering, the risk of inhalation of toxic combustion plastics (such as wire insulation) could quickly harm the crew. Appropriate warnings and actions are required.

The smoke detector for the avionics equipment will be an ionization-type device located within the pressurized avionics space. It will signal the crew as to the location of any fire hazard behind the avionics panels. Since the crew compartment is a wide open space and will be occupied at all time, detection of smoke or flame in this area will be dependent on crew alertness. Suppression

will consist of a single crew-operated fire extinguisher used directly on a source of fire in the crew compartment and used indirectly through fire extinguisher ports provided in the instrument panels in the case of a fire within the pressurized avionics compartment.

1-3.2.7   Landing and Recovery System

Following the hypersonic reentry, the STV will need to decelerate and land safely. This terminal phase of the flight involves several stages, each requiring separate hardware and procedures. The problem is one of dissipating energy in the most reliable, cost (and weight) effective manner while keeping the deceleration loads on the crew to a minimum.

In this section, each portion of this descent and recovery will be discussed separately (including a contingency water landing), although each is interrelated.

Descent Phase. The major issue to addressed during the descent phase is deceleration from supersonic flight to rest. In addition, this must be done with an eye toward the requirements for low cost, minimum weight, and minimum configuration impact (e.g., volume).

Deceleration is initiated at high speed, typically at or before terminal velocity, and should result in a significant reduction in vertical velocity. The terminal flight phase involves a final deceleration to attenuate the ground impact force (discussed in detail in the next section). At all phases of the descent, there should be no adverse deceleration forces on the human occupants.

Many options for descent phase hardware have been built and flown. The major options are described in the following paragraphs. The selection of a preferred concept must also include the concept for impact attenuation once the configuration reaches the ground.

Aerodynamic, high-drag devices would include parachutes, inflatable ballutes or balloons, and foldout speedbrake panels. There have been many designs flown using these techniques. These devices tend to be mechanically very

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simple and pack into fairly dense containment volumes. When fully deployed, they provide a stable, predictable descent. The issues associated with non-rigid, high-drag devices are related to two areas: the reliability of the deployment sequence and the ability to accurately land the vehicle.

The use of wings is another option of providing a stable, controllable descent. These wings can be either fixed or deployable. If the wings are deployable (as in the Rogallo wing envisioned for the Gemini program), they tend to be difficult to integrate into the configuration. They are also mechanically complex and if they have a fail-safe mechanism this also increases their complexity.

An all-propulsive deceleration system, or retrorockets, could also be used. There have been several planetary vehicles, most notably the Apollo lunar module, that used this technique. Earth landing vehicles using all propulsion have been studied extensively (such as for a vertical takeoff, vertical landing single stage to orbit vehicle) but have not been used operationally. The concept uses a rocket pointed into the direction of flight to fire and slow the vehicle, finally firing immediately before ground impact to reduce the vertical velocity to zero.

Impact Attenuation. There are a variety of strategies for impact attenuation, most all of which have been built and tested in the past. All of the terminal deceleration options fall under one of two stratagems: they either reduce the vertical velocity before ground contact or dissipate the energy of impact over some finite distance. Some aerospace systems (aircraft most notably) use a combination of both techniques.

There are two general methods for reducing the terminal vertical velocity before ground impact. One would entail firing a propulsive system to produce a thrust opposite from the direction of flight. The other is to aerodynamically change the low-speed L/D ratio to decrease velocity.

In the case of a rigid wing, high-lift devices, such as flaps, would be necessary. The drawbacks of the horizontal runway landing concepts include the mechanically complex mechanisms required and the requirement for high-speed landing gear (with brakes). The flight test program is fairly involved, and if

a pilot is to have control, forward vision and appropriate controls and displays are also required. The major issue, though, is the horizontal landing velocity. Figure 1-3.2.7-1 depicts the classic relationship between the wing area and touchdown velocity for a range of lift coefficients. As mentioned earlier, the configuration requires that the wing mass (and size) be minimized to reduce the STV's performance penalty. High touchdown speeds also reduce decision times if a human pilot is required to perform critical flair maneuvers. Even with high-lift devices, the fairly blunt shapes associated with low hypersonic L/D vehicles have a very high subsonic drag, which reduces subsonic L/D, resulting in a poor "airplane" for runway landing.

For a non-rigid lifting surface, such as a parafoil, it is possible to deploy large wing areas. An aerodynamic flair or stall can be effected by simple trailing edge deflection and will significantly reduce the vertical and horizontal velocity (see Figure 1-3.2.7-2). The issues associated with this technique involve the control system reliability and the need to accurately sense altitude to initiate a properly timed flair.

Using a retrorocket for impact attenuation in combination with another deceleration device is an attractive alternative. Several aerospace programs have used this technique (such as the Soyuz capsules). A one-shot retrorocket package initiated during the last few feet of descent (Soyuz used a weighted line to contact the ground, Gemini explored a telescoping rod, and modern radar altimeters or Lidars would also work) can very effectively and reliably eliminate most or all of the vertical velocity.

None of the methods considered reduce the vehicle's vertical velocity to zero. Some additional energy dissipation method must be used to bring the vehicle to rest. All aircraft, for example, incorporate a stroking strut as part of the landing gear. Recoverable drones have used airbags, and planetary spacecraft have used retrorockets. The optimum solution for STV may use several techniques for energy dissipation.

Stroking struts provide a controlled, compact deceleration. The struts could be external, as in a conventional aircraft landing gear, or internal (much like Apollo). A fixed chamber is attached to the vehicle, and a sliding piston moves

Figure 1-3.2.7-1. *Wing Area Versus Touchdown Velocity*

- Weight = 21,700 lbs for reference only

$C_{L_{Touchdown}}$

Touchdown Velocity - kts

Wing Area Required - ft²

- Initial high wing loading configuration to penetrate winds

- Disreef to low wing loading landing configuration to reduce vertical and horizontal velocities for landing

- Dynamic flare reduces touchdown velocities further by rapid trailing edge retraction (driven by a rigging shift)

AR = 2.1
Vvert = 35 ft/s
Vhor = 80 ft/s

AR = 3.0
Vvert = 17 ft/s
Vhor = 54 ft/s

AR = 3.0
Vvert = 12.5 ft/s
Vhor = 47 ft/s

Figure 1-3.2.7-2. Parafoil Flair Characteristics

inside the chamber, dissipating energy to either a fluid or a crushable solid. When used externally, a ground contact device, typically a wheel or a skid/pad, is used to spread the load over a much larger area than the strut.

Inflatable airbags have been used on a number of previous vehicles, most often with recoverable drones. Airbags pack efficiently and can use a variety of landing terrain and soils. In the past, airbag designs were fairly intolerant of horizontal landing velocities and rollover was a problem. With staged deflating bags, modern applications (such as envisioned for the ALS P/A module) are more robust. The issues related to airbags are primarily associated with the inflation and integrity of the airbag.

Previous studies indicated that in the size of recovery system for the crew module that the recovery system options are of similar mass.

Water Impact and Floatation. With a dry land landing as a primary recovery mode for the ground-based vehicle, the terminal descent and impact attenuation hardware are designed by the requirements related to hard landings. There are contingency operations, particularly after a launch abort, where a water landing is unavoidable. Because of the problems associated with immersing hardware in salt water, the vehicle may or may not be salvageable for reuse; however, the water impact must be survivable.

Water landing can act to reduce the impact deceleration by a gradual stop over a short distance. On the other hand, impact velocities on the water can produce very high values of dynamic pressure, resulting in structural failure.

The hydrodynamics of water impact is a complex balance of momentum, buoyancy, and drag, which fortunately can be approximated accurately with a less than complete model. Physically, at entry (while the forward part of the vehicle is wetted), the STV crew module imparts a physical, principally transverse velocity to the water, and then the flow separates from the body with the generation of a cavity. Air rushes in to fill the void. Later, the splash forms a dome that closes over the entry point of the body and seals the cavity from the air above. When this surface closure (or seal) occurs, the cavity usually is expanding so that the pressure in the cavity decreases. Because the water

pressure is greater than that in the cavity, the cavity is pushed down into the water and travels down with the body into the water. The pressure differential forces the walls of the cavity to accelerate inward to collapse, leaving the body fully wetted. At this point, the cavitation can be ignored in the analysis and the body's buoyant force and downward momentum are eventually balanced before the rebound to the surface occurs.

The shape of the vehicle affects the build up of drag and the buoyancy force over time as the vehicle penetrates the water's surface. In Figure 1-3.2.7-3, it was seen that a "pointier" shape (such as the biconic) penetrates the water with lower g's that a flatter bottomed entry. Figure 1-3.2.7-4 shows the effect of the same shape entering the water at different attitudes. The recovery system, in this case the parafoil, should therefore be designed to allow the STV crew module to hang in an attitude best suited for water entry. In this case, that probably would entail cutting some of the support risers after the flair maneuver; the vehicle would then swing into a "vertical" orientation for water entry.

Once the vehicle has come to a stop, it will float at an attitude with the pointed end slightly down into the water. This will help ensure that both hatches remain out of the water. Auxiliary floatation bags, such as righting bags, should not be necessary but can be housed in the parachute bay. Further analysis would be required to determine if the floatation characteristics are acceptable or if the addition of sea anchors or other stabilization devices is required.

Lunar Landing Gear Sizing. Part of the STV vehicle design analysis included a preliminary exercise to determine lunar landing gear spread, given a variety of vehicle types and a range of lunar landing conditions. Landing gear was then sized for both the space-based and ground-based vehicles, assuming the vehicle lands in a horizontal attitude on a maximum 15-degree slope, with a landing velocity of 2.0-m/s vertical and 0.5-m/s horizontal velocity. The analysis does not account for landing gear stroke and damping characteristics, but the Apollo lunar module landing conditions and gear spread were used as a comparison model.

Background. The Apollo lunar module was used as an analysis model, based on landing conditions of 1-m/s horizontal velocity, 2-m/s vertical velocity, and a
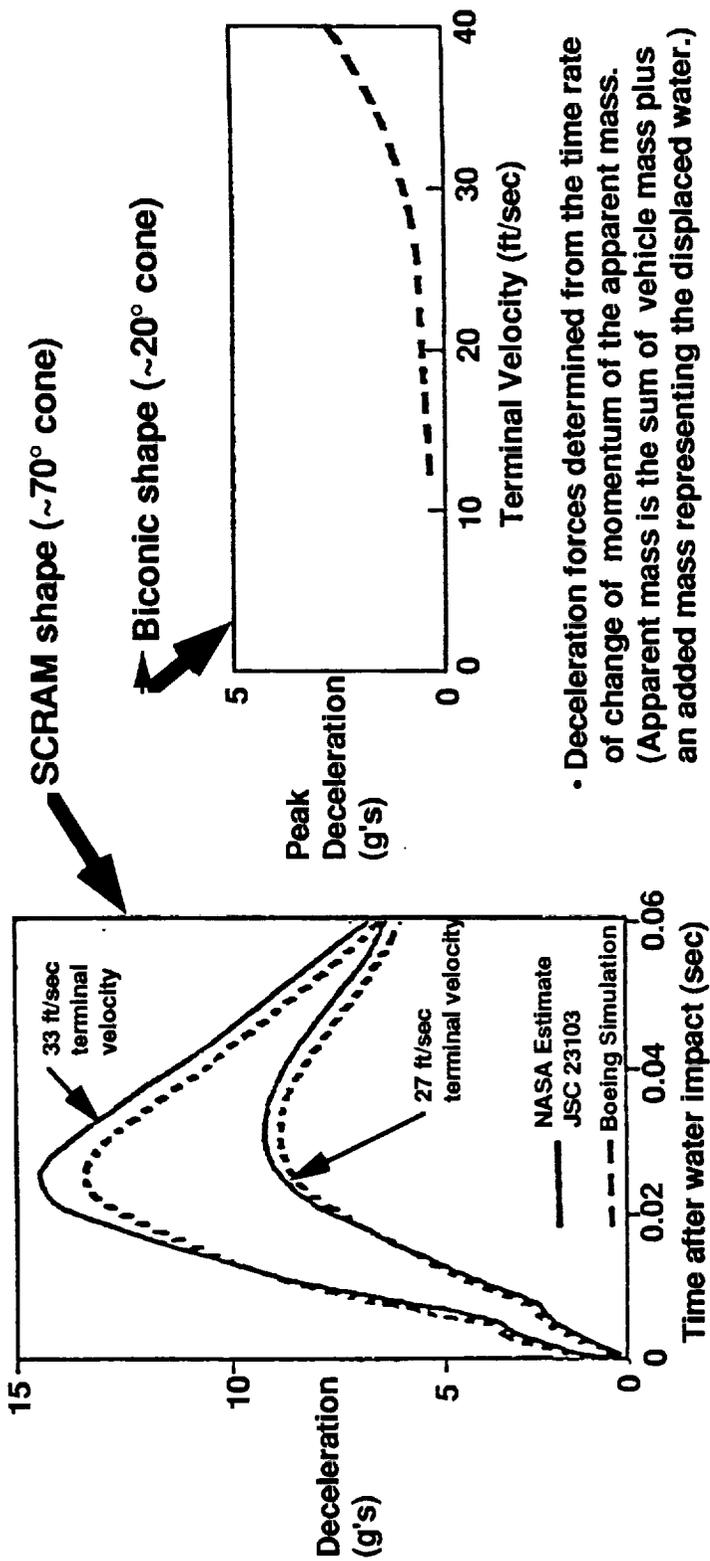
Figure 1-3.2.7-3. Water Entry Comparison
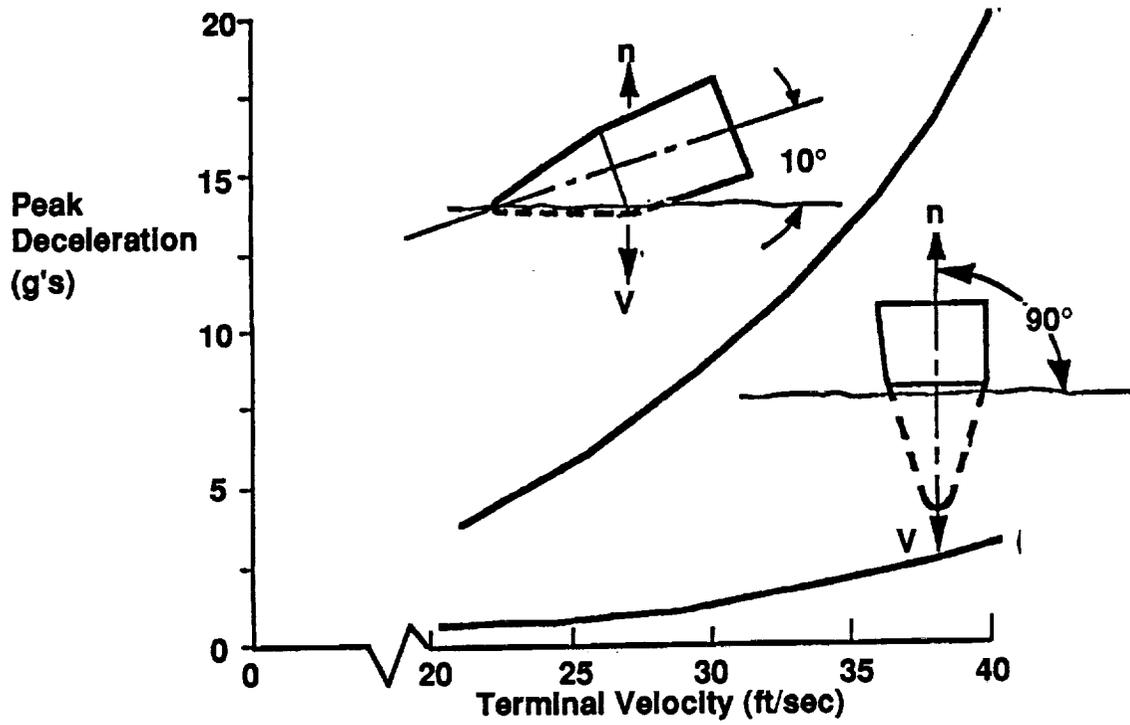
Figure 1-3.2.7-4. Effect of Water Entry Attitude

15-degree landing surface slope. With a center of mass estimated at 3.4m above the landing pad plane, the lunar module landing pad radius was 4.5m from the centerline, or 3.2m from the vehicle center plane. With a landing mass of 7,482 kg, the lunar module landing moment of inertia was estimated to be 24,240 kg-m^2.

Requirements for the STV lunar landing are given in Figure 1-3.2.7-5, and Figure 1-3.2.7-6 presents the space-based and ground-based vehicle landing mass characteristics assumed for this analysis. For the analysis, the landing gear moment required to resist the vehicle descent kinetic energy converted to overturning moment was computed for a range of landing surface slopes and a range of horizontal velocities. Vertical descent velocity was assumed to be 2.0 m/s.

Results. Figures 1-3.2.7-7 and 1-3.2.7-8 show the space-based vehicle landing gear radius as a function of landing surface slope and horizontal velocity for two cases, including a horizontal vehicle attitude case and a case with the vehicle centerline parallel to the velocity vector. The horizontal attitude case is the least conservative of the two, but it was chosen as the design case based on the assumed ability to maintain a horizontal landing attitude. Figure 1-3.2.7-9 shows the ground-based vehicle landing gear radius as a function of landing surface slope and horizontal velocity for only the horizontal vehicle attitude case.

For the final vehicle designs, an unpiloted landing on a 15-degree slope was assumed as the design requirement. Assuming improvement over Apollo-type controllabilty, a maximum horizontal velocity of 0.5 m/s was assumed, resulting in a minimum landing gear radius of 9.4m (6.6m from the center plane) for the space-based vehicle and 9.7m (6.9m from the center plane) for the ground-based vehicle.

