SPACE TRANSFER VEHICLE CONCEPTS AND REQUIREMENTS

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FOREWORD

This report, prepared by Martin Marietta Corporation, is submitted to George C. Marshall Space Flight Center, National Aeronautics and Space Administration (NASA), Marshall Space Flight Center (MSFC), Alabama, in response to the DR-5 requirements of contract NAS8-37856, Space Transfer Vehicle Concept and Requirements. It is the DR-5 identified in Data Procurement Document No. 709.

GLOSSARY

ACC	Aft Cargo Carrier
ACS	Attitude Control System
AFE	Aeroassist Flight Experiment
Al-Li	Aluminum Lithium
ALS	Advanced Launch System
APCM	Advanced Programs Cost Model
ASE	Airborne Support Equipment
ATLO	Acceptance, Test, Launch, and Operations
ATR	Advanced Technology Roadmap
BOE	Basis of Estimate
C&DM	Configuration and Data Management
CAD	Computer-Aided Design
CDR	Critical Design Review
CER	Cost Estimating Relationship
CG	Center of Gravity
CLAAS	Closed-Loop AeroAssist Simulation
CNDB	Civil Needs Data Base
COLD-SAT	Cryogenic Onorbit Liquid Depot Storage, Acquisition, and Transfer Satellite
CSLI	Civil Space Leadership Initiatives
DDT&E	Design, Development, Test, and Evaluation
DOD	Department of Defense
DR	Data Requirement
DRM	Design Reference Missions
ETO	Earth-to-Orbit
ETR	Eastern Test Range
GEO	Geosynchronous Earth Orbit
GN&C	Guidance, Navigation, and Control
GPS	Global Positioning Satellite
GSE	Ground Support Equipment
H/W	Hardware
I/F	Interface(s)
ILC	Initial Launch Capability
IMU	Inertial Measurement Unit
IR	Interim Review
IR&D	Independent Research and Development
IRD	Interface Requirements Document
KSC	Kennedy Space Center
L/D	Lift-to-Drag Ratio
LAD	Liquid Acquisition Devices
LCC	Life-Cycle Cost
LEO	Low-Earth Orbit

LeRC	Lewis Research Center (NASA)
LEV	Lunar Excursion Vehicle
LTV	Lunar Transfer Vehicle
LV	Launch Vehicle
MAP	Manifesting Assessment Program
MDC	McDonnell Douglas Corporation
MLI	Multilayer Insulation
MMS	Multimission Modular Spacecraft
MSFC	Marshall Space Flight Center
MSS	Manned Space Systems
NASA	National Aeronautics and Space Administration
NASP	National Aero-Space Plane
OTV	Orbital Transfer Vehicle
P/A	Propulsion/Avionics
PDF	Probability Distribution Function
PRD	Preliminary Requirements Document
RAMP	Risk Assessment and Management Program
RCS	Reaction Control System
RFP	Request for Proposal
S/W	Software
SE	Support Equipment
SEI	Space Exploration Initiative
Sh-C	Shuttle-C
SOFI	Spray-On Foam Insulation
SSF	Space Station Freedom
STAS	Space Transportation Architecture Study
STCAEM	Space Transportation Concepts and Analysis for Exploration Missions
STIS	Space Transportation Infrastructure Study
STS	Space Transportation System
STV	Space Transfer Vehicle
STVIS	Space Transfer Vehicle Information System
TCS	Thermal Control System
TEI	Trans-Earth Injection
TMI	Trans-Mars Injection
TPS	Thermal Protection System
TT&C	Telemetry, Tracking, and Control
TVC	Thrust Vector Control
TVS	Thermodynamic Vent System
UNIS	Unified Information System
USRS	Upper Stage Responsiveness Study
VCS	Vapor Cooled Shields
WTR	Western Test Range

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1.0 INTRODUCTION

With the initiative provided by the president to expand the exploration and habitation of space, a need arose to define a reliable and low cost system for transporting man and cargo from the earth surface or orbit to the surface of the moon or Mars. The definition of this system is two fold, the need for an low cost heavy lift Earth-To-Orbit system represents one of the major emphasis the other is the transportation system itself. The STV study has analyzed and defined an efficient and reliable system that meets the current requirements and constraints of both the existing and planned ETO systems as well as the surface habitation needs, as well arriving at the definition of key technologies needed to accomplish the these further needs. The results of the study provide a family of systems that support a wide range of existing and potential space missions. The simplest of the systems support the near earth orbital payload deliveries for both NASA and the DoD, requiring very short mission duration with no recovery of any portion of the system. The more complexity systems prove support for the interplanetary manned missions to both the moon and to Mars. These system represent state of the art systems that provide safety as well as reusable characteristics that allow the system to be used spaced based, the next step in the expansion of mans' presence in space.