1-3.2.8   Launch Escape System

On the ground-based configurations (GB and GO), the crew for the vehicle is aboard the vehicle when it leaves the ground. In this option situations can arise (Figure 1-3.2.8-1) that would require that the crew be able to escape from the

	Basing		Requirement Source		
	Space	Ground	Given	Derived	Provis.
Lunar Landing					
Initial and far side autonomous landing on unprepared surface.	X	X		X	
TBD m diameter landing site					
TBD degree slope					
TBD surface irregularity					
Piloted - landing on prepared surface	X	X	X		
50 m diameter landing site					
<2 degree slope					
<0.2 m surface irregularity					
Capable of cargo jettison in case of landing abort.	X	X		X	

Figure 1-3.2.7-5.   Lunar Landing Requirements

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Vehicle	Mass, kg	Landing CG, m from Surface	Landing Moment of Inertia, kg-m2
Space-based Lander, Piloted	67,300	8.78	1,076,800
Ground-based Lander, Piloted	59,300	8.40	1,366,272

Figure 1-3.2.7-6. Vehicle Landing Mass Characteristics
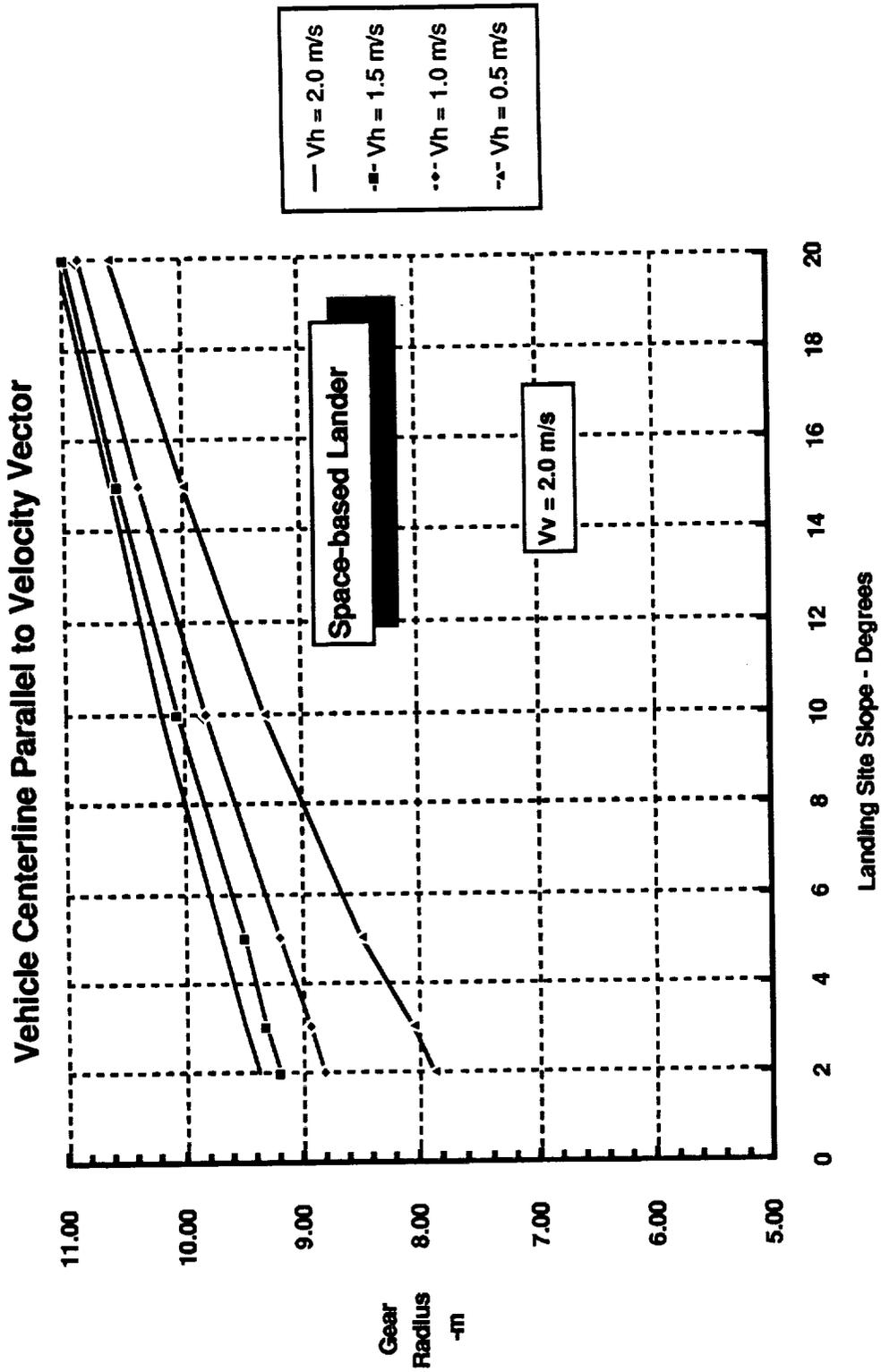
Figure 1-3.2.7-7. Space-Based Vehicle Landing Gear Sizing
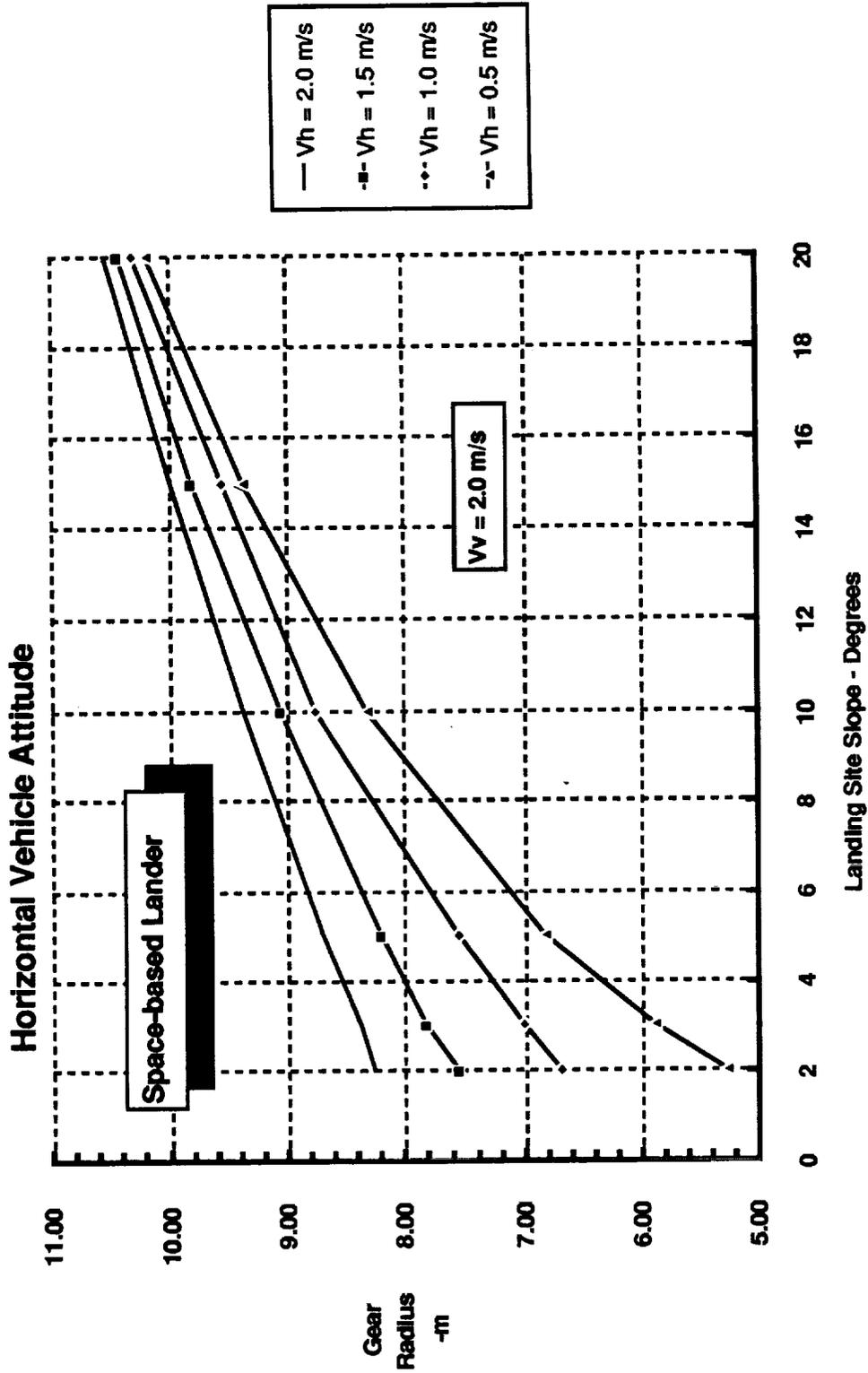
Figure 1-3.2.7-8. Space-Based Vehicle Landing Gear Sizing
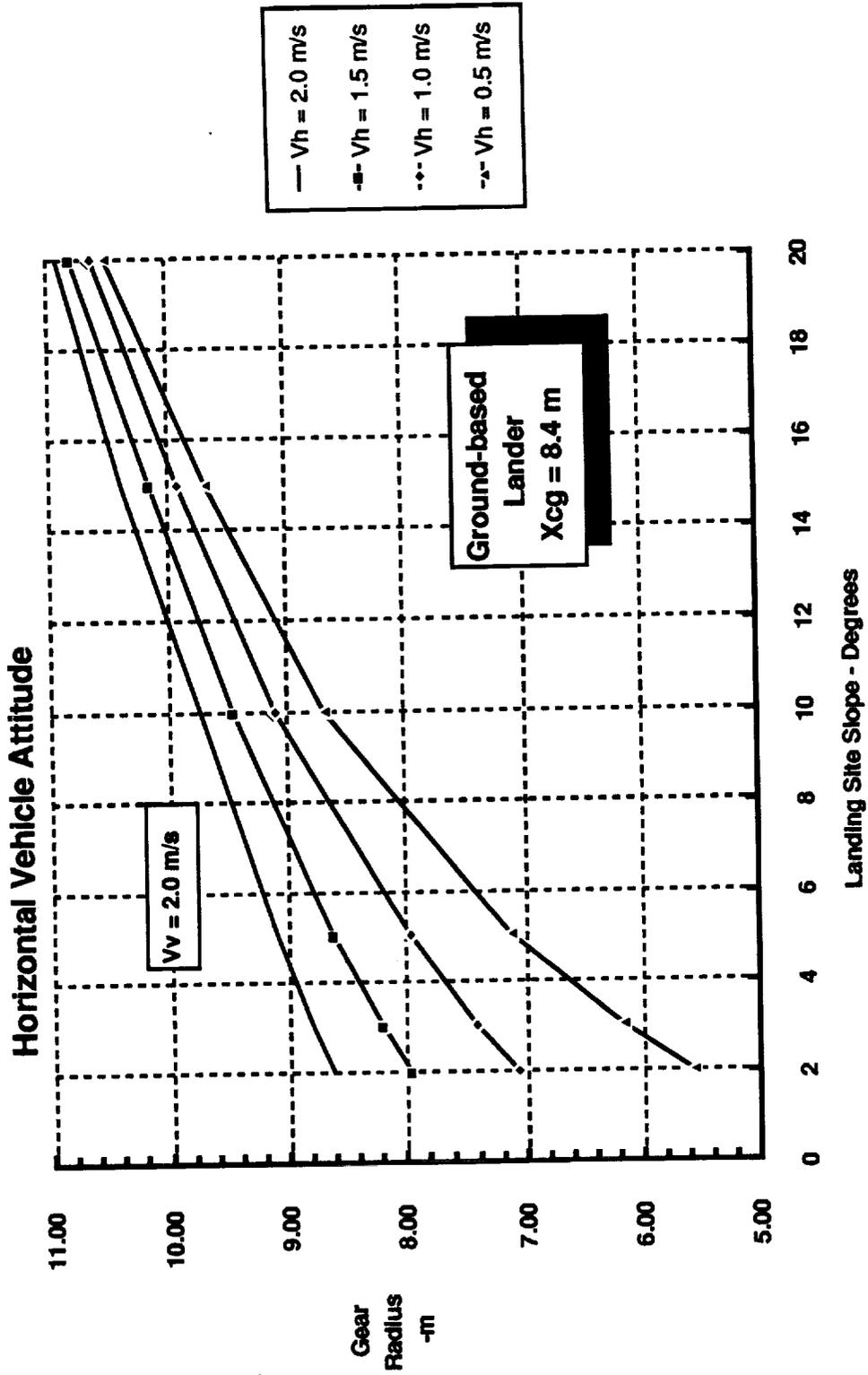
Figure 1-3.2.7-9. Ground-Based Vehicle Landing Gear Sizing

Emergency Conditions Requiring Abort
Concept Definition ——— MSFC- *BOEING*

STV
Space Transfer Vehicle

EMERGENCY	POSSIBLE CAUSES	TIME AVAILABLE FOR ESCAPE	COMMENTS
Explosion	• Propellant detonation	<5 seconds	More time may be available with appropriate sensors; distance also enhances survivability
Fire	• Flammable fluids/materials	5 - 20 seconds	Depends on location and intensity
Loss of Control	• Control system failure • RCS failure • Loss of thrust • Collision • Structural failure • Actuator/valve failure	1 second to 10 minutes	Depends on attitude, moments of inertia, and dynamic pressure
Damaged vehicle	• Collision • Meteoroids	1 second to minutes	Depends on location and energy
Graceful System Degradation	• Instrument failure • Loss of power • Loss of thrust	30 seconds to hours	Examine strategies for redundancy, reliability, and work-arounds
Hazardous Environment	• ECLSS failure • Toxic gas (fire) • Loss of pressure	5 seconds to hours	Depends on size and type of backup systems

Figure 1-3.2.8-1. Emergency Conditions Requiring Abort

launch vehicle. Therefore, it is necessary to provide the crew with a launch escape system to permit crew escape during the boost phase of the launch.

Previous studies have traded many options for escape provision. Of all the options, the clear winner for a vehicle carrying 10 people is to physically separate the pressurized crew cab from the rest of the vehicle (typically using a launch escape system (LES)) and return that cab to Earth intact. To determine the design requirements for such a system, one must first examine what hazards or emergencies would precipitate an abort.

The primary concern that sizes the LES is the danger presented by fire and explosion in the booster. Explosive and fire emergencies would result primarily from chemical reactions involving propellants and/or high-pressure gas storage (e.g., ECLSS tanks). The reaction rate varies considerably with propellant type, containment and structural arrangement, method of initiation, degree of mixing, and the environment. All explosive reactions, though, are characterized by significantly increased temperature and pressure, which can lead to secondary failure modes. The hazards associated with explosions include:

1. Shock wave and detonation wave.
2. Thermal radiation.
3. Shrapnel.
4. Fireball.

The shock wave is a pressure pulse radiating out from the point of explosion. Technically, the shock wave propagates at Mach 1 and contains virtually none of the total energy released in an explosion. The detonation wave, on the other hand, is the violent "blow up" that contains most of the released energy of the explosions (in some cases close to 100%) and typically travels outward at about Mach 10. Both the peak overpressure and the duration of the pulse are significant. For example, humans will sustain lung damage when experiencing a 15 psi pulse for 0.1 seconds; much higher pressures are survivable if the pulse duration is reduced. Cryogenic fuels tend to produce detonation waves of short duration and high intensity; propellants such $UDMH/N_2O_4$ deflagrate with longer periods and lower overpressures. In addition to the danger to humans,

structures will subsist if a powerful shock wave is short in comparison to the structural response time.

Thermal radiation damage depends on factors such as heat transfer rate, luminosity, temperature intensity, and spectral distribution. Except for emergencies that are inside or have penetrated the crew pressure vessel, the humans will probably be adequately shielded. However, other components, such as exposed launch escape solid rocket motors, would be significantly affected.

Shrapnel damage depends on design, failure mode, and relative spatial orientation. At the conceptual design level, it may be difficult to assess requirements for crew protection.

Fireballs are maybe the least understood explosive phenomena. Unlike the detonation wave, which is virtually impossible to outrun, previous manned spacecraft escape systems were all sized to avoid the predicted fireball. A fireball is formed as a result of a temporary equalization of gas flow that becomes an isotropic (although highly turbulent, formation of incandescent gases, typically representing only 1% to 5% of the total energy released) and can locally travel at speeds up to Mach 5. Avoiding or escaping the fireball reduces hazards caused by fragmentation, temperature rise (burning), spectral energy, toxicity, and exposure to unburned propellants. As in the case of thermal radiation, the crew cabin is vulnerable, as is the exposed escape system.

The type of launch vehicle propellant directly sets the requirements for a launch escape system. Figure 1-3.2.8-2 depicts some representative boosters and the response time that would be available in the event of a catastrophic event. Note the systems that use solid propellants (which are fully mixed oxidizer and fuel) are extremely short. The TNT equivalent column is presented to give a relative sense of the potential explosive force that is available. Figure 1-3.2.8-3 shows the TNT equivalent effect in an explosion. Although not all propellant detonations behave as TNT, it is an accepted practice to use these equivalents for comparison purposes. For example, various Government agencies rate LO_2/LH_2 as 20% to 60% the TNT equivalent by mass.

Flight Phases / Hazard Condition	Pre-Launch ~4h	Launch/initial ascent ~2m	Hypersonic ascent ~6m	Orbital flight 0 to 66h	Re-entry ~45m	Landing/Post Landing ~1h
Propulsion Systems:						
Booster Propulsion	<1s to 2m	<1s to 30s	15s to 1m	5m to 66h	5s to 1m	
OMS/RCS Propulsion	<1s to 30s	<1s to 30s	1m to 6m	to 66h	5s to 10s	
Fuel lines, valves, pumps, tanks			<1s to 3			
Thermal Protection				? to 66h	5s to 10s	
ECLSS						
Pressurization	? to 4h	? to 5m	? to 10s	? to 10s	? to 10s	
Oxygen supply		? to 2m	? to 6m	? to 4h	? to 45m	5s to 30m
Contamination	5s to 30m	5s to 2m	5s to 6m	5s to 30m	5s to 30m	
Aerodynamic devices				10s to 66h	1s to 1m	
Collision	<1s to 30s	<1s to 30s	<1s to 30s	10s to 12h	10s to 1m	<1s to 30s
Chemical Explosion	5s to 30s	5s to 30s	5s to 30s	5s to 30s	5s to 30s	5s to 30s
Cabin Fire	5s to 30s	5s to 30s	5s to 30s	5s to 30s	5s to 30s	5s to 30s

s = second   m = minute   h = hour

Figure 1-3.2.8-2. Typical Escape Times Available

STV Launch Vehicle Catastrophic Failures
Concept Definition
Space Transfer Vehicle ——————— MSFC- **BOEING**

- Launch vehicle selection directly sets the requirements for LES

Booster Options	Propellant Combination	TNT Equivalent	Available Response Time (seconds)		
			Leak	Blast Wave	Fireball
•Titan IV	NTO/A-50 AP/PBAN/Al	20% 100% (595 tons)	0.02-0.05	0.002-0.005 (100%)*	3-10 (liquids)
• Liquid (ALS-type)	LH_2/LO_2	20% (200 tons)	1-5	0.5 (1-5%)*	3-20
• Liquid (S-IC type)	LOX/RP-1	(~100 tons)	1-5	0.5 (1%)*	3-40
• Shuttle C	LH_2/LO_2 AP/HTPB/Al	(1800 tons)	0.02-0.05	0.002-0.005 (100%)*	3-60 (liquids)

* energy converted to deflagration

Launch vehicle options that utilize solid propellants significantly decrease opportunities to save personnel in abort/failure situations!

Figure 1-3.2.6-3. Launch Vehicle Catastrophic Failures

With liquid propellants, it is highly unlikely that the maximum energy potential of an explosion will be encountered. The problem is basically on of incomplete mixing. Even in experiments where full mixing was attempted before detonations, the full potential was never realized. Figure 1-3.2.8-4 depicts a time-phased altitude plot of a postulated detonation and the warning time required to effect a successful launch escape. In this case, a STV crew module sits atop an ALS vehicle with close to 2 million pounds of propellants. At time zero, sensors indicate that a failure is imminent and the LES is initiated. At time 0.5 seconds (conservative by proven systems), the LES ignites and pulls the STV crew module away (the multiple traces representing various acceleration levels). In this example, the ALS detonates (at 3 seconds representing a typical time between warning and actual detonation) after a hypothetical complete mixing in the region between the oxidizer and fuel tank. The blast wave moves out very rapidly but diminishes quickly. The pressures shown would be attenuated, such that the crew would not feel those values (even a simple aluminum skin would reduce the pressure by an order of magnitude). The fireball would eventually "liftoff", rise, and dissipate, but much later, well after the STV crew module is departed. Note also the normalized curve for an actual Atlas Centaur detonation that does not come close to the theoretical worst case. From this example, one can see that with a few seconds warning time, a catastrophic booster detonation should be survivable. With a solid rocket, the detonation point would be moved close to time zero (reflecting the minimal warning time associated with a failure, such as a crack in the propellant) and no LES would be effective.

Abort Trajectories. Two abort modes exist for the vehicle during the atmospheric boost phase (Figure 1-3.2.8-5). The first abort mode involves the use of the LES. The crew cab is lifted away from the launch stack with an altitude increase of of approximately 10,000 feet and the vehicle is also sent downrange to clear the launch system. The deceleration device (parachute/parafoil) is deployed around 5,000 feet and the STV crew module is then recovered. This abort scheme is typically used until the vehicle achieves a perigee altitude of 40 to 50 nmi. An abort in the early phases will result in the STV crew module landing in the ocean. When the launch system reaches Mach 10 to 12, the recovery can be extended to land.

- ALS 1-1/2 stage launch vehicle with 1,960,00' of cryogenic propellants
- Energy potential converted to deflagration represents an equivalent of 200 tons of TNT
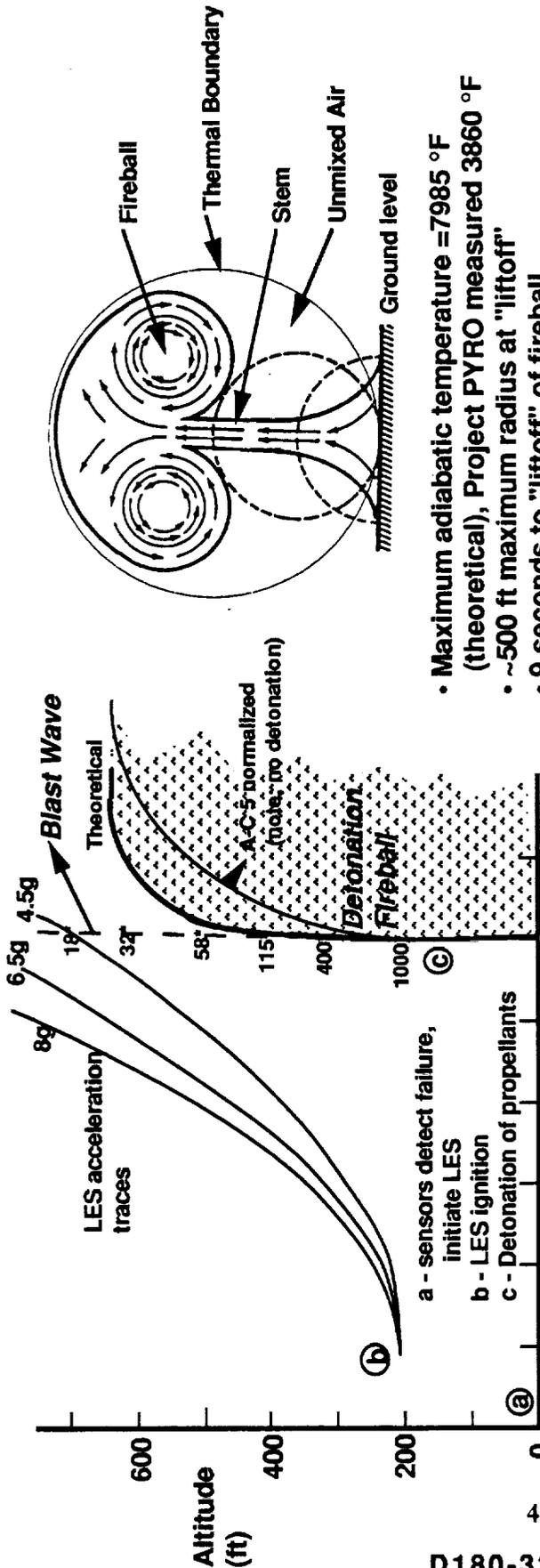
- Maximum adiabatic temperature =7985 °F (theoretical), Project PYRO measured 3860 °F
- ~500 ft maximum radius at "liftoff"
- 9 seconds to "liftoff" of fireball
- 23.7 seconds total duration to snuff

* Pressure factor = overpressure (psi) x attenuation factor

a - sensors detect failure, initiate LES
b - LES ignition
c - Detonation of propellants

Figure 1-3.2.8-4. Altitude Versus Time for a Postulated Launch Vehicle Explosion

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STV Launch Abort Scenarios
Concept Definition

MSFC - *BOEING*

STV
Space Transfer Vehicle

Category	Altitude	Description
On Pad	On Pad	Should launch vehicle become unsafe the LES shall be activated
Launch	L/O - 300k ft	Any aborts through 300k ft will use LES to put crew and capsule in the Atlantic Ocean
Post Launch	300k ft to Orbit	Core engines will separate the core vehicle from the ALS and place crew cab in safe landing trajectory
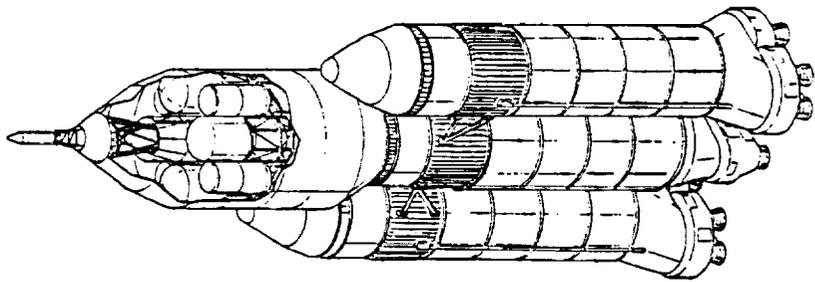
Figure 1-3.2.8-5. STV Launch Abort Scenarios

The second major abort scenario is the abort to orbit. The window for this type of abort is very dependent on the booster system. For a typical ALS this abort can occurs as early as liftoff with an engine failure. The STV crew module is injected into a low (20 by 80 to 100 nmi) orbit and the vehicle reenters without any maneuvering.

LES Sizing. Sizing the LES is based on the most demanding energy requirement for successful abort. This case corresponds to the off-the-pad scenario, where the launch vehicle is not moving, but the STV must ascend to an altitude sufficient for recovery devices to deploy. With the preferred configuration using a parafoil, test data indicates that a minimum of 3,000 feet is required to ensure successful parafoil deployment from any attitude. Adding another 2,000 feet for conservatism, the LES will require around 606 ft/s ΔV capability to pull to STV crew module to an apogee of 5,000 feet. Figure 1-3.2.8-6 shows how the STV design point compares to the Apollo system. The STV LES will probably be overdesigned and will approach the performance of the Apollo system.

Having established the LES requirements, there are several options to consider. The physical location, interface reusability, and propellant/thruster combination must all be considered simultaneously. The object is to incorporate a LES that is the most inexpensive and reliable and least obtrusive to the rest of the STV design.

Addressing first the issues involving the location of the LES, it becomes apparent that the major issues are efficiency of the load-carrying structure, the ability to jettison the LES upon achieving a specified altitude, and abort mass. The three configurations addressed had a bottom-mounted, side-mounted, and top-mounted LES.

In a bottom-mounted LES, the rocket motors that would propel the vehicle off the stack sit under the crew module, which in this configuration, places the escape motor inside the vehicle itself. This location makes jettisoning this motor after reaching a point where it is no longer needed very difficult.

Launch Escape System(LES) sizing is
sensitive to minimum parachute opening altitude
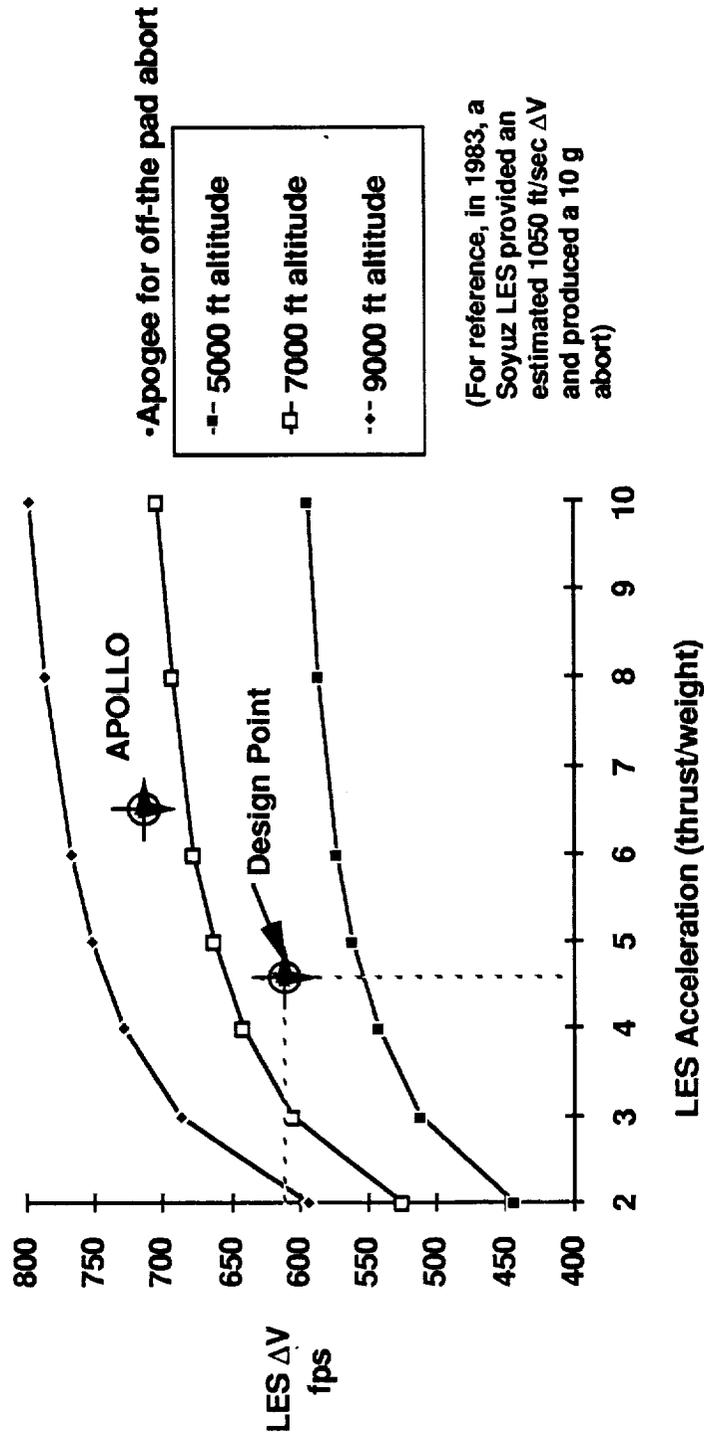
Figure 1-3.2.8-6.   STV LES Sizing

BOEING

For a side-mounted LES, the escape motors are placed alongside the crew module during the launch. This configuration is relatively easy to jettison when it is no longer needed; however, when these motors are ignited they introduce large moments into the crew module structure.

The top-mounted LES is a configuration that has been used on several manned missions in the past (e.g., Mercury, Apollo, and Soyuz). This configuration allows the LES to be easily jettisoned during the boost trajectory and also allows the escape system structure to pick up the crew module's main structural members and smoothly transfer the loads into the STV.

The propellant/thruster combination is not independent of the choice of LES location. If the escape system uses a liquid rocket motor, the tankage and plumbing required to move the propellant to the engine becomes a significant issue. This integration is the major issue with the choice of the LES propellant and thruster. This integration issue is strongly in favor of a solid rocket motor. In a solid rocket motor, the thrust is provided by a self-contained package requiring no plumbing or tankage to supply its propellant.

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1-4.0   OPERATIONS AND LOGISTICS

1-4.1   90-DAY STUDY AND FOLLOW-ON STUDY

During the first several months of the contract, the study schedule was suspended to allow our team to support the MSFC Skunkworks team. Several tasks were performed in support of operations, primarily in Earth-to-orbit (ETO) manifesting. We looked at different methods of packaging the lunar vehicle into Shuttle-C vehicles. The aerobrake was the biggest problem because of its size. The best combination was to put the aerobrake parts, which are low mass and high volume, with a propellant tank, which is low volume and high mass.

Several lunar vehicle designs would not package into the ETO vehicle efficiently. This brought out a design goal in which the lunar vehicle's mass should be targeted just below an integral multiple of the ETO capability. It was recognized by the operations community that expect to achieve this goal as a requirement while both the booster and the STV were in preliminary design phase was illogical, but it should be kept in mind by the design community that the two vehicles are interrelated.

We looked at KSC facilities to determine if any were suitable for the processing of a cryogenic vehicle as large as the lunar transfer system (LTS). We concluded that the hazards associated with the tanking tests and the size of the tanks dictated a new facility.

1-4.1.1   DRS1

Design reference scenario 1 (DRS1) is a starting point for analyzing assembly and refurbishment operations associated with space basing the Option 5 LTS. The goal was to determine if basing the LTS in space was feasible or not (Figure 1-4.1.1-1).

Assumptions. The following assumptions were made:

1.   The lunar base will be man tended, with one mission per year with a 6-month stay time per mission.

Figure 1-4.1.1-1. Design Reference Scenario 1 (Sheet 1 of 3)
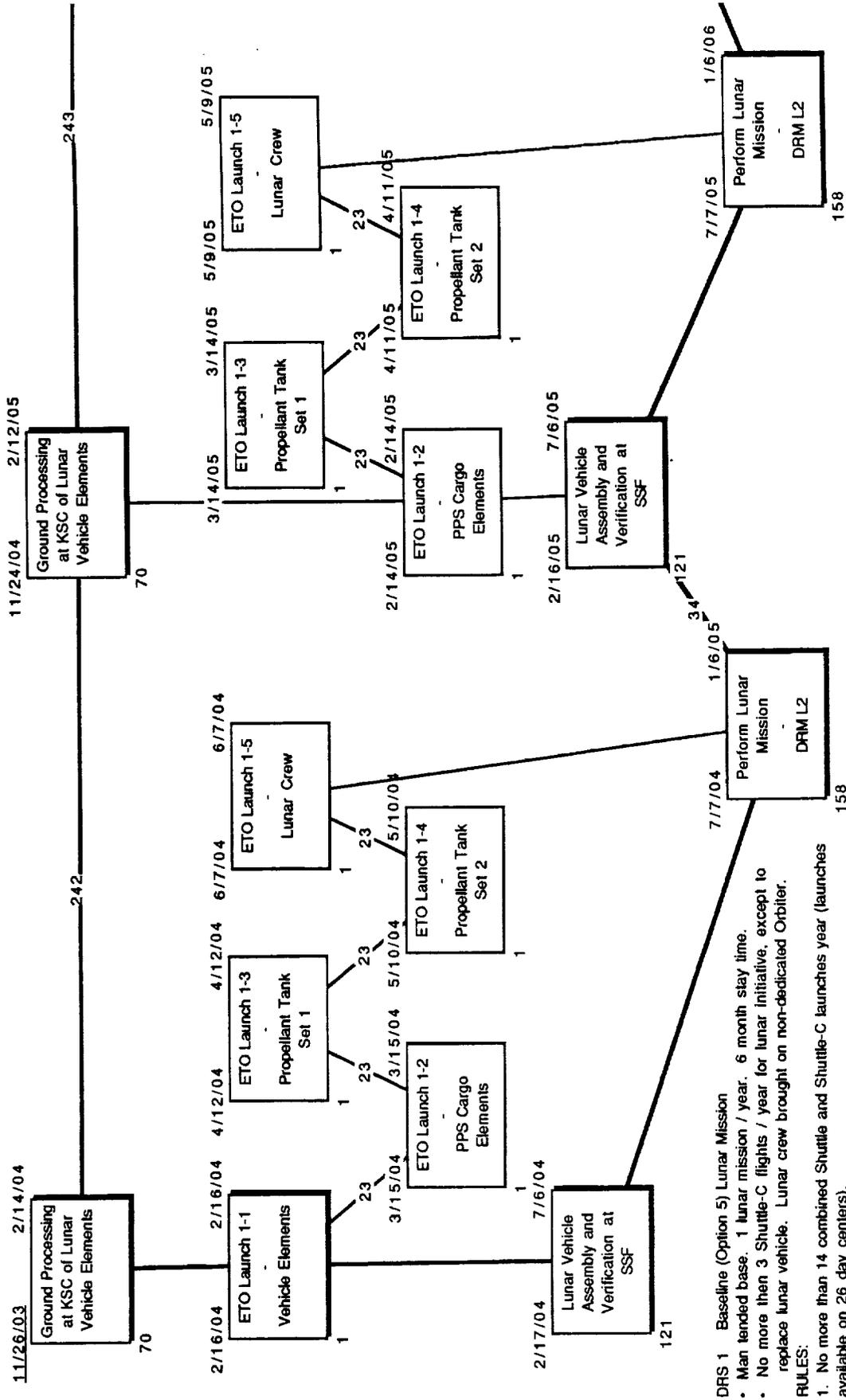
DRS 1 Baseline (Option 5) Lunar Mission
• Man tended base. 1 lunar mission / year. 6 month stay time.
• No more then 3 Shuttle-C flights / year for lunar initiative, except to
  replace lunar vehicle. Lunar crew brought on non-dedicated Orbiter.
RULES:
1. No more than 14 combined Shuttle and Shuttle-C launches year (launches
   available on 26 day centers).
2. SSF Turnaround time for lunar mission is 121 shifts (MDSSC)
3. Schedule is based on a 6 day / 48 hour work week.

Figure 1-4.1.1-1. Design Reference Scenario 1 (Sheet 2 of 3)
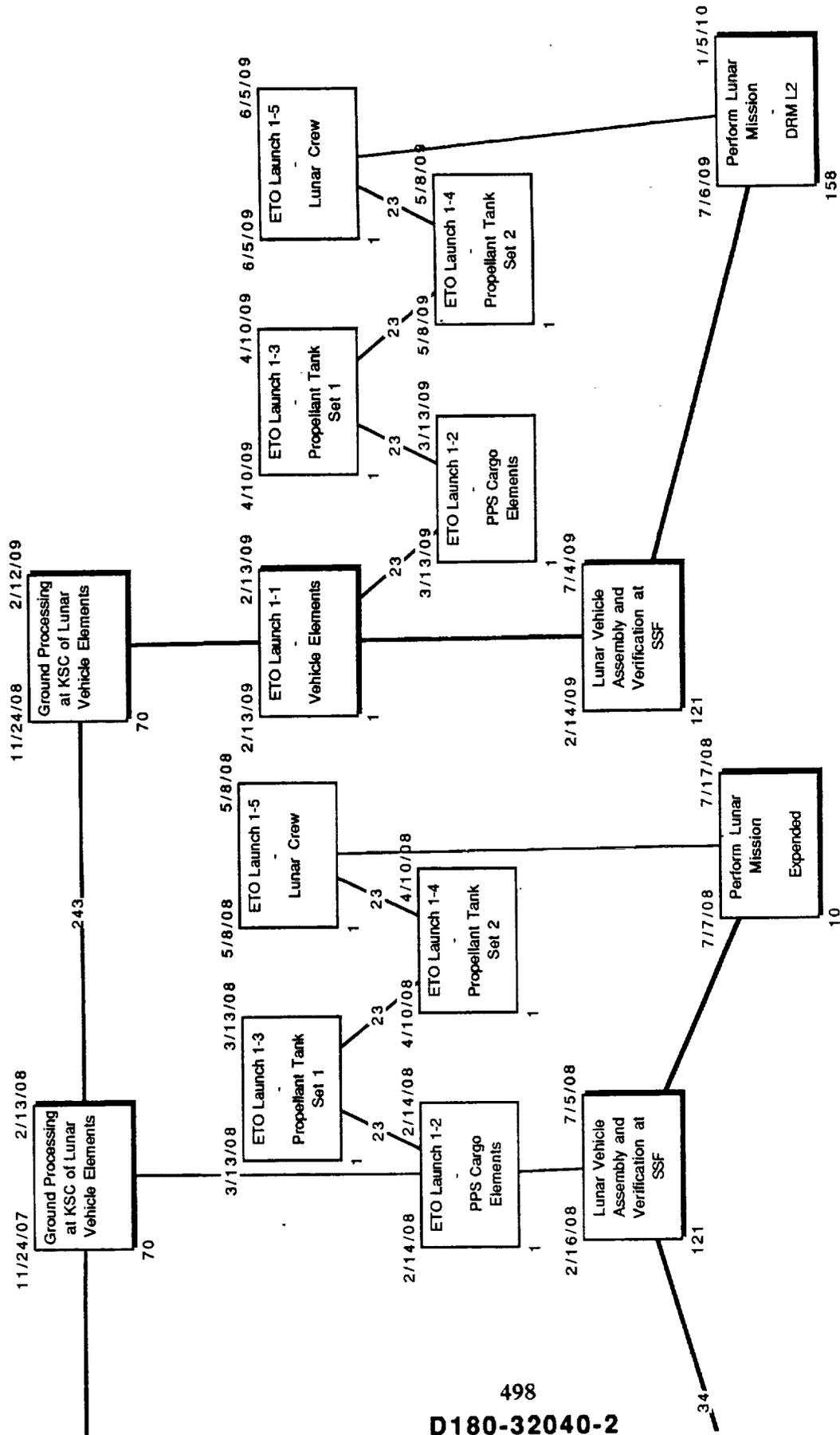
Figure 1-4.1.1-1. Design Reference Scenario 1 (Sheet 3 of 3)

2. LTS is space based and reusable, and Space Station Freedom (SSF) is the LEO base. There will be four piloted flights and the fifth flight is cargo only and expended at the moon.