The time to develop this STV family is now. Its role in complementing the space transportation infrastructure, keeps the United States of America as the world leaders in science, defense, and commercial space ventures for the 21 st century.

The space transportation tasks that the STV system must perform to transport humans with mission and science equipment from Earth to high earth orbits or the surfaces of the moon or Mars can be divided into three phases. (1) Transportation to-and-from low Earth orbit (LEO) being accomplished by the NSTS, ELVs, and new heavy-lift launch vehicles (HLLV) capable of 75 to 150 t cargo delivery; (2) space transfer vehicles providing round-trip transportation between LEO, lunar, and planetary orbits; and (3) excursion vehicles providing transportation between lunar/planetary orbits and their surfaces. Where one mode of transport gives way to another, transportation nodes can be utilized. In low Earth orbit, Space Station Freedom or a co-orbiting platform can serve that need. Elements of the space transfer and excursion vehicles are delivered by the HLLV and crews by the NSTS. Once all the elements have been delivered crews from SSF assemble, checkout, and then launch the vehicle. Following completion of the planned stay at the orbital node, lunar surface, or Mars, the transfer vehicles return the crew and a limited amount of cargo to LEO where the vehicles are refurbished and serviced for additional missions. Performing the transportation functions in this manner maximizes the commonality and synergism between the lunar and Mars space transportation systems and brings the challenge of the exploration initiatives within the reach of orderly technology advancement and development.

Our final report addresses the future space transportation need and requirements based on our current assets and their evolution through technology/advanced development using a path and schedule that supports our world leadership role in a responsible and realistic financial forecast. Always, and foremost, our recommendations place high values on the safety and success of missions both manned and unmanned through a total quality management philosophy at Martin Marietta.

2.0 STV CONCEPTS AND REQUIREMENTS STUDY

Per the 20 July 1989 presidential directive, NASA prepared a plan for sustaining planetary exploration spanning 1990 to 2030. Elements of the plan include Mission to Planet Earth, return to the moon to stay by creating a manned lunar outpost, followed then by manned missions to Mars. The charter of the STV Concepts and Requirements definition program was to initiate a new era of space-basing, capitalizing on the economic benefits achieved by reuse of major hardware elements. The principal LEO element that supports this reusability goal, is Space Station Freedom and its precision proximity operations support equipment. Provided through this node is the space support that includes; launch, refurbishment, and control for both development and operational missions, for a reusable, space-based STV system is possible.

The STV program began with the NASA/contractor defining preliminary program options to support the lunar and Mars initiative. The results of this effort, was a family of transportation vehicles that were capable of supporting Near Earth and lunar missions, with a growth potential for supporting the Mars missions, and an integrated program plan that defines a six year Space Transfer Vehicle and ETO Phase C/D development program, with unmanned development validation flights starting in 2002. The family of vehicles represent unmanned expandable cargo vehicles that transport the critical lunar habitation elements to the moon beginning in 2004. These expendable vehicles evolve into a reusable system prior to placing a crew in the system. This evolution provides a test bed for the critical flight elements within the system to be tested and validated without the costly expense of a unique test article. In 2005, a four man crew is transported from LEO to the lunar surface with a cargo of 14.6 tonnes, and returned after a 30 day stay on the surface. The following piloted missions increase in surface stay duration until a maximum stay time of six months is achieved. This lunar program is made up of four major phases of the operation- Precursor, Emplacement, Consolidation, and Utilization as defined in the Space Exploration Initiative (SEI) Requirements Document. Technology/advanced development activities over the next decade will allow these accomplishments with lower operating costs and increased confidence over today's level of engineering design though the initiation and demonstration of engineering solutions in low cost, laboratory environments prior to committing to full scale hardware developments.

2.1 Study Objectives

The objectives of the STV Concepts and Requirements studies were to provide sensitivity data on usage, economics, and technology associated with new space transportation systems. The study was structured to utilize data on the emerging launch vehicles, the latest mission scenarios, and SEI payload manifesting and schedules, to define a flexible, high performance, cost effective, evolutionary space transportation system for NASA. Initial activities were to support the MSFC effort in the preparation of inputs to the 90 Day Report to the National Space Council (NSC). With the results of this study establishing a point-of-departure for continuing the STV studies in 1990 additional options and mission architectures were defined. The continuing studies will update and expand the parametrics, assess new cargo and manned ETO vehicles, determined impacts of the redefined Phase 0 Space Station Freedom, and to develop a design that encompasses adequate configuration flexibility to ensure compliance with on-going NASA study recommendations with major system disconnects.