3. The Shuttle-C HLLV is available (4.6m diameter by 71 tons and 7.6m diameter X 61tons). There will be two launches per lunar mission.for LTS elements, and the third launch will be for boosting replacement LTS vehicles. Lunar cargo and crew will ride on the orbiter (different flights) and launch on 26-day centers (14 launches per year).

4. The MDSSC On-Orbit Assembly/Servicing Task Definition Study mid-term report will be used for LTS task times at the Space Station.

Conclusions. Nominal vehicle processing does not stress any support systems. Also, the SSF planned growth path can support processing requirements.

1-4.1.2   DRS2

NASA was considering a permanently manned lunar base. The LTS requirements were the same as in DRS1, but the processing rate would double so a new lunar crew could be landed at lunar base prior to the first crews departure (Figure 1-4.1.2-1). DRS2 was to determine if the Space Station could process two lunar vehicles per year.

Assumptions. The following assumptions were made:

1. The lunar base will be man tended, with two missions per year with a 6-month stay time per mission. The vehicle will stay with the crew (vehicle and crew are a team).

2. LTS is space based and reusable, and Space Station Freedom is the LEO base. There will be four piloted flights and the fifth flight is cargo only and expended at the moon.

3. The Shuttle-C HLLV is available (4.6m diameter by 71 tons and 7.6m diameter X 61tons). There will be two launches per lunar mission.for LTS elements, and the third launch will be for boosting replacement LTS vehicles. Lunar cargo and crew will ride on the orbiter (different flights) and launch on 26-day centers (14 launches per year).

BOEING

Figure 1-4.1.2-1. Design Reference Scenario 2 (Sheet 1 of 4)
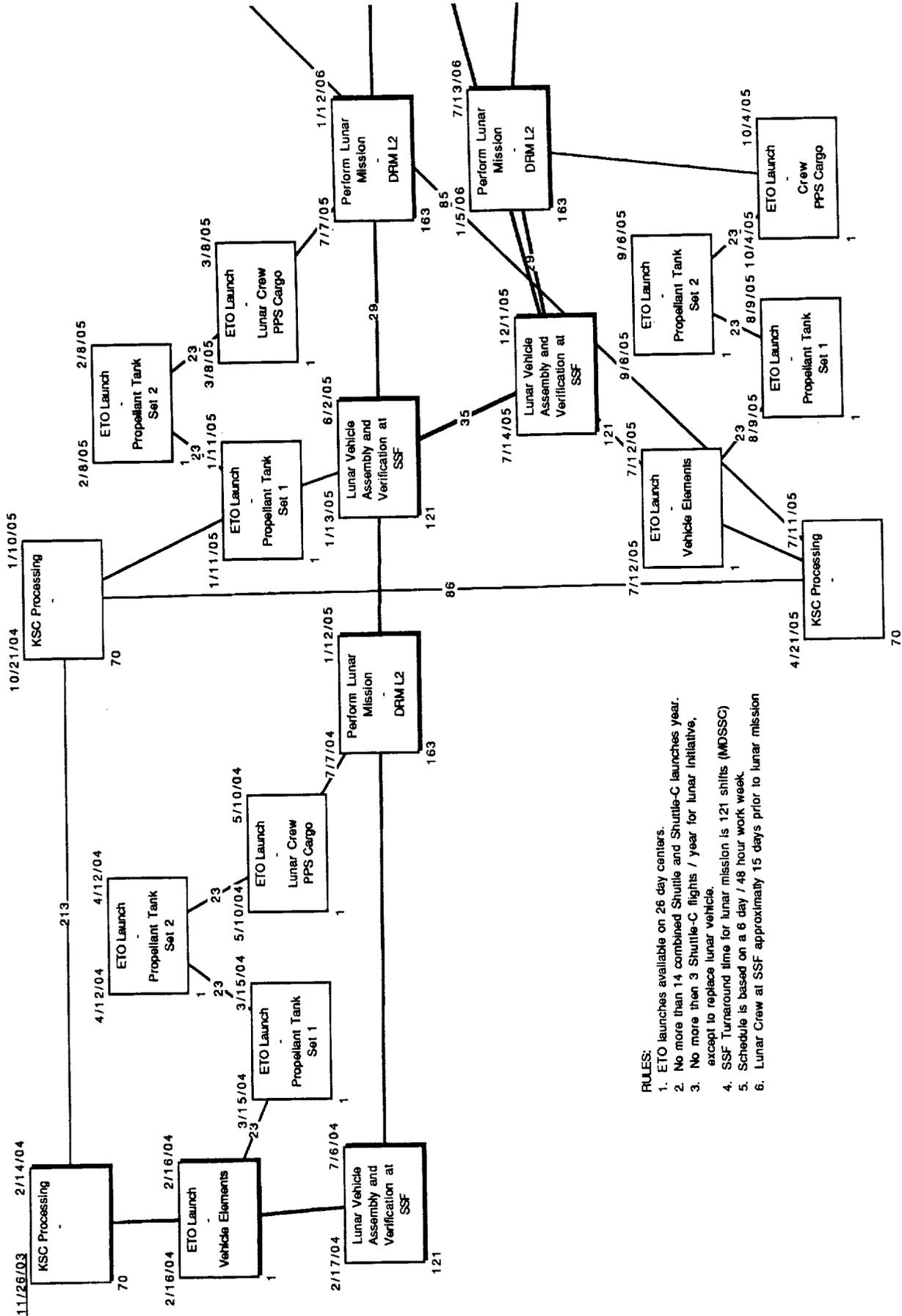
RULES:
1. ETO launches available on 26 day centers.
2. No more than 14 combined Shuttle and Shuttle-C launches year.
3. No more than 3 Shuttle-C flights / year for lunar initiative, except to replace lunar vehicle.
4. SSF Turnaround time for lunar mission is 121 shifts (MDSSC)
5. Schedule is based on a 6 day / 48 hour work week.
6. Lunar Crew at SSF approximatly 15 days prior to lunar mission

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BOEING
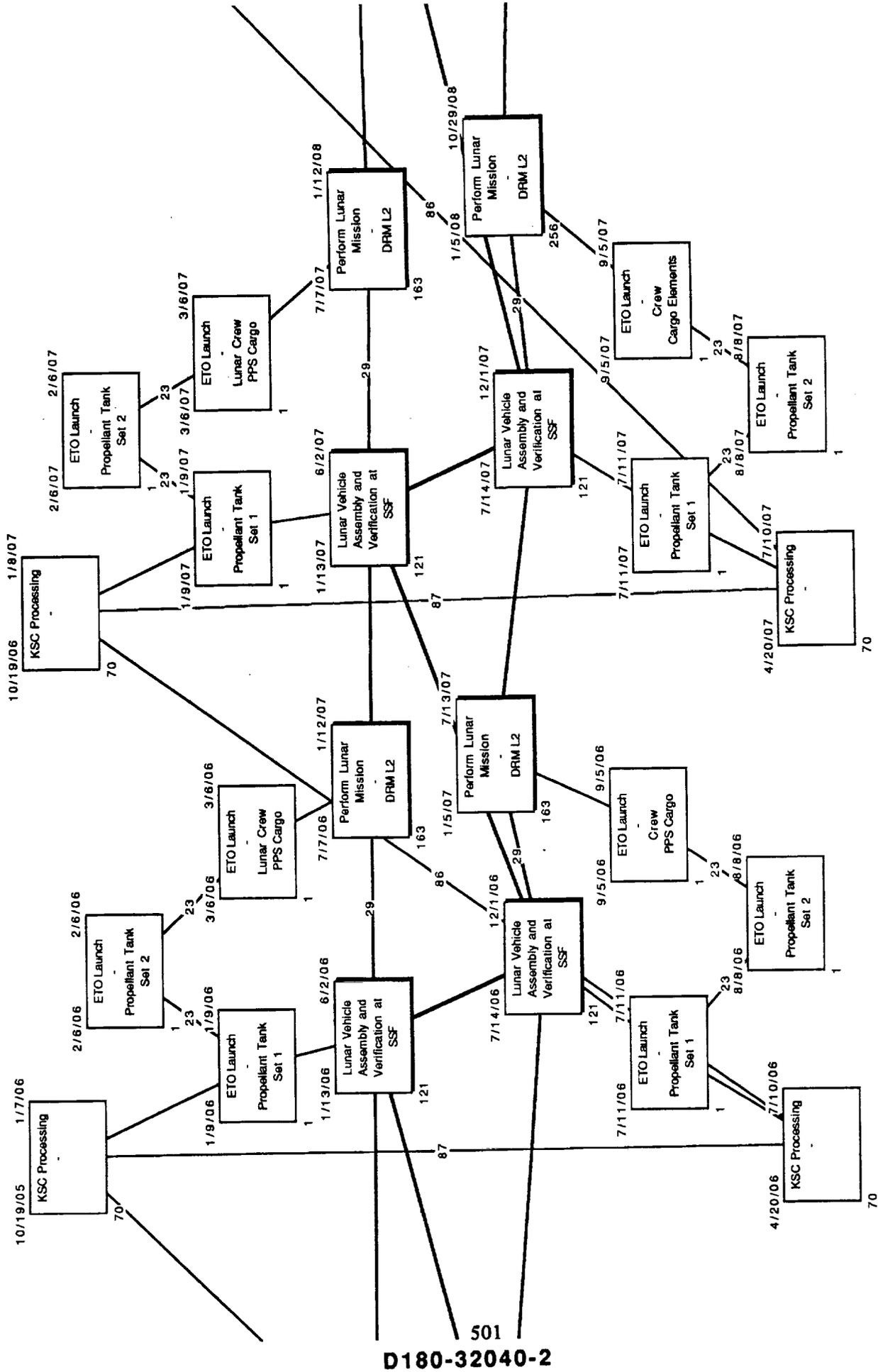
Figure 1-4.1.2-1. Design Reference Scenario 2 (Sheet 2 of 4)

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Figure 1-4.1.2-1. Design Reference Scenario 2 (Sheet 3 of 4)

BOEING

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BOEING

Figure 1-4.1.2-1. Design Reference Scenario 2 (Sheet 4 of 4)

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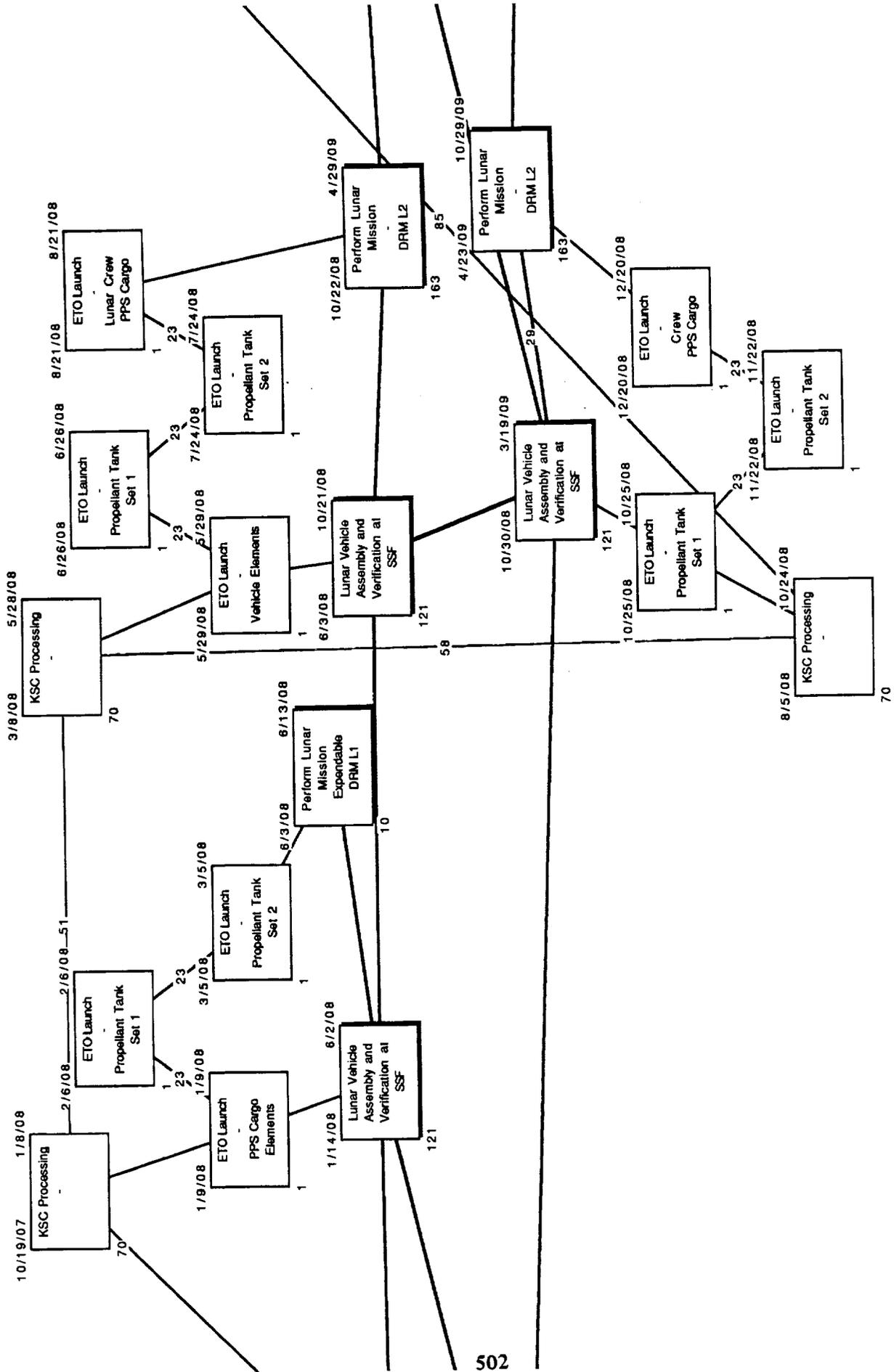
4. The MDSSC On-Orbit Assembly/Servicing Task Definition Study mid-term report will be used for LTS task times at the Space Station.

**Conclusions.** The DRS2 schedule can be divided into two logical parts: steady-state operations and vehicle replacement operations.
Figure 1-4.1.2-2 shows that the SSF processing facility is nearly schedule saturated. Figure 1-4.1.2-3 shows that the SSF is schedule saturated.

SSF operations are critical path and program constraining (within the accuracy of the operations estimates). Vehicle processing activities occur 77% of the year for steady state and 98% of the year for vehicle replacement years. Vehicle replacement occurs every 2.5 years.

Vehicle replacement causes one lunar mission to be extended by 108 days to maintain manned presence at lunar base every 2.5 years. Reduction of the amount of time and manpower required to perform assembly and refurbishment tasks should probably be reduced. An operations-reduction study should be initiated to determine the potential return on investment and programmatic risk reduction for different operations reduction techniques.

Four Shuttle-C flights per year (steady state) exceeds the proposed baseline. Five flights are required in vehicle replacement years. The Shuttle-C/Shuttle program as proposed seems incapable of meeting both SEI and SSF requirements with any schedule margin. Reliance on a single booster system to service an expanding manned presence in space may also have unacceptable programmatic risk.

Resource contention remains a political issue. As such, resources can be made available, if they are _made_ available. Power, thermal, telemetry, and manpower appear sufficient but at the expense of other proposed users.

Some new technologies are considered enabling for space basing and are list below:

Figure 1-4.1.2-2. Design Reference Scenario 2

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STV DESIGN REFERENCE SCENARIO 2 (CONT)

MSFC- BOEING

Vehicle Replacement Operations, Shifts

Space Station
Vehicle 1 Refurb — 121
Vehicle 3 Refurb — 121
slack — 8
Vehicle 2 Refurb — 121

(steady state resumed)

Ground Processing
Vehicle 1 Processing — 70
Launches — 53
Vehicle 3 Processing — 70
Launches — 57
Vehicle 2 Processing — 70
Launches — 70

(steady state resumed)

#1 Expended
#2 Extended Stay
#3 Replacement
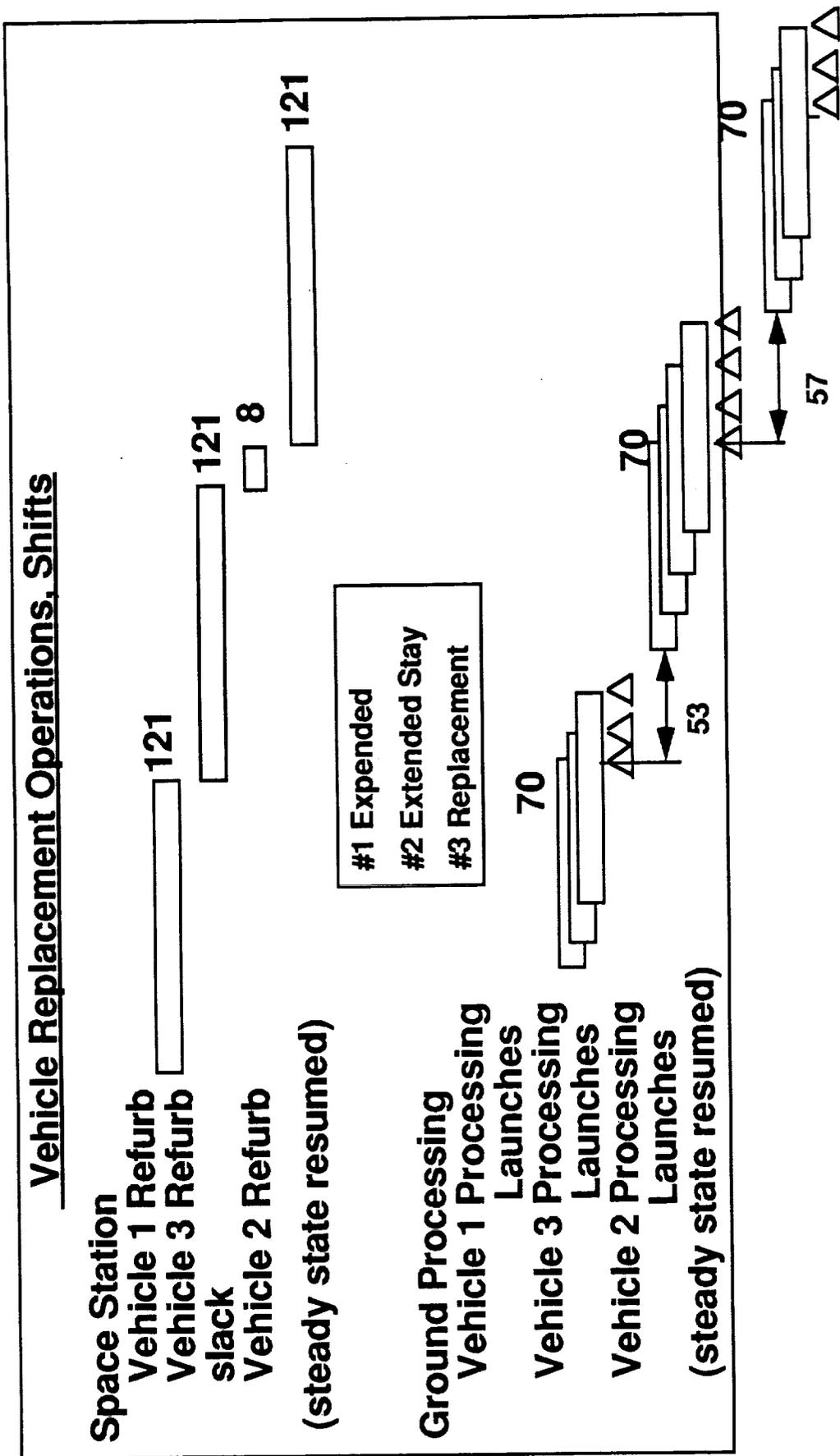
Figure 1-4.1.2-3. Design Reference Scenario 2

1. A very large RMS capable of manipulating the largest vehicle element (70-ton cryogenic tank). Positioning torques must be considered in both the remote manipulator system (RMS) and the SSF structure.