In terms of general requirements, the objectives of the STV system and its mission profiles will address crew safety and mission success through a failure-tolerant and forgiving design approach. These objectives were addressed through: engine-out capability for all mission phases; built-in-test for vehicle health monitoring to allow testing of all critical functions such as, verification of lunar landing and ascent engines before initiating the landing sequence; critical subsystems will have multiple strings for redundancy plus adequate supplies of onboard spares for removal and replacement of failed items; crew radiation protection; and trajectories that optimize lunar and Mars performance and flyby abort capabilities.

The results of the study were developed through a series of major analysis activities that included requirements analysis, configuration analysis and definition, operational analysis and interface definition, programmatic assessment of both the configuration and operations, and an integrated technology/advanced development plan. Details of the activity that made up this effort will be discussed in detailed throughout the remainder of this document. At this point, however, it is necessary to define in some depth the 90-Day study results that represents the foundation for the recommended LTS/STV systems.

Data derived from the MASE baseline regarding the Space Exploration Initiative (SEI) during the period from July through December, 1989 and many of the initial study results was used to develop the "90 Day Report", that MSFC submitted to the NSC as a recommendation for an approach for conducting the lunar and Mars exploration programs. From this study the reference 2-1/2 stage vehicle configuration, Figure 2.1-1, was adequate for the required missions but was optimized for cost and performance. This system utilized SSF as the LEO transportation node, required an 15 foot diameter x 71 t ETO capability, with an five mission reusability goal supported by a rigid aerobrake for Earth reentry. The operational scenario recommended for this system,

Figure 2.1-2, transported both the transfer and excursion vehicles to Low Lunar Orbit (LLO), where the transfer vehicle was left in orbit while the excursion vehicle descended to the lunar surface. Upon completion of the lunar stay, the excursion vehicle ascended to LLO where it docked with the transfer vehicle and the crew is transferred from the excursion to the transfer vehicle. The two vehicles separate and the excursion vehicle is left in LLO and the transfer vehicle returns to Earth using the aerobrake for reentry followed by a series of orbital maneuvers to rendezvous and dock the vehicle with SSF.

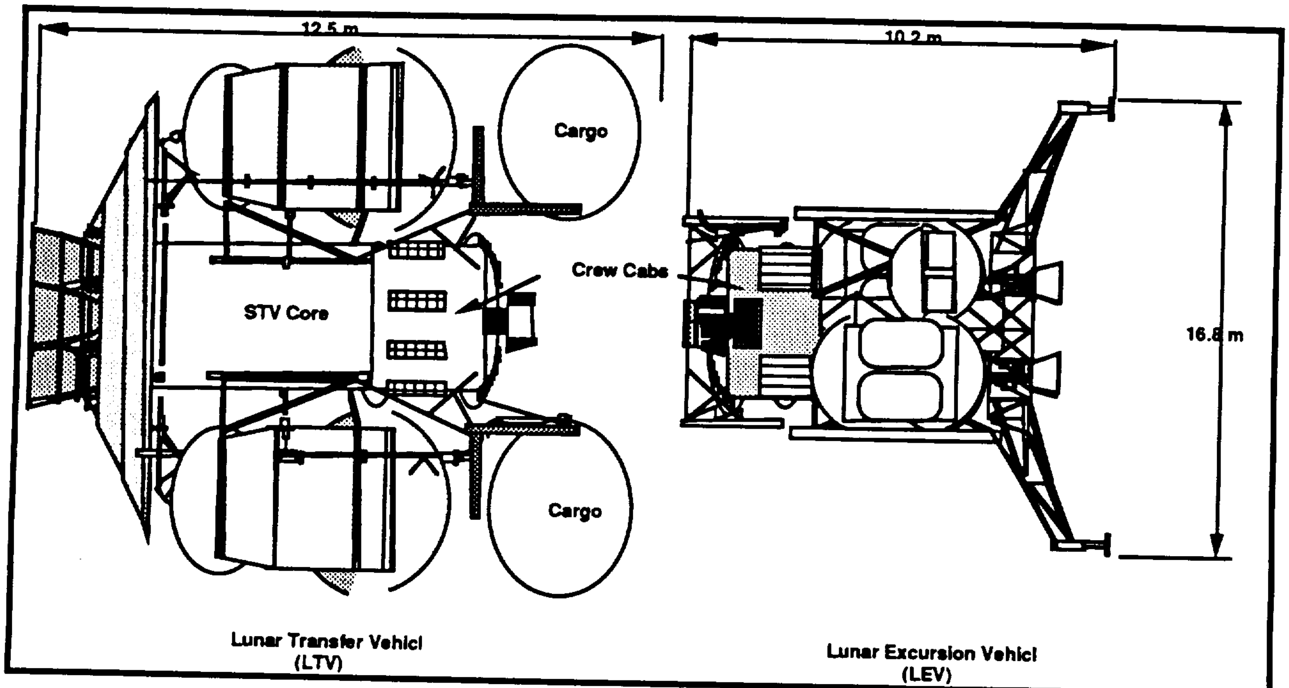


Figure 2.1-1: 90-Day Reference Configuration

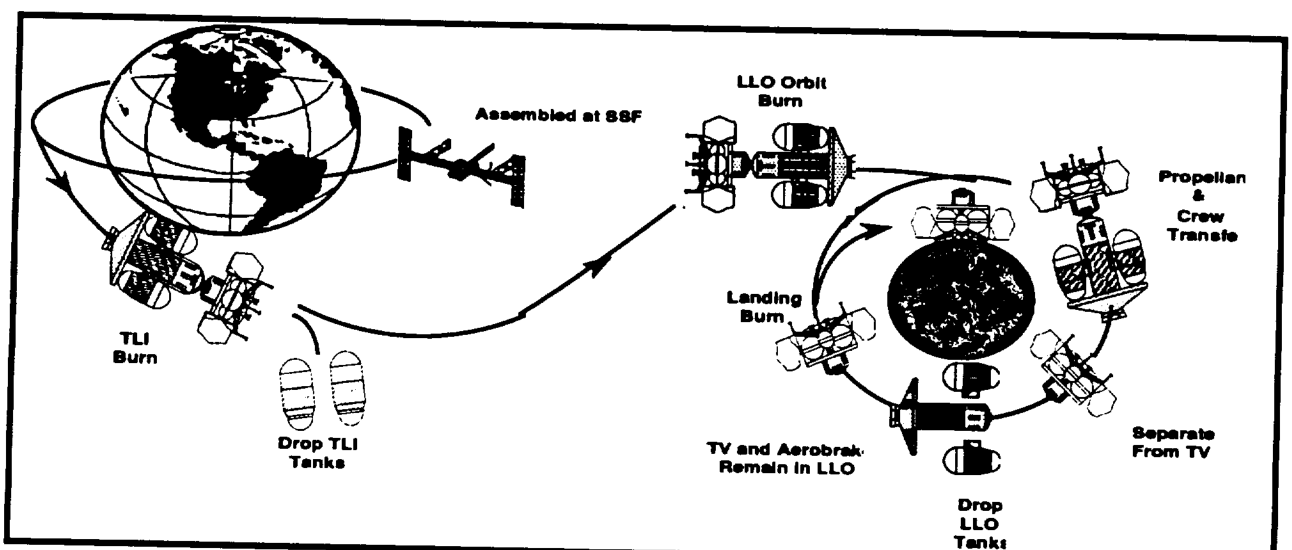


Figure 2.1-2: 90-Day Reference Configuration Operations Scenario

2.2 Systems Engineering And Requirements

The objective of the systems engineering task was to develop and implement an approach that would generate the required study products as defined by program directives. This product list included a set of system and subsystem requirements, a complete set of optimized trade studies and analyses resulting in a recommended system configuration, and the definition of an integrated system/technology and advanced development growth path. A primary ingredient in Martin Marietta's approach was the TQM philosophy stressing job quality from the inception.