2. Levels of built-in test beyond what is available today. The Option 5 vehicle design leaves vehicle elements in lunar orbit that will be required to complete the mission. These elements must be known to be functional prior to the lunar crew leaving the LEO.

3. Interfaces that are made on orbit should be self-mating and self-verifying. The potential return on investment should be determined, but it appears that the cost and risk of assembling vehicles in space in the same manner we assemble stages at KSC is prohibitive.

1-4.1.3   Preliminary DRS3

Figure 1-4.1.3-1 shows the preliminary design reference scenario 3.

Assumptions. The following assumptions were made:

1. The lunar base will be manned, with two missions per year with a 6-month stay time per mission. The vehicle does not stay with the crew.

2. LTS is space based and reusable, and Space Station Freedom is the LEO base. There will be four piloted flights and the fifth flight is cargo only and expended at the moon.

3. The Shuttle-C HLLV is available (4.6m diameter by 71 tons and 7.6m diameter X 61tons). There will be two launches per lunar mission.for LTS elements, and the third launch will be for boosting replacement LTS vehicles. Lunar cargo and crew will ride on the orbiter (different flights) and launch on 26-day centers (14 launches per year).

4. The MDSSC On-Orbit Assembly/Servicing Task Definition Study mid-term report will be used for LTS task times at the Space Station.

5. Capture all of 89CNDB with the STV. There will be one HLLV launch for propellant and one orbiter launch for payload.

Goals. DRS3 as defined above was to be used as a model to determine the operational sensitivities of the proposed STV program (lunar + 89CNDB). Derivatives of DRS3 were to be used to explore the effects of changes to the

Figure 1-4.1.3-1. Preliminary Design Reference Scenario (Sheet 1 of 6)

Prelim 3   Baseline (Option 5) Lunar Mission + Secondary Mission Capture
- Man tended base. 1 lunar mission / year. 6 month stay time.
- No more then 3 Shuttle-C flights / year for lunar initiative, except to replace lunar vehicle. Lunar crew brought on non-dedicated Orbiter.

RULES:
1. No more than 14 combined Shuttle and Shuttle-C launches year (launches available on 26 day centers).
2. SSF Turnaround time for lunar mission is 121 shifts (MDSSC)
3. Schedule is based on a 6 day / 48 hour work week.

Figure 1-4.1.3-1.   Preliminary Design Reference Scenario (Sheet 2 of 6)

Figure 1-4.1.3-1. Preliminary Design Reference Scenario (Sheet 3 of 6)
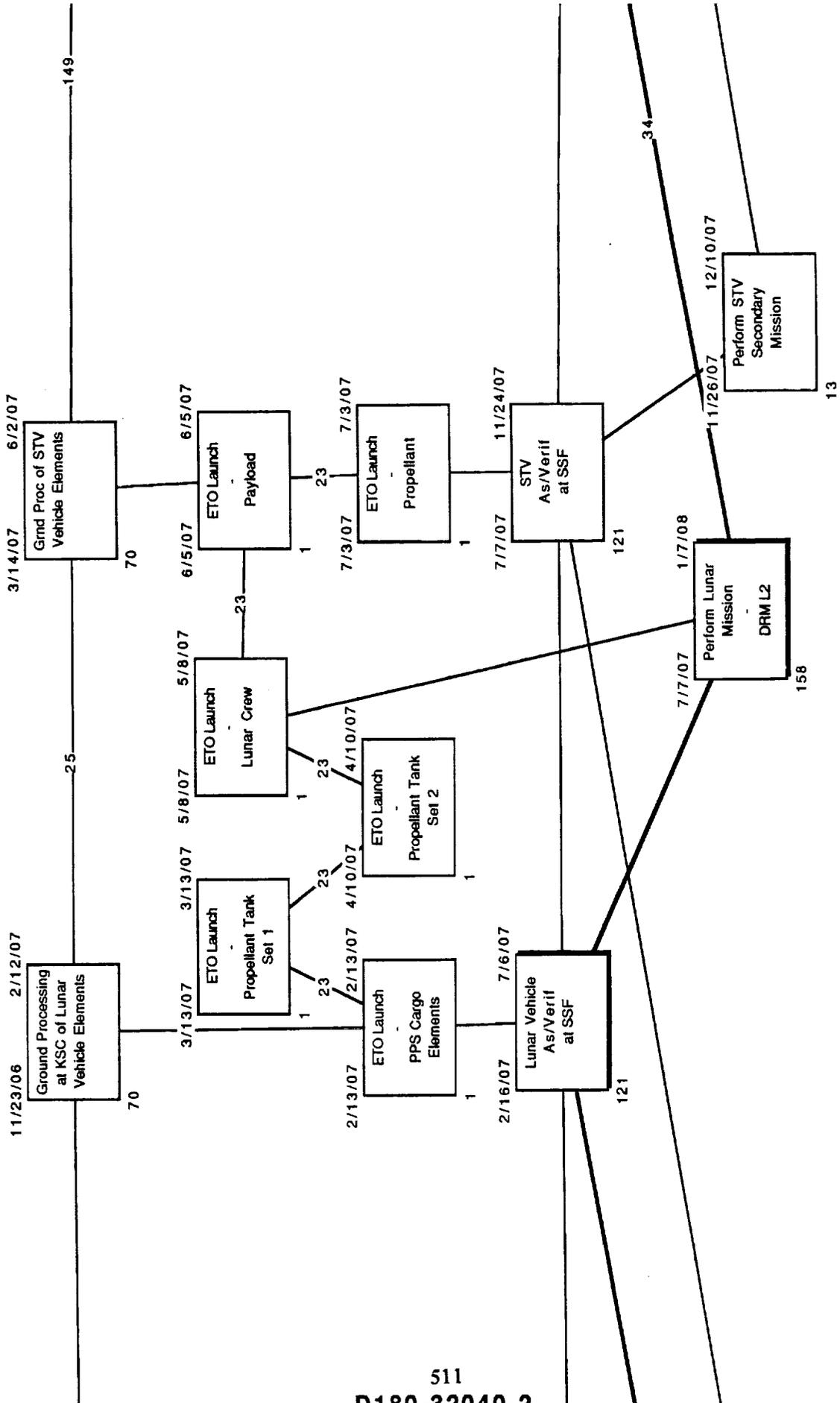
Figure 1-4.1.3-1. Preliminary Design Reference Scenario (Sheet 4 of 6)
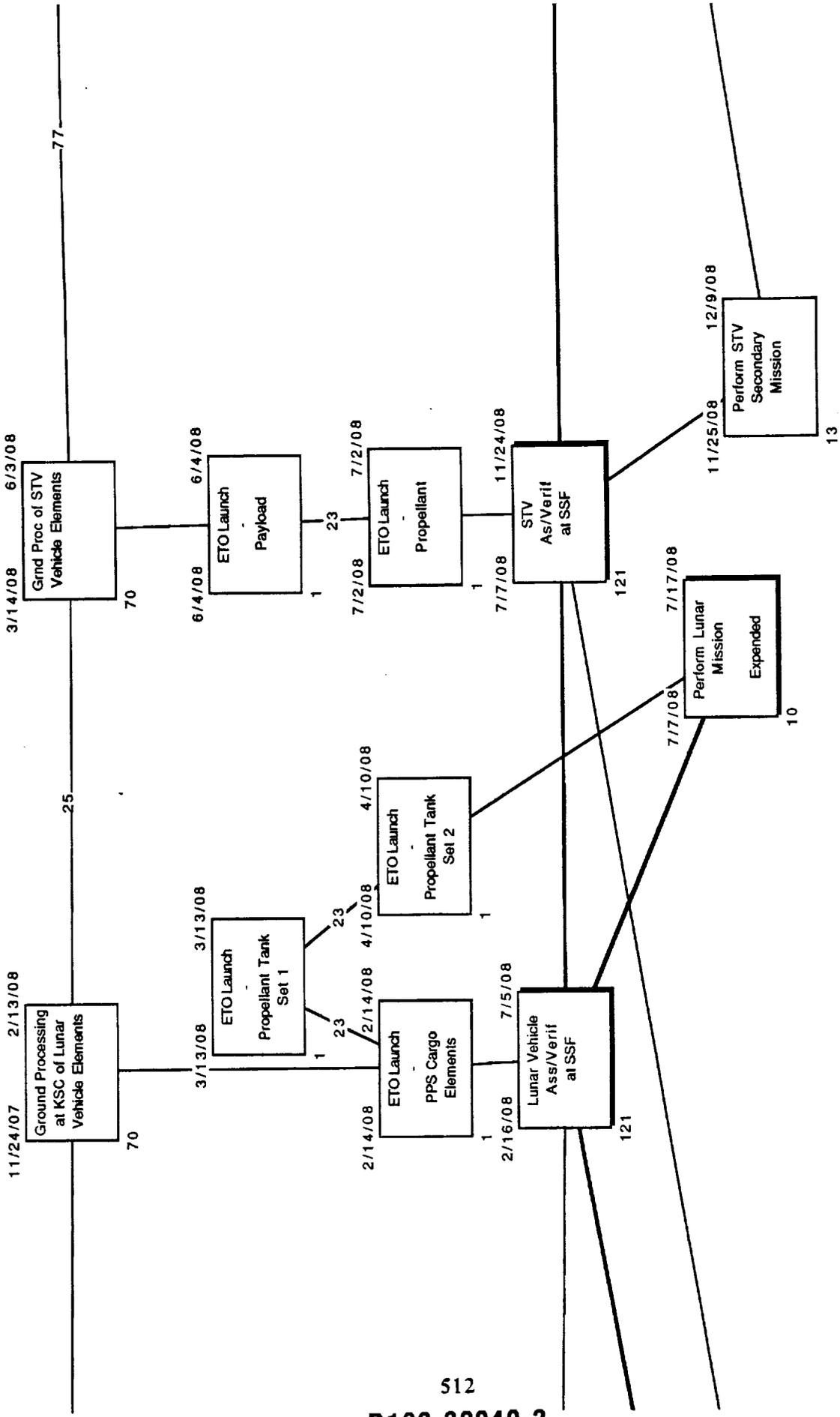
Figure 1-4.1.3-1. Preliminary Design Reference Scenario (Sheet 5 of 6)

BOEING
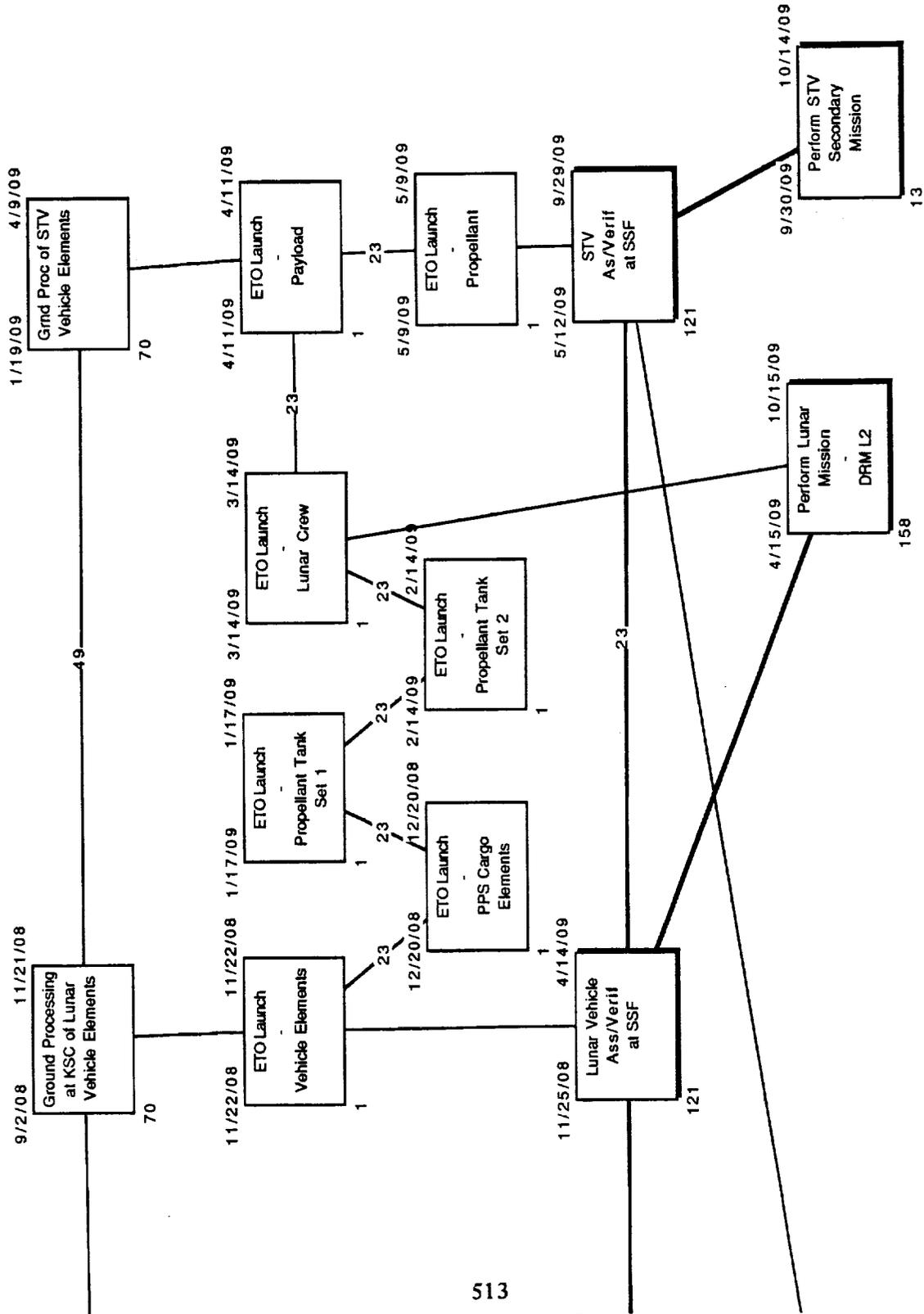
Figure 1-4.1.3-1. Preliminary Design Reference Scenario (Sheet 6 of 6)

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space infrastructure. Starting with DRS2, the second lunar mission per year was replaced with a CNDB mission. The next step was to increase the processing capacity of SSF until the mission model was captured. Work on DRS3 was stopped at this time because of several reasons

First, MacProject II was not capable of handling that number of tasks. We will abandon MacProject II in further complex DRS analysis. Second, the mission model grossly exceeded the Shuttle/Shuttle-C program capabilities. We need to consider a different ETO booster system.

1-4.2   ARCHITECTURE STUDY

Program management decided to perform a wide reaching architecture study in which 95 LTS concepts were defined. Twenty-nine of these were analyzed in detail to provide data for a Taguchi-style analysis. Operations supported this task by defining the processing steps and operations required to perform a complete mission. A distinction between the first mission of a reusable system and a follow-on mission was made where required. The analysis was performed at a level of accuracy that allowed a "building block" approach to building the scenarios. A core stage took a certain number of task-hours to process, regardless of the mission architecture. Adjustments to processing times were made only if the mission profile for that element changed in a major way. An example would be that a stage that performs ascent and descent would be more complex then a transfer stage, and therefore, take longer to process or refurbish. MacProject II was used to analyze the operations required to perform the missions for all 29 concepts. Printouts of these files are in Appendix C. When the analysis was complete, the team elected to downselect to three vehicle concepts. These were SB1-1.5S (space based, 1.5 stage, single crew cab), GB1-1.5S (ground based, 1.5 stage, single crew cab), and GO1-1.5S (ground based with on-orbit self-assembly, 1.5 stage, single crew cab). The top-level MacProject II files are Figures 1-4.2-1 through 1-4.2-3. Copies of the top-level flow are in the Appendix C along with the associated subflows.
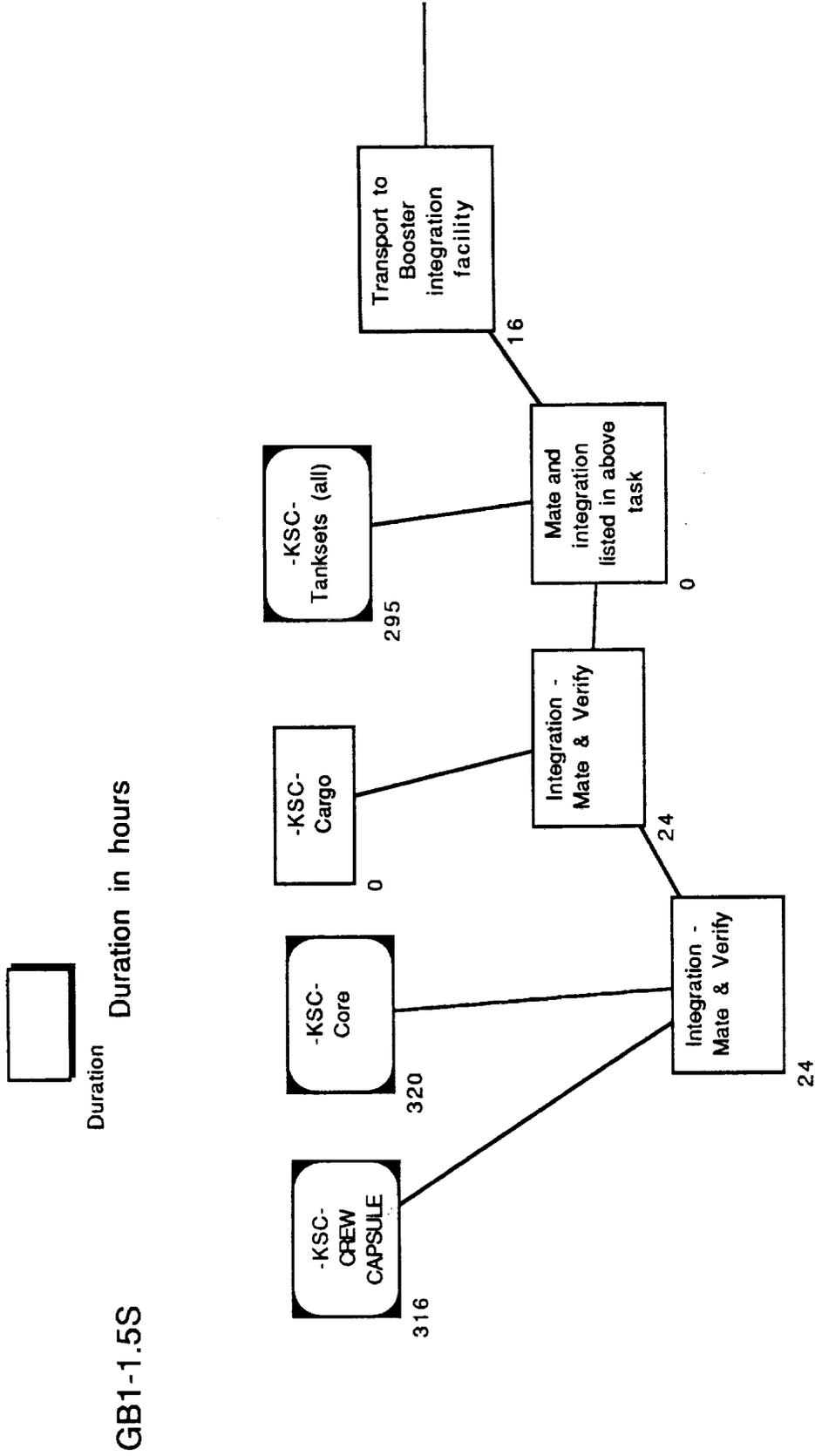
Figure 1-4.2-1. GB1-1.5S Top-Level Process Flow (Sheet 1 of 2)
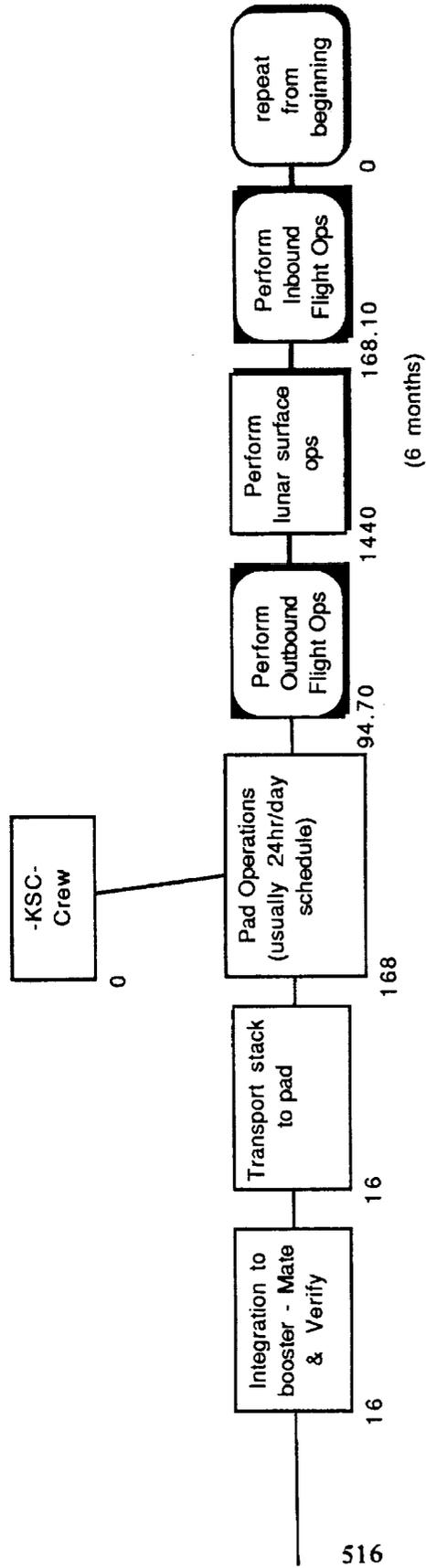
Figure 1-4.2-1. GB1-1.5S Top-Level Process Flow (Sheet 2 of 2)

GOB-1.5S

Duration in hours

Schedule is based on 8-hour shifts, 7 shifts per week.