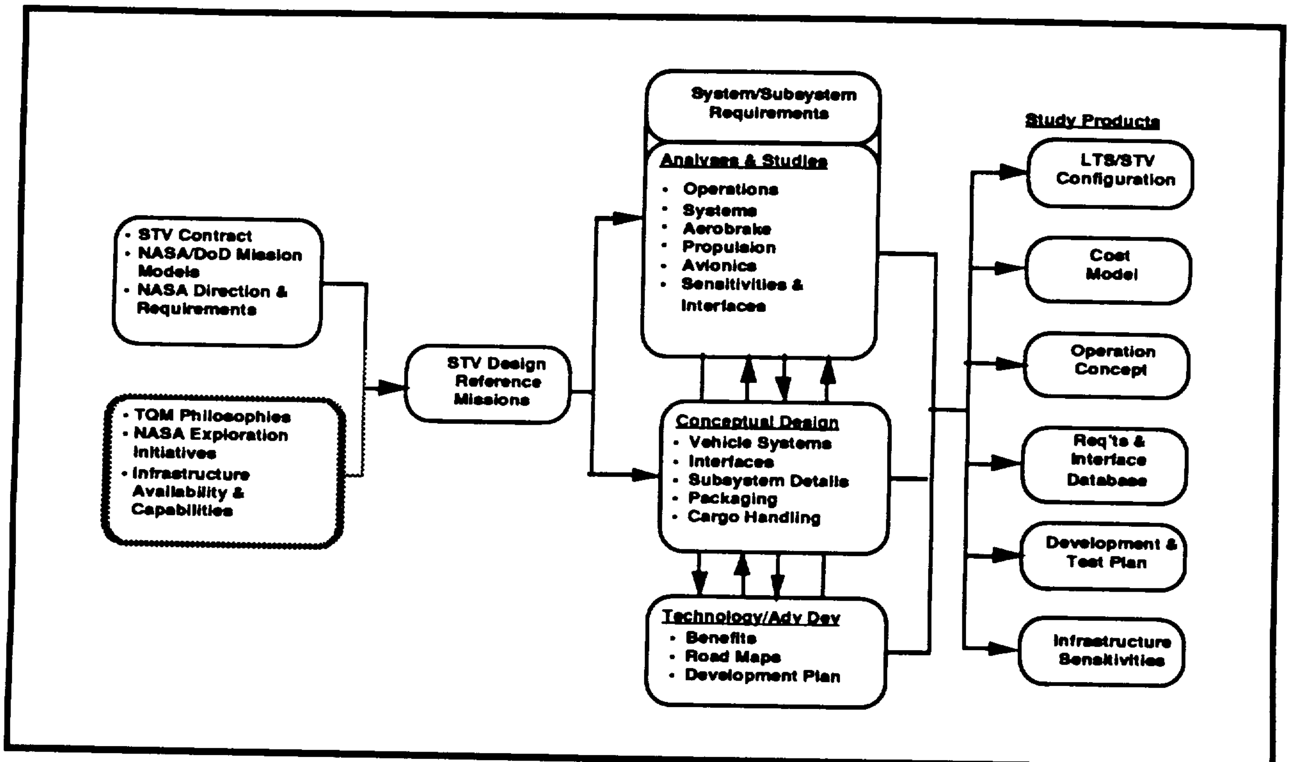


Figure 2.2-1 Systems Engineering Approach

The systems engineering approach, see Figure 2.2-1, used a reference baseline from past study documentation to establish the foundation for further study. Derived from this reference database were the Design Reference Missions (DRMs), system and subsystem requirements, conceptual design, and the studies and analyses, technology/advanced development effort, all resulting in the recommended LTS/STV configuration shown in Figure 2.2-2, a cost model, an operations concept for conducting manned lunar missions, system and subsystem requirements and interfaces database, a development and test plan, and defined infrastructure sensitivities. This recommended LTS configuration supports several different operations scenarios that including, Piloted, Reusable Cargo, and Expendable Cargo, with minor element changes. The basic flexibility of the LTS configuration also provided several alternative configurations, shown in Figure 2.2-3.

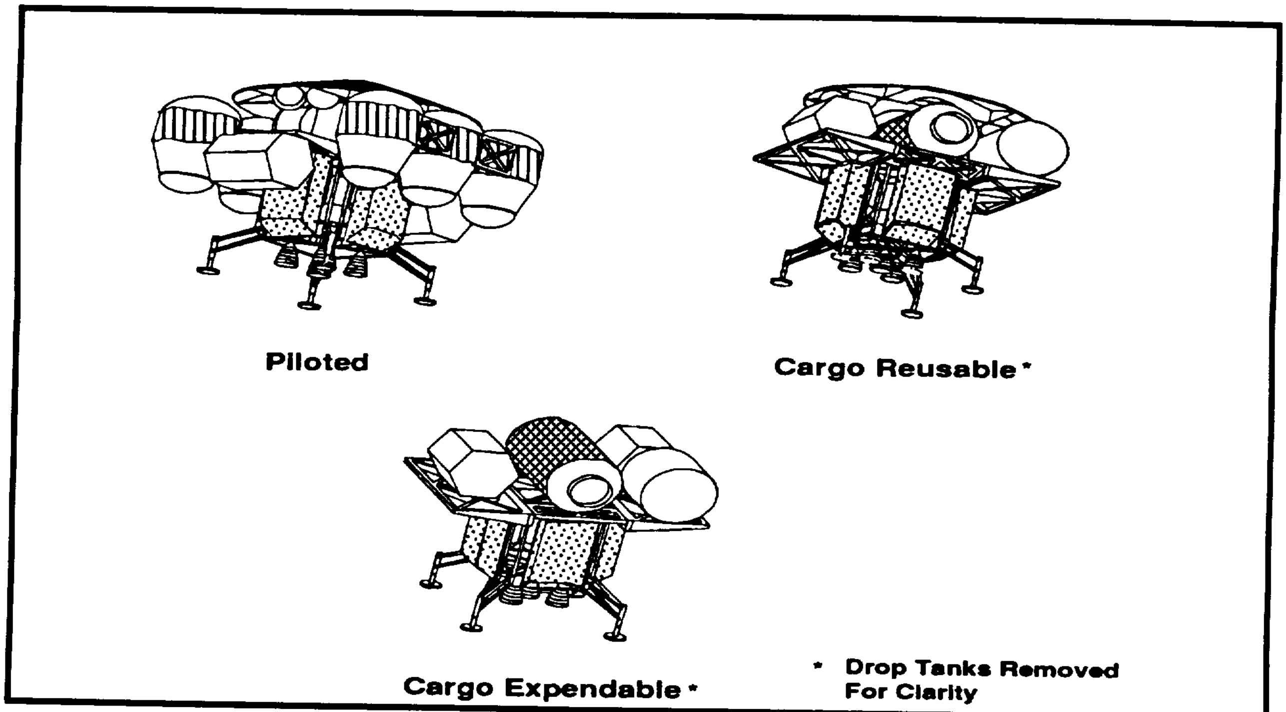


Figure 2.2-2 Piloted, Reusable Cargo, and Expendable Cargo LTS Configuration

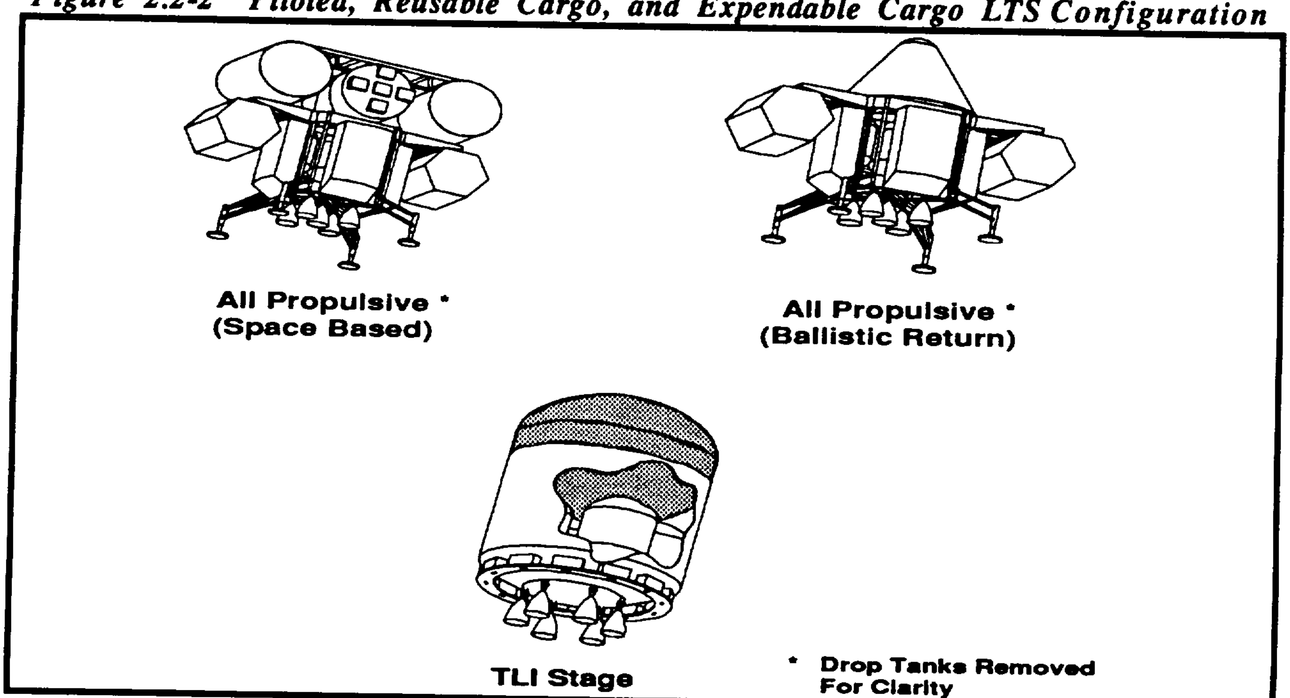


Figure 2.2-3 Alternative Configurations

These configurations represented an All-Propulsive Space-Based Configuration, an All-Propulsive Non-Space-Based Configuration, and a High Energy Upper Stage for use with an HLLV or the

LTS. The High Energy Upper Stage has generated considerable interest as a means of increasing the mission capture potential of the new National Launch System (NLS) vehicles that are under consideration.

Additional analyses and studies of the systems comprising the LTS configuration (aerobrake, propulsion, avionics and structure) show key links to similar system functions in other planned infrastructure components such as the proximity operations vehicle, and deep space exploration systems.

Seven Design Reference Missions (DRMs), represent three destinations, Near Earth, Lunar, and Mars. These DRMs provide the bounding requirements, Figure 2.2-4, for defining the final recommended STV/LTS family of vehicles.