Figure 1-4.2-2. GO1-1.5S Top-Level Process Flow
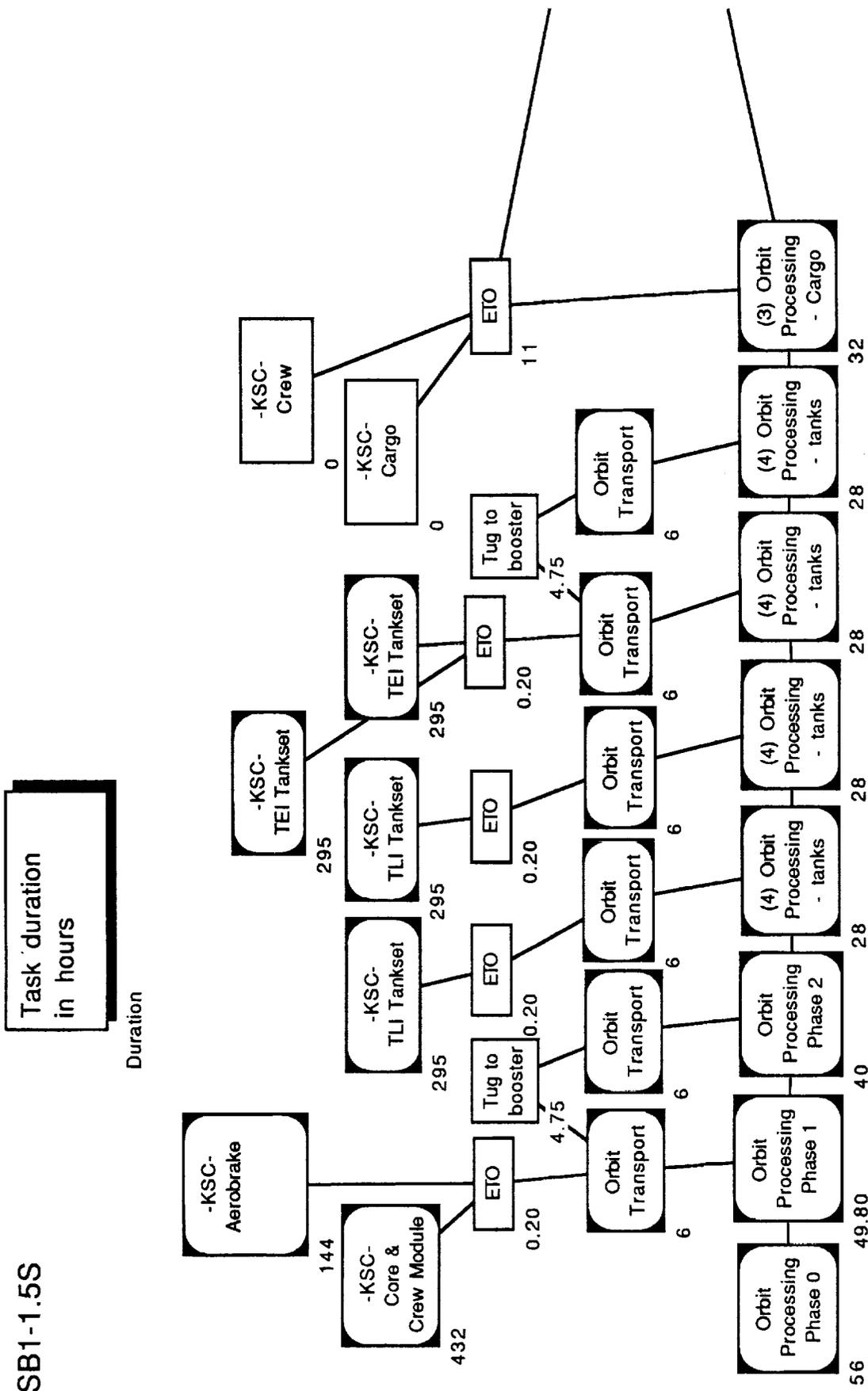
Figure 1-4.2-3. SB1-1.5S Top-Level Process Flow (Sheet 1 of 3)
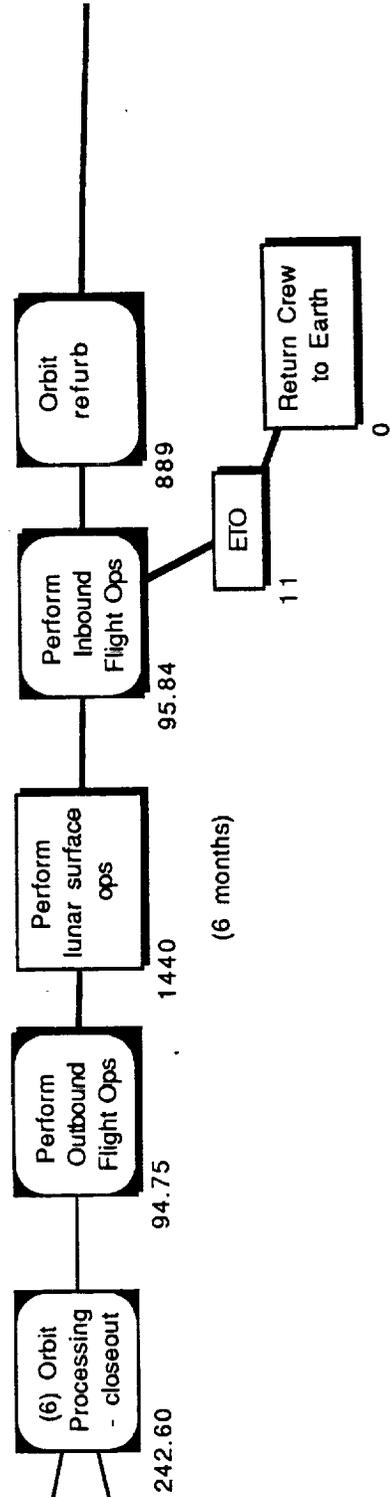
Figure 1-4.2-3. SB1-1.5S Top-Level Process Flow (Sheet 2 of 3)

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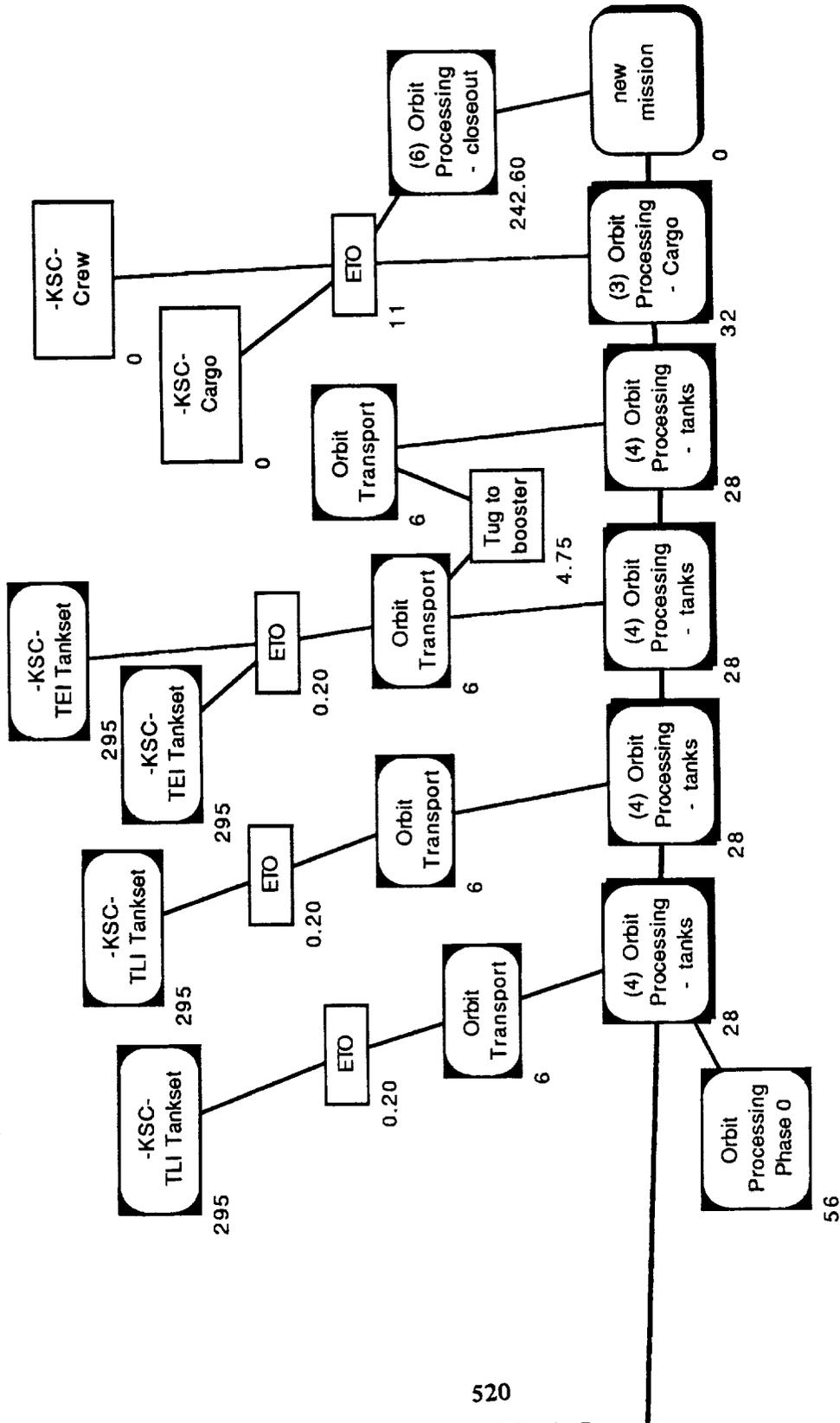
Figure 1-4.2-3. SB1-1.5S Top-Level Process Flow (Sheet 3 of 3)

1-4.2.1 DRS3

DRS3 was developed to analyze the resources and support equipment required to perform the lunar program and all the 89CNDB missions. The mission model is shown in Figure 1-4.2.1-1. Tug missions were added as the STV required them to bring vehicle elements from the ETO boosters to the SSF.

KSC Impacts. The STV would be a reusable core vehicle based at the SSF. It was sized to perform the lunar mission by using droptanks. The core and crew returned to the SSF with the help of an aerobrake. We hypothesized different vehicle elements that would allow this core to capture the 89CNDB missions. These elements are shown in Figure 1-4.2.1-2. Complete vehicles configured for the 89CNDB missions are shown in Figure 1-4.2.1-3.

We determined in the preliminary DRS3 that the Shuttle-C/Shuttle system could not support this mission model. We chose a 70-ton ALS with no launch rate constraints as the supporting booster.

By analyzing the mission model without facility constraints, we hoped to determine the processing rate requirements. First, the vehicle elements required to fly each STV mission were accounted for on a yearly basis. The major elements were propellant and tanks, core vehicle, and aerobrakes. Other elements were not a major factor in driving STV facilities but were accounted for in the summary.

An average of 1,000 tons per year of propellant is required, with an average of 25 tanks per year. Figure 1-4.2.1-4 shows that the majority of the tanks are the in-line set required to support the DoD missions, which are the bulk of the mission model. This equated to three tank processing lines at KSC, with 13 heads touch-labor working full time.

We assumed that a core vehicle could be processed at KSC in 13 weeks. At an average of four cores per year, this equated to one core processing line, with five heads touch-labor, operating full time. There is no margin in this estimate, so to reduce programmatic risk a second line might be needed. Further refinements in processing estimates would be the indicator for this.

STV

MSFC- BOEING

TITLE
Ground Operations Splinter

DRMs

	L1	L2	P1	G1	G2	S1	N1	C1	CONSTRAINED DOD GEO	DOD HEO	est. TUG
2001	1		3			1			7	5	32
2002	1							1	7	5	28
2003			1			1			7	5	30
2004		1	2					1	4	4	24
2005		1		2				2	7	5	34
2006	1				1				5	6	28
2007		1			1	1			5	5	24
2008		1		1	1	1			6	4	28
2009		1		2					7	4	28
2010	1			3			1		5	6	32

Civil = 35   DoD = 109   Total = 144

- Space Tug is estimated
  - Space Based STV based at SSF
  - Booster is not allowed to rendezvous with node
  - Each element requires one round trip by tug

Figure 1-4.2.1-1.   Ground Operations Splinter

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NON-LUNAR VEHICLE ELEMENTS

Interface Issues: DRS-3

STV

MSFC- BOEING

Small Aerobrake

- **Smaller then manned system brake**
- **May fit in well with vehicle evolution**
- **Used in 88.3% of mission model**

Upgraded Flight Telerobotic Servicer
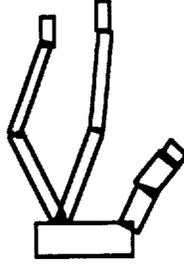
- **Longer arms**
- **Hardened to survive aeromaneuver**
- **Required on all servicer missions**

In-Line Tank Set
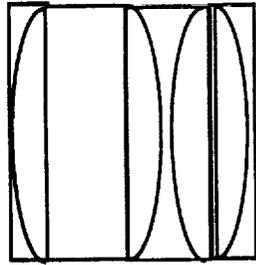
- **Load bearing structure**
- **Launched wet from KSC**
- **Allows optimum Lunar vehicle core use in 89% of mission model**

33.5 ton capacity

Figure 1-4.2.1-2. Non-Lunar Vehicle Elements

STV

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VEHICLES FOR SERVICING MISSIONS

Interface Issues: DRS-3

GEO Servicing (G2)

Propellant Required: 79.5t

Capacity Available: 97.4t

Mission Duration: 15.6 hrs + service

% of Mission Model: 2.1%

LEO Polar Servicing (S1)

Propellant Required: 133.5t

Capacity Available: 130.9t

Mission Duration: 110.4 hrs + service
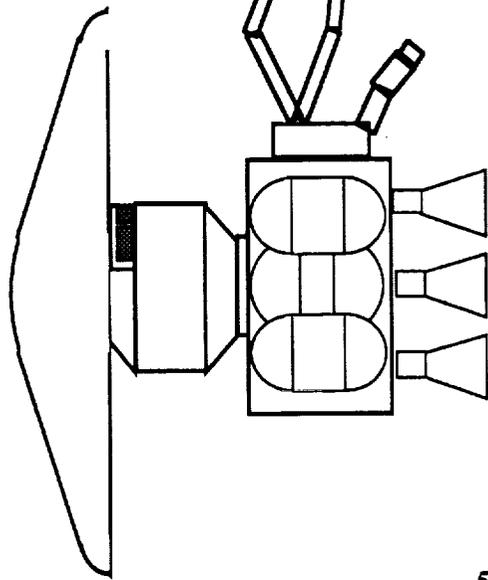
% of Mission Model: 2.8%

Sample Capsule Recovery (C1)

Propellant Required: 19.3t

Capacity Available: 20.8t

Mission Duration: 24.4 hrs

% of Mission Model: 2.8%

Figure 1-4.2.1-3. Vehicles for Servicing Missions (Sheet 1 of 2)

VEHICLES FOR UNMANNED MISSIONS

Interface Issues: DRS-3
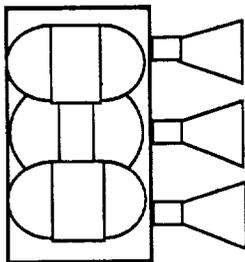
Tug Missions (T1)
Propellant Required: 0.5t

Capacity Available: 20.8t

Mission Duration: 13.8 hrs

Modeled at ≈ 30 missions/yr
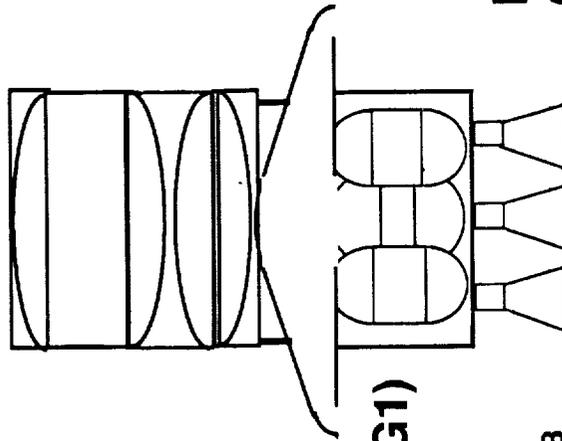
Geo delivery (G1)
Molniya (D1)
Propellant Required:
   G1..54.3t D1..53.8

Capacity Available: 54.3t

Mission Duration:
   G1..39.6hrs D1..75.7hrs

% of Mission Model: 81.3%
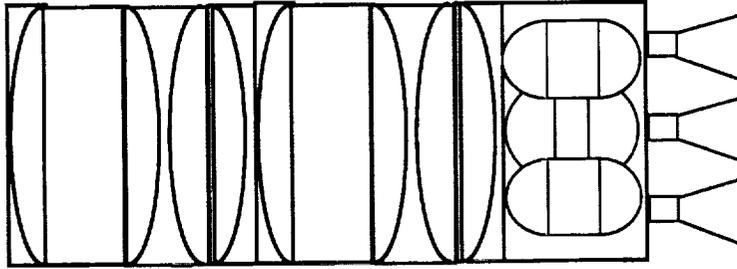
Nuclear/Debris disposal (N1)
Propellant Required: 70.4t