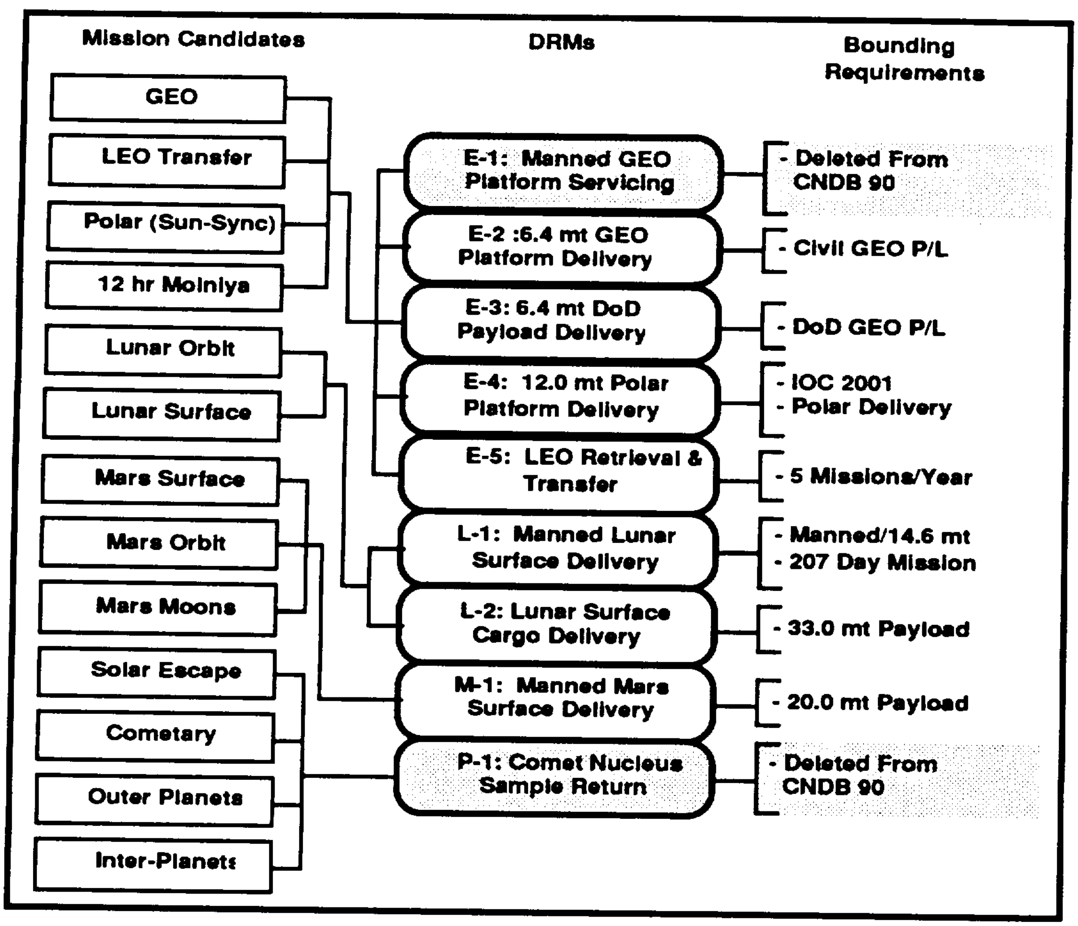


Figure 2.2-4 STV DRMs and Their Corresponding Requirements

Using the process illustrated in Figure 2.2-5, these missions were selected from several reference sources: the 1989/90 CNDB, supplemented with the STV augmented CNDB (09 Aug 1989); the