Capacity Available: 87.8t

Mission Duration: 200 days

% of Mission Model: 0.7%
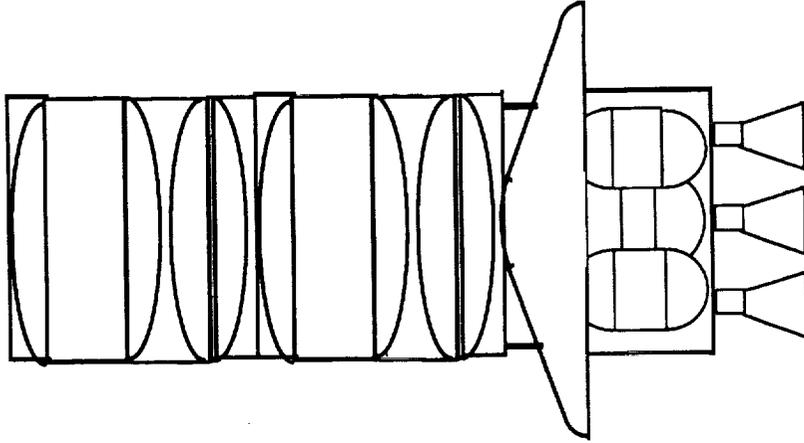
Planetary Delivery (P1)
Propellant Required: 75.2t

Capacity Available: 87.8t

Mission Duration: 357.7 hrs

% of Mission Model: 4.2%

Figure 1-4.2.1-3. Vehicles for Servicing Missions (Sheet 2 of 2)

TANKSET PROCESSING RATE

Interface Issues: DRS-3

STV

BOEING

NUMBER OF TANKS (BY TANK TYPE)

In-line tank set

Descent tank set

TLI tank set

Tank Farm tank set
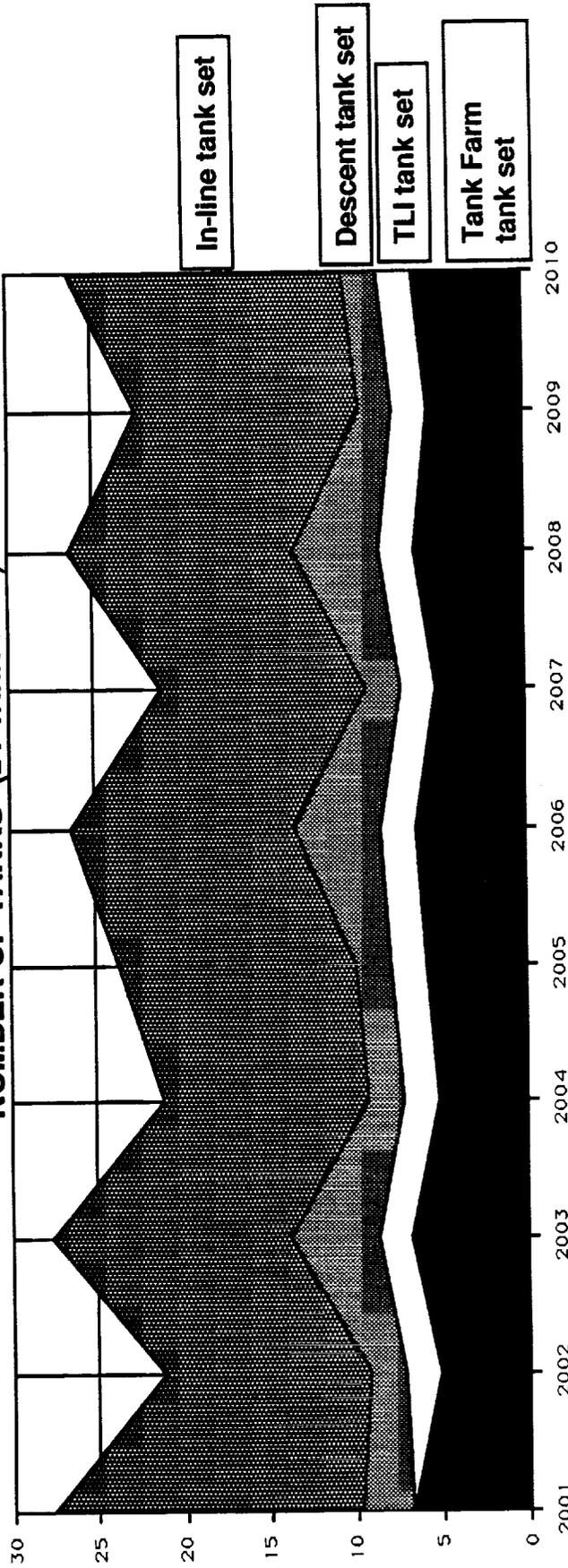
- Processing must handle an average of 25 tanks/year
- Using current tank processing methods and procedures:
  - 30 - 8 hour shifts / tank
  - 1000 man-hours per tank

3 tank processing lines operating in parallel

Figure 1-4.2.1-4. Tankset Processing Rate

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Because all core vehicles have aerobrakes four new aerobrakes are required each year also. We estimated 2 weeks for processing, with five heads touch-labor, working 16% of the year. It should be noted that at this point in the design that the aerobrake was structure and TPS, with no avionics or propulsion subsystems, or doors and penetrations.

Personnel for support functions was estimated using factors, with a result of 69 personnel required.

A new STV processing facility is needed based on the above data and the quantity of elements being processed, the size of the elements, and the nature of testing done to cryogenic vehicles. Processing rates are summarized in Figure 1-4.2.1-5. A second issue with the processing facility is the content of the mission model: 80% is military payloads. The design of the processing scheme and the facility must account for the military requirements.

Figure 1-4.2.1-6 summarizes the processing rates of the elements processed at KSC. The data shown are an average yearly rate, and related STV elements are also shown.

SSF Impacts. Based on the on-orbit processing definition work performed for the architecture trade study, we created a table of task-hours similar to the mission model table. This is shown in Figure 1-4.2.1-7. As a gross estimate, we determined that a team was capable of performing 2,500 task-hours per year with personnel rotation in and out of the team. This was not discounted for EVA overhead time. This requires five full-time teams based at the SSF to process all of the STVs.

The assembly area must be sized to accommodate the vehicle plus enough margin to manipulate elements during the assembly process. The assembled lunar vehicle is approximately a 24.4m- by 15m-diameter cylindrical shape. Other vehicles may be processed in a shorter area, but the aerobrake sizes the diameter. Analysis indicates that the assembly facility should be sized with two areas for the lunar vehicle plus two shorter areas for DoD missions.

BOEING

STV

DRS 3: STV PROCESSING FACILITY
Ground Operations Splinter

MSFC- BOEING
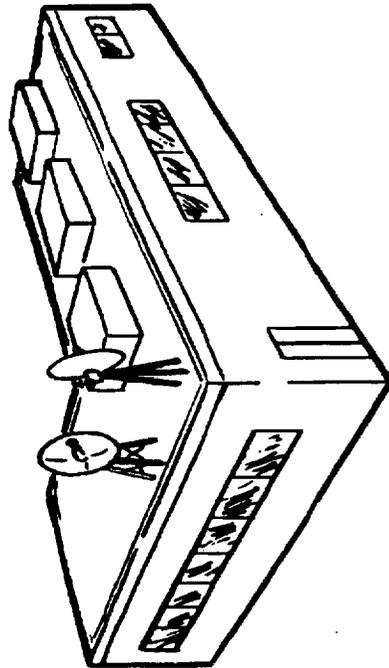
- 3 tank processing lines
  - 13 heads (average)
  - Full time
- 1 core processing line
  - 5 heads (average)
  - Full time
- Aerobrake & Crew Module area
  - 5 heads (average)
  - 16% time

- Related Element Processing (optionally in this facility)
- Receiving and inspection area, and element activation area
- Spares and support equipment storage and servicing area
- Offices and personnel for support functions
- Approximated at 3 times the hands-on crew
  - 69 support personnel

Figure 1-4.2.1-5. DRS3 - STV Processing Facility

BOEING

STV

MSFC- BOEING

DRS-3 KSC IMPACTS: YEARLY RATES

Ground Operations Splinter

Yearly rates based on average over 10 year period

Manned Flights
- 1.2 Manned flights
  - Peaks at 3 in 2005

ALS payloads
- Support 14.4 STV missions
- Estimate 20 to 30 HLLV flights per year

Stage Processing Facility
- 4 vehicles
- 25 tanksets
- 3 Aerobrakes
- 0.2 Crew Modules

Payloads Processing
- 12.3 S/C
  - 1.4 Civil
  - 10.9 DoD
- 10t to 30t Lunar Cargo
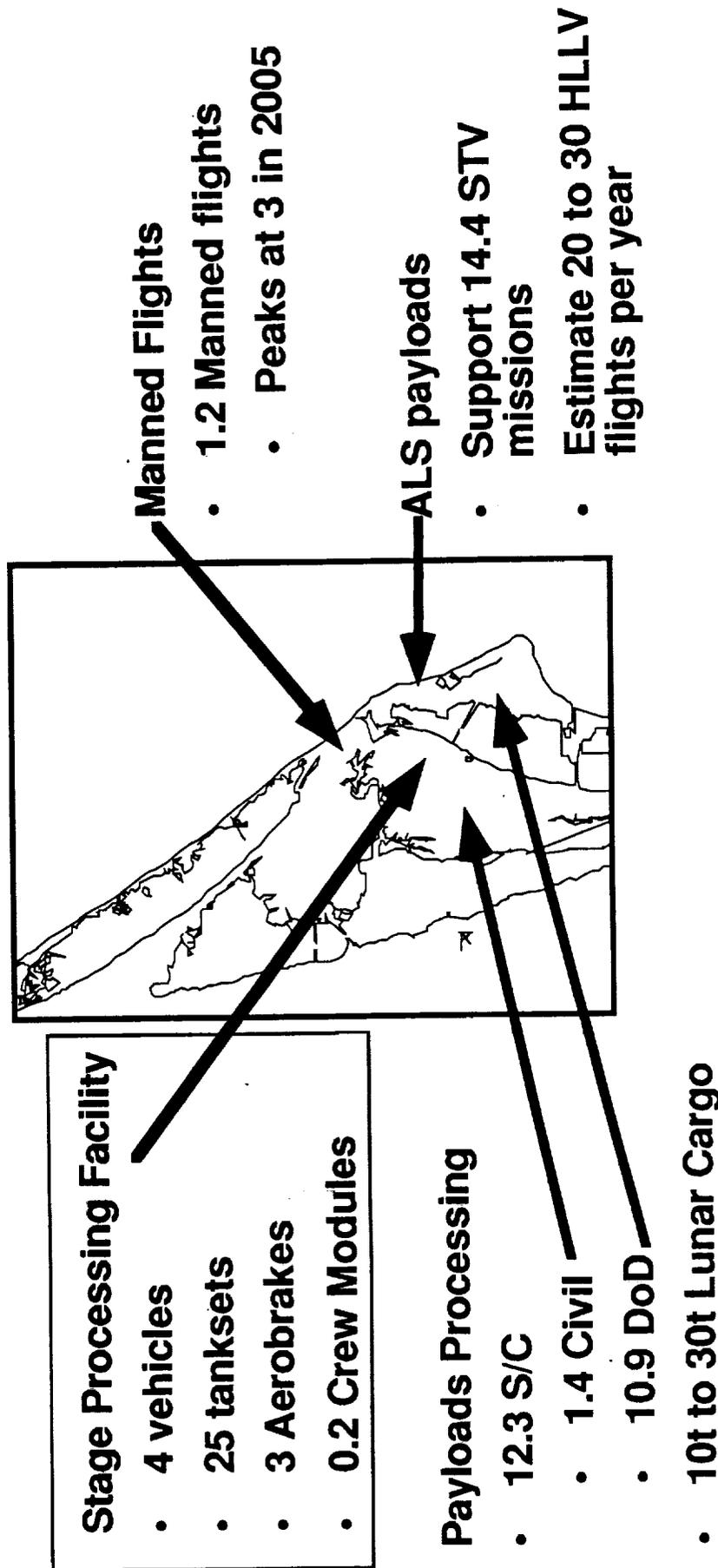
Figure 1-4.2.1-6. DRS3 KSC Impacts (Yearly Rates)

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STV TASK HOURS PERFORMED AT SSF

Interface Issues: DRS-3

MSFC - BOEING

	TASK HOURS REFURBISHMENT AND ASSEMBLY								DOD GEO	DOD HEO	DOD TUG	Yearly TOTAL
	L1	L2	P1	G1	G2	S1	N1	C1				
2001	0	0	2340	0	0	810	0	0	5250	3750	0	10692
2002	840	0	0	0	0	0	0	1190	5250	3750	0	9706
2003	840	0	780	0	0	810	0	0	5250	3750	0	10058
2004	0	1310	1560	0	0	0	0	1190	3000	3000	0	8853
2005	0	1310	0	1500	0	0	0	2380	5250	3750	0	12487
2006	840	0	0	0	1250	810	0	0	3750	4500	0	9812
2007	0	1310	0	0	1250	0	0	0	3750	3750	0	8853
2008	0	1310	0	750	1250	810	0	0	4500	3000	0	10226
2009	0	1310	0	1500	0	0	0	0	5250	3000	0	9733
2010	840	0	0	2250	0	0	680	0	3750	4500	0	10578

- **A team is capable of 2500 task hours / year**
  - **8 hours / day, 6 days / week**
    - **Not discounted for EVA preparation time**
- **Requires 4 to 5 full time teams**
- **Substantial growth of SSF**
  - **Triple planned logistics support**

Figure 1-4.2.1-7. STV Task-Hours Performed at SSF

In addition to the assembly area, storage areas also are required. Two tug "garages" are required. The tug is a 9.1m- by 9.1m-diameter cylinder. Storage of orbit replaceable units (ORU) will be required.

The operations area must include habitation for 25 people, 8 to 10 teleoperation and support stations, and a support area for the lunar crew and other manned STV flights. We determined that refurbishment of the crew module could best be done if the module was separated from the STV and berthed to the stations habitable area. Therefore, the operations area would require a dedicated crew module servicing hatch, along with the resources to service the crew module.

A propellant farm has hypothesized to refuel the reusable core. This farm would accommodate up to two 70-ton tanks at a time.

1-4.2.2   DRS4

Goal. After seeing what type of space station was required for DRS3, we returned to the basic requirement and reexamined the rational for space basing. In DRS3, the core vehicle was designed to be part of the lunar transportation system. In our architecture study that looked at different LTSs, we compared space based/reusable to ground based/expendable. Space basing was never decoupled from reusability in our vehicle concept analysis. We needed a way of determining the best method of capturing the civil missions within the budget and time constraints of the study. DRS4 was an effort to capture the trend of this concept from a system behavior point of view.

There are two technical reasons for choosing LEO as the basing point. It can provide a base to support activities associated with reusing the vehicle, such as refurbishment, refueling, and payload integration. LEO basing also decouples the size of the completed vehicle from the size of the ETO booster.

Assumptions. For DRS4 we assumed that the nation would not be willing to build a booster in which SEI is the only identified customer. We used the ALS designed to support military and civil payloads. As stated above, we didn't believe that space-based reusability had been proven to be cost effective. Therefore, the following rules were used to develop DRS4:

1. All missions are ground based, except missions whose initial mass in LEO is greater then 60 tons (booster limit less 15% for ASE).

2. All equipment is expendable, except crew modules and the upgraded flight telerobotic servicer. These items are of very high value compared to other mission elements; are usually returned to the SSF as part of the mission design, regardless of reuse; and can be serviced IVA by either bringing inside (FTS) or docking to a port (crew module), potentially reducing the cost of space-based refurbishment.

Figure 1-4.2.2-1 is the mission model used for DRS4. The only part that changed from DRS3 is the estimated tug missions. Since the tug is not in the 89CNDB, we only included those missions that the STV would require in bringing elements from the ETO booster to the SSF.

KSC Impacts. As in DRS3, the core vehicle is sized for the lunar mission as shown in Figure 1-4.2.2-2. Additional vehicle elements were conceptualized that would aid the core in performing the 89CNDB missions. These are shown in Figure 1-4.2.2-3. Using the mission model along with the conceptual vehicle designs, we determined the processing rates and supporting resources on a yearly bases.

Propellant consumption was reduced by an average of 45% because of the redesigned mission definition. Because not all missions returned to the SSF for vehicle reuse, the propellant to bring them back was not required. Tank processing rates were also reduced to an average of 18 tanks per year, as shown in Figure 1-4.2.2-4. This equated to two tank processing lines operating at near full time.

An average of 15 core vehicles per year are required to perform all missions in an expendable mode. This requires four core processing lines operating at 88% capacity, with 12 weeks per vehicle as the processing time.

The requirement for aerobrakes is greatly reduced compared to DRS3. For DRS4 only manned missions and the polar servicer missions have aerobrakes to assist the vehicle in returning to the SSF. (Return trajectories using the propulsion system instead of the aerobrake were not considered for this

STV

MISSION MODEL
DRS-4 (Point Design)

MSFC - BOEING

BOEING

	L1	L2	P1	G1	G2	S1	N1	C1	DOD GEO	DOD HEO	est. TUG
2001			3						7	5	1
2002	1								7	5	4
2003	1		1			1		1	7	5	5
2004		1	2					1	4	4	4
2005		1		2		1		2	7	5	4
2006	1				1				5	6	6
2007		1			1				5	5	5
2008		1		1	1	1			6	4	6
2009		1		2					7	4	4
2010	1			3			1		5	6	4

DBMs

Civil = 35

DoD = 109

Total = 144

533

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- Civil Needs Data Base and customer direction provides us with the mission model, except tug
  - Space Tug is estimated
    - Booster is not allowed to rendezvous with node
    - Each STV vehicle element for a mission requiring on orbit assembly (FAT SAT) requires one round trip by tug

Figure 1-4.2.2-1.   Mission Model

BOEING

LUNAR VEHICLE CONCEPT
DRS-4 (Point Design)

STV

MSFC- BOEING

TLI Tankset (2): 64.8t propellant Staged after TLI burn
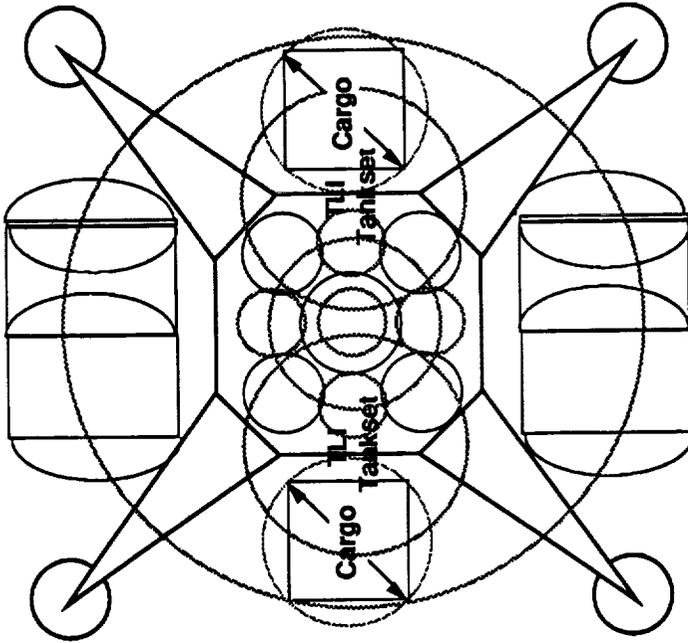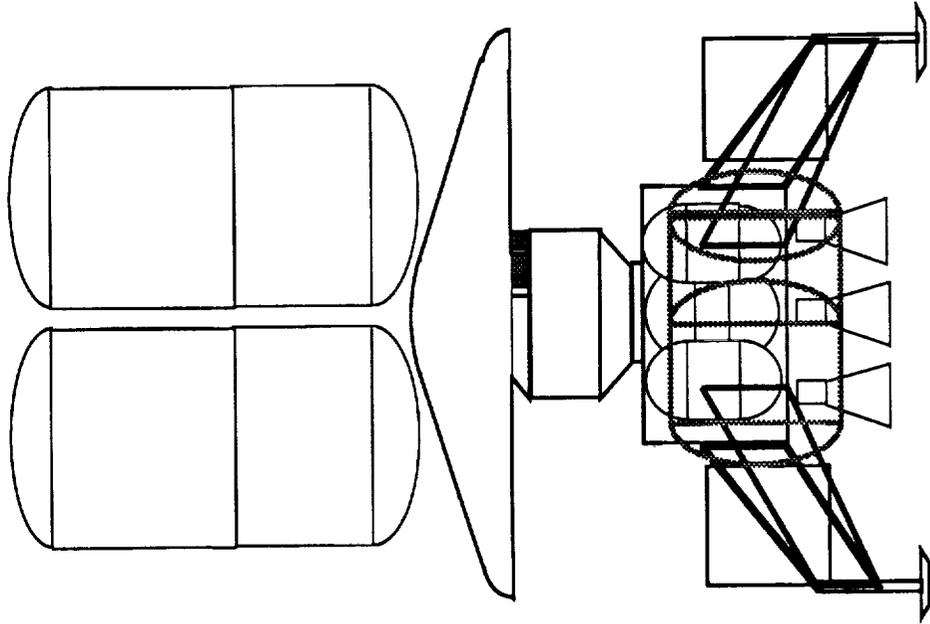
Aerobrake: Structural and propellant line penetrations

Core: 20.8t propellant

**Descent Tankset (2): 38.3t propellant Dropped on Lunar Surface**

Figure 1-4.2.2-2. Lunar Vehicle Concept

NON-LUNAR VEHICLE ELEMENTS

DRS-4 (Point Design)
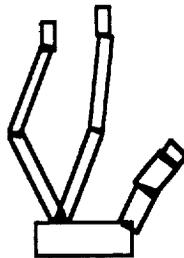
STV

Upgraded Flight Telerobotic Servicer

- Longer arms
- Hardened to survive aeromaneuver
- Required on all servicer missions

In-Line Tank Sets

- Load bearing structure
- Launched wet from KSC
- Allows optimum Lunar vehicle core use in other missions
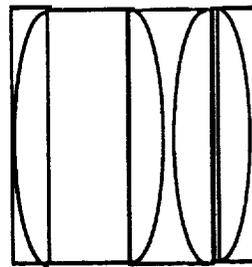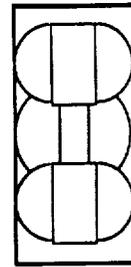
9.1 ton capacity

33.3 ton capacity

Figure 1-4.2.2-3. Non-Lunar Vehicle Elements

TANK PROCESSING RATES
DRS-4 (Point Design)

STV

MSFC- BOEING

BOEING
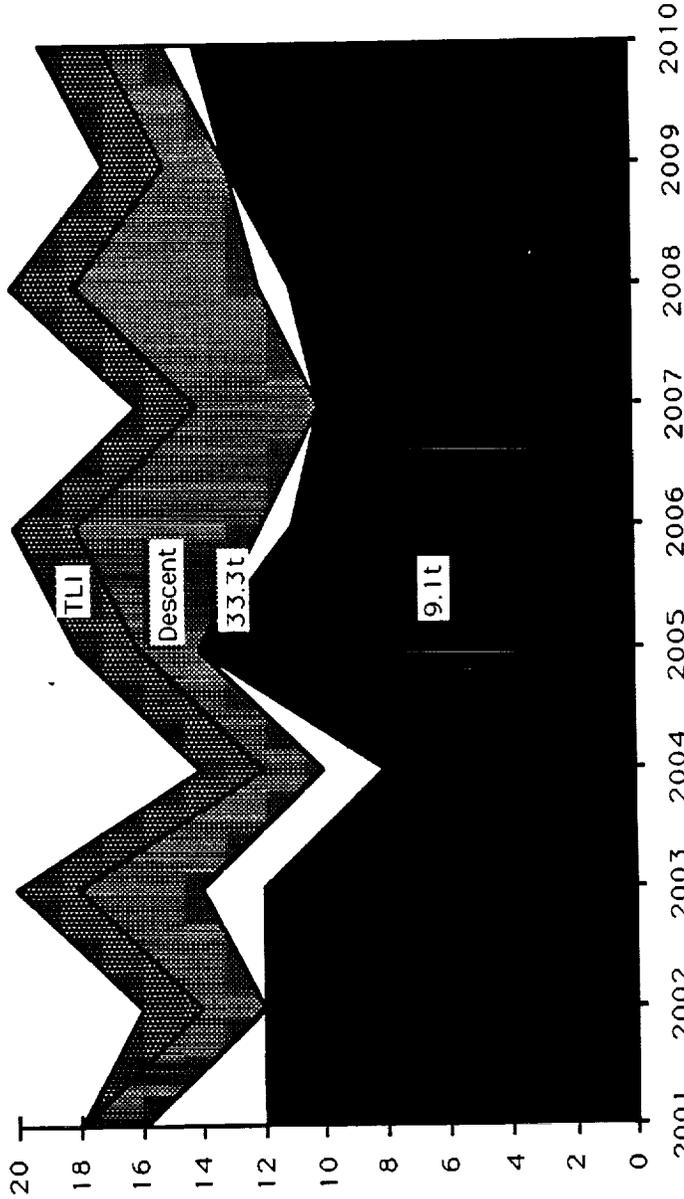
- Processing must handle an average of 18 tanks/year
- 30 - 8 hour shifts/tank, 4 person hands-on team
- Maintain full capacity through low years to buffer high years

2 tank processing lines operating in parallel

Figure 1-4.2.2-4. Tank Processing Rates

analysis.) This low usage equates to three new aerobrakes in 10 years if each aerobrake is used to perform 5 missions.

Just like DRS3, a new STV processing facility is needed based on the above data and the quantity of elements being processed, the size of the elements, and the nature of testing done to cryogenic vehicles. Processing rates are summarized in Figure 1-4.2.2-5. The fact that 80% of the mission model is military missions is still a concern for ground processing. The design of the processing scheme and the facility must account for the military requirements.

Figure 1-4.2.2-6 summarizes the processing rates of the elements processed at KSC. The data shown are an average yearly rate.

SSF Impacts. We used the on-orbit processing definition work done for the architecture trade study for estimating the amount of work required to be performed at the SSF. Figure 1-4.2.2-7 shows both the space-based mission model and the description of the types of facilities identified.

An average of two missions per year are flown out of the SSF per year, and as shown in DRS2, this is within the capability of a single assembly area. Storage area for the tug will be required to provide debris and thermal protection between missions. Storage for aerobrakes, crew modules, servicer front ends, and spares are also required. Additional resources from the SSF are required that go along with supporting two 3-person crews. The assembly area and possibly the storage areas will require power, telemetry, and thermal support. The stations external manipulators will be required to support the assembly and servicing tasks. A large RMS capable of manipulating up to 70 tons will be required to place vehicle elements in the assembly area. Remote manipulators may be required in the assembly area and storage areas, depending on the type of tasks performed. Interfaces between vehicle elements will be made, and the type of tasks required of the manipulators will depend on the interface design. If the servicer front end is an upgraded flight telerobotic servicer, it could be used between its STV missions to service both the STV and the SSF.

STV

DRS 4: STV PROCESSING FACILITY
Ground Operations Splinter

- 2 tank processing lines
  - 8 heads (average)
  - Full time
- 4 core processing line
  - 16 heads (average)
  - 88% time
- Aerobrake & Crew Module area
  - 5 heads (average)
  - 6% time
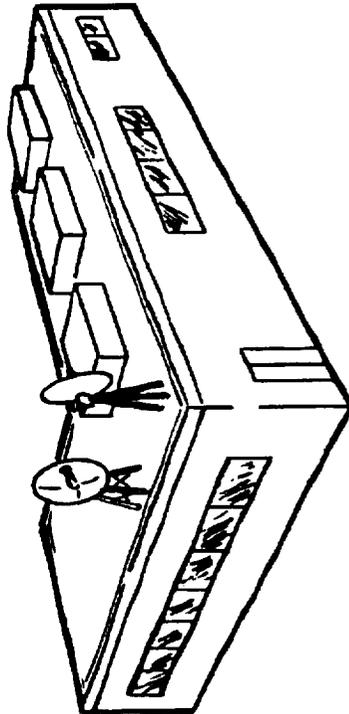
- Related Element Processing (optionally in this facility)
- Receiving, inspection and activation area
- Spares and support equipment storage and servicing area
- Offices and personnel for support functions
  - Approximated at 3 times the hands-on crew
    - 87 support personnel

Figure 1-4.2.2-5. DRS4 - STV Processing Facility

STV

MSFC- BOEING

DRS-4 KSC IMPACTS: YEARLY RATES

Ground Operations Splinter

Yearly rates based on average over 10 year period

Manned Flights
- **1.2 Manned flights**
  - Peaks at 3 in 2005

ALS payloads
- **Support 14.4 STV missions**
- **Estimate 20 to 30 HLLV flights per year**

Stage Processing Facility
- **15 vehicles**
- **18 tanksets**
- **0.3 Aerobrake**
- **0.2 Crew Modules**

Payloads Processing
- **12.3 S/C**
  - 1.4 Civil
  - 10.9 DoD
- **10t to 30t Lunar Cargo**
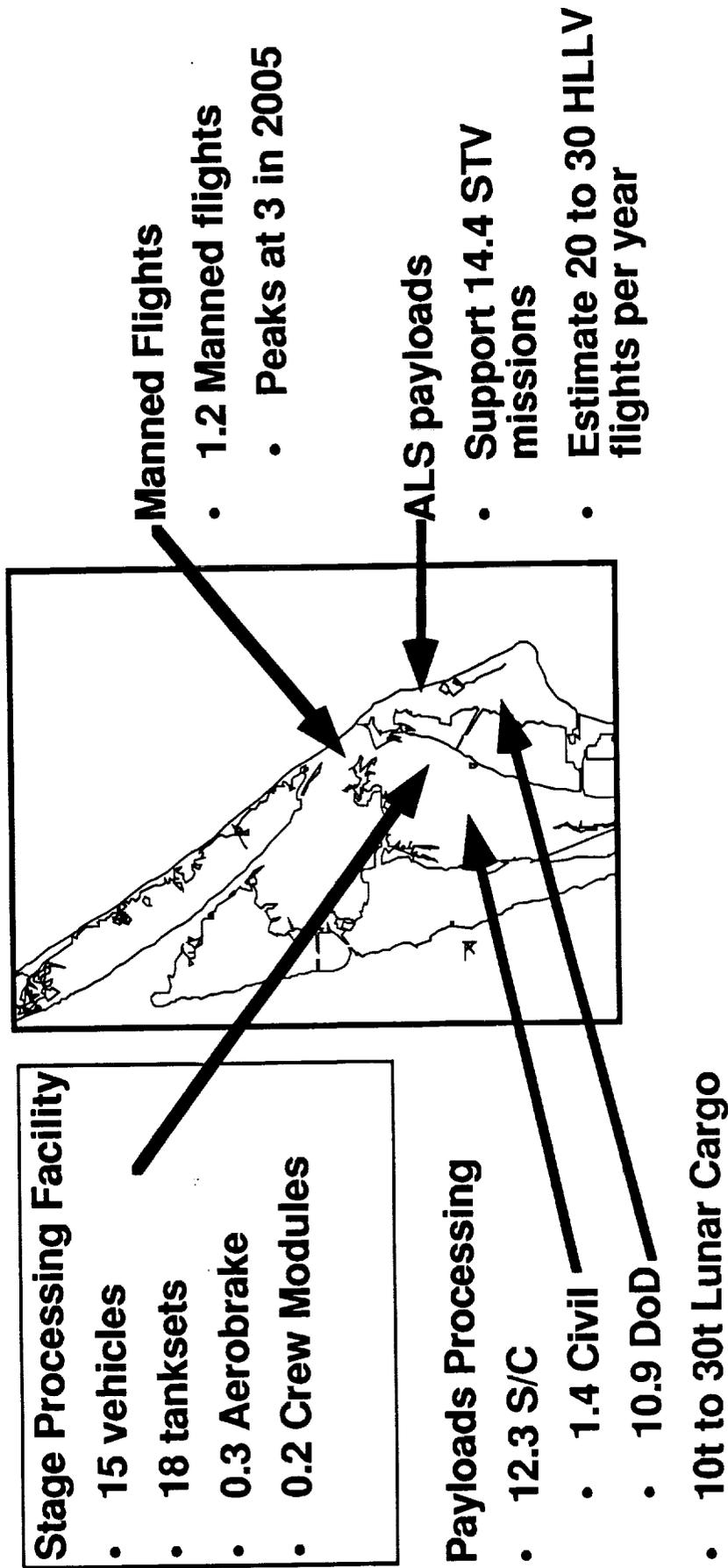
Figure 1-4.2.2-6. DRS4 - KSC Impacts (Yearly Rates)

STV

DRS 4: SPACE STATION / STV FACILITIES

On-Orbit Assembly Splinter

MSFC- BOEING

ASSEMBLY AREA

- Lunar vehicle sizes area volume

- Vehicle is 80' long X 50' diameter cylinder (24.4m X 15m) fully assembled

- Crew module refurbished IVA while attached to main habitation areas

TUG STORAGE

- Core is 30'X30' cylinder (9.1m X 9.1m)

- Debris protection

- Thermal: cryo storage and "sleeping" avionics

- Power

- Health monitor

Mission model of vehicles assembled or stored at SSF			
		total	
	Lunar	Servicer	Tug
2001	0	1	1
2002	1	1	4
2003	1	1	5
2004	1	1	4
2005	1	2	4
2006	1	2	6
2007	1	1	5
2008	1	2	6
2009	1	0	4
2010	1	0	4

OPERATIONS AREA

- Habitation for crew

- Teleoperation station for tug and for "arms" in assembly area

- Docking port for crew module

- Resources for CM refurbishment

STORAGE AREA

- Volume and services to store mission elements:

  - 1 to 2 crew modules

  - 1 to 2 aerobrakes

- Servicer front end (Upgraded FTS)

- Spares, tools and consumables

Figure 1-4.2.2-7. DRS4 - Space Station and STV Facilities

1-4.3   LESSONS LEARNED

1-4.3.1   Reusability

Compared to performing standard upper stage missions with an expendable ground-based upper stage, a space-based reusable upper stage is not competitive because of the risks and costs associated with returning and refurbishment.

Costs include increased ETO costs (return propellant weighs more then stage); the recovery system; recovery operations; refurbishment operations; and design costs associated with reusability.

Risks include a longer mission timeline (return flight) and recovery operations.

At first look one would expect a savings in ETO costs. After all, the stage is not boosted to LEO every time. However, the extra propellant to perform the return maneuver actually weighs more then the stage. For example, the GEO delivery mission (DRM G1) requires 27 tons of propellant to deliver the payload and 26 tons of propellant to return the stage to the SSF using an aerobrake. The dry weight of the stage is less the 10 tons. So in this case, reusability has the penalty of boosting an extra 16 tons per mission.

Recovery systems are hardware subsystems that are not required on an expendable upper stage and are directly additive to hardware costs. Recovery operations are a series of tasks not performed with an expendable stage and are additive to operations costs. Refurbishment operations replace assembly and checkout tasks of a new upper stage. If the vehicle is designed with refurbishment operations in mind, this can be a savings instead of a cost when compared to an expendable stage.

The potential benefits of reusability in a cryogenic upper stage should be demonstrated as attainable and as capable of outweighing the costs and risks. The primary potential benefit is to reduce cost through reduced hardware acquisition. This can be done in two ways. First is to reduce the cost of procuring a stage. This effects both sides of the reusability equations. Both

expendable stages and reusable stages would benefit from reducing hardware costs. The second method is to increase the number of flights that an individual stage can perform in its life.

1-4.3.2   Space Basing

There are several reasons for basing an upper stage in space. It helps in reusability by saving the cost of reboosting the stage every flight. It also decouples the size of the mission from the maximum capability of the ETO vehicle. Reusability was previously discussed.

The initial mass in LEO for a lunar mission is approximately 250 tons. This greatly exceeds the capability of any ETO booster required for other civil or military missions. It can also be argued that as soon as you build a booster big enough for one mission, some one will design another mission or station that's even bigger. At some point it is more economical to boost the payload in pieces and assemble them in LEO. The more efficient the assembly process, the more the breakeven point moves toward a smaller booster. The exact determination of this breakeven point is beyond the scope of this study.

DRS4 had 14% of the mission model based at the SSF, with the other 86% flown as ground based. As mentioned in DRS4, the facility at the SSF was a single assembly area with an average of two missions per year. DRS2 demonstrated that this was the maximum that could reasonably be processed at such a facility. DRS2 was the full refurbishment, integration, and checkout of two lunar vehicles. In DRS4, the two missions only involve vehicle integration and refurbishment of the crew module and aerobrake. Therefore, DRS2 is actually a conservative estimate when compared to DRS4, inspiring confidence that such a facility could realistically perform DRS4.

Space-Based Nodes. Basing at the SSF has problems associated with each aspect. None are insurmountable, but they all need to be addressed. The presence of the large debris shield used for vehicle storage, assembly, and testing increases station reboost requirements. Presence of a fully fueled lunar vehicle puts the habitable areas out of specification for the 5-μg environment. The quantity of cryogenic propellant may cause safety concerns because of the

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possibility of a leak, propulsive puncture, or explosive burst. Assembly activity and increased human presence may cause sufficient vibration accelerations to ruin the microgravity experiments.

We examined different locations for performing different activities in LEO. One option considered all activities being performed at the SSF, and as reported above, considerable work was done in analyzing this option. The other end of the spectrum is to use the vehicle as a node by itself. The vehicle concept that did this was called GO for ground based, on-orbit assembly.

GO originally was up to six pieces that were launched separately. The core stage would rendezvous with the next booster as it arrived in LEO and dock with the vehicle element that was the payload. This required a relatively small booster of the 70-ton class but did not require the SSF. The problem with this concept was that it required self-making, self-verifying interfaces and contained a high degree of program risk should something go wrong in the assembly process.

The GO concept evolved until it required only two 130-ton boosters. One booster launched most of the oxygen in a tanker and the second one launched the vehicle. The vehicle docked with the tanker, took on the oxygen, discarded the tanker, and proceeded to the moon.

No analysis was done on how this vehicle would capture the non-lunar missions described in 89CNDB, but it does demonstrate another way of assembling vehicles in LEO that are too big to launch on a single booster. The next step in the study could have been to look at a hybrid space-basing mode. A possible scenario might be to perform crew module refurbishment, crew rotation, and element storage at the SSF as described in DRS4. The remainder of the tasks performed in DRS4 at the station would be moved off station in a "self-node" mode like in GO.

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16. Abstract

This report presents the results of systems analyses and conceptual design of space transfer vehicles (STV). The missions examined included piloted and unpiloted lunar outpost support and spacecraft servicing, and unpiloted payload delivery to various earth and solar orbits. The study goal was to examine the mission requirements and provide a decision data base for future programmatic development plans. The final lunar transfer vehicles provided a wide range of capabilities and interface requirements while maintaining a constant payload mission model. Launch vehicle and space station sensitivity was examined, with the final vehicles as point designs covering the range of possible options. Development programs were defined and technology readiness levels for different options were determined. Volume I presents the executive summary, Volume II provides the study results, and Volume III the cost and WBS data.

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