1989 Air Force Space Command National Mission Model; and the Human Exploration Study Requirements Document.

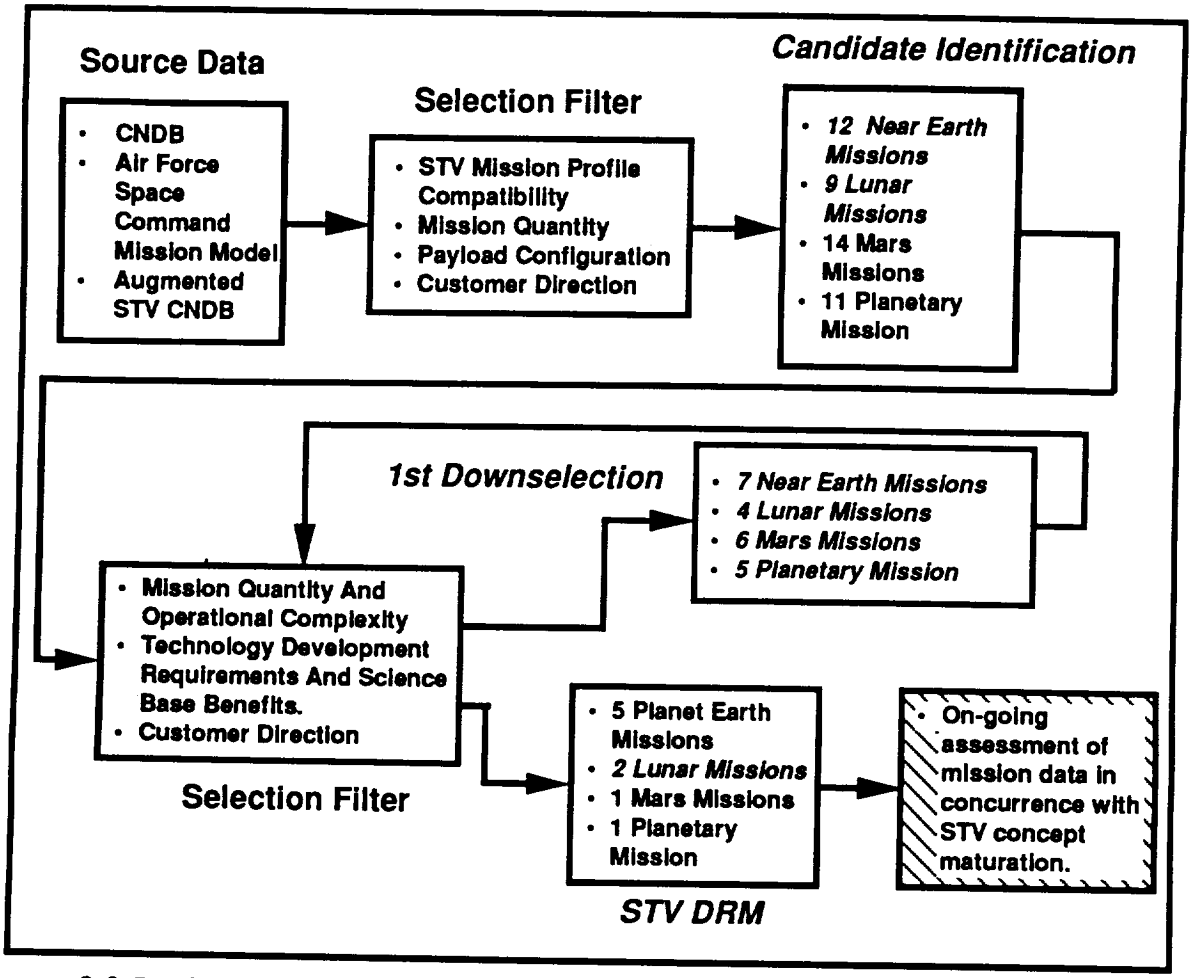


Figure 2.2.5 STV DRM Selection Process

As shown, these bounding requirements include key areas of interest such as Man-rated/Reusable, Payload Type, Payload mass, first flight, number of missions, duration of each mission, and the total mission Δ-velocity. Listed below are the key requirements that imposed the most influence on the LTS/STV development activity. It should be noted that the characteristics associated with the LEO Payload Retrieval/Transfer mission were not considered drivers in the definition of the LTS/STV configurations, but were accommodated by the operational system.

1) **First Flight shall occur in 2001:** Across all missions, the IOC date of 2001, represents an impact to and integration of technology, scheduling of the DT&E test program, and support node (i.e. SSF) availability.

2) **Provide a total Δ-velocity up to 9.5 km/s:** With a Δ-velocity ranging from 9.5 to 2.9 km/s a direct correlation exists to vehicle sizing, ETO interfaces and performance, support node accommodation, and the propulsion system.
3) **System shall be capable of injecting a payload mass of up to 33 tonnes:** Combined with the performance requirements of 9.5 to 2.9 km/s the mass delivered defines vehicle sizing and structural configuration, support equipment, and directly influences the system operational cost.
4) **Mission Duration of up to 50 days of full up operations and the capability of maintaining system operations for 207 days, shall be accommodated:** Operational time impacts are constrained primarily to the manned missions. It should be noted, however, that of the 207 days required for the Manned Lunar mission, only 30 days of full up operations is needed.

Of the seven STV DRM's, the Lunar missions (both manned and unmanned) represent the primary contributor to the STV growth requirements. To ensure the proper implementation of these requirements, the emphasis during the system concept definition and development phases focused on the lunar missions, with evolutionary considerations given to the GEO, Planetary, and Mars missions.

Using the bounding requirements established through the above STV DRMs, a set of system level requirements was developed, Figure 2.2-6, and carried forward into the definition of the transportation vehicles. These requirements include basing, man-rating, maintenance and service life, earth return, propellant, autonomy, and operations and interfaces. They were derived from NASA documentation, system and configuration trades and analyses, or the STV contract SOW.

This requirements base is defined in two categories: 1) General requirements that are imposed on systems supporting all transportation scenarios, and 2) mission unique requirements that impact specific missions such as lunar and Mars.

The general STV requirements define manned operations, interfaces, mission environment, design, and verification. Key requirements that will be imposed on all configurations and operations of the STV system have been shown below.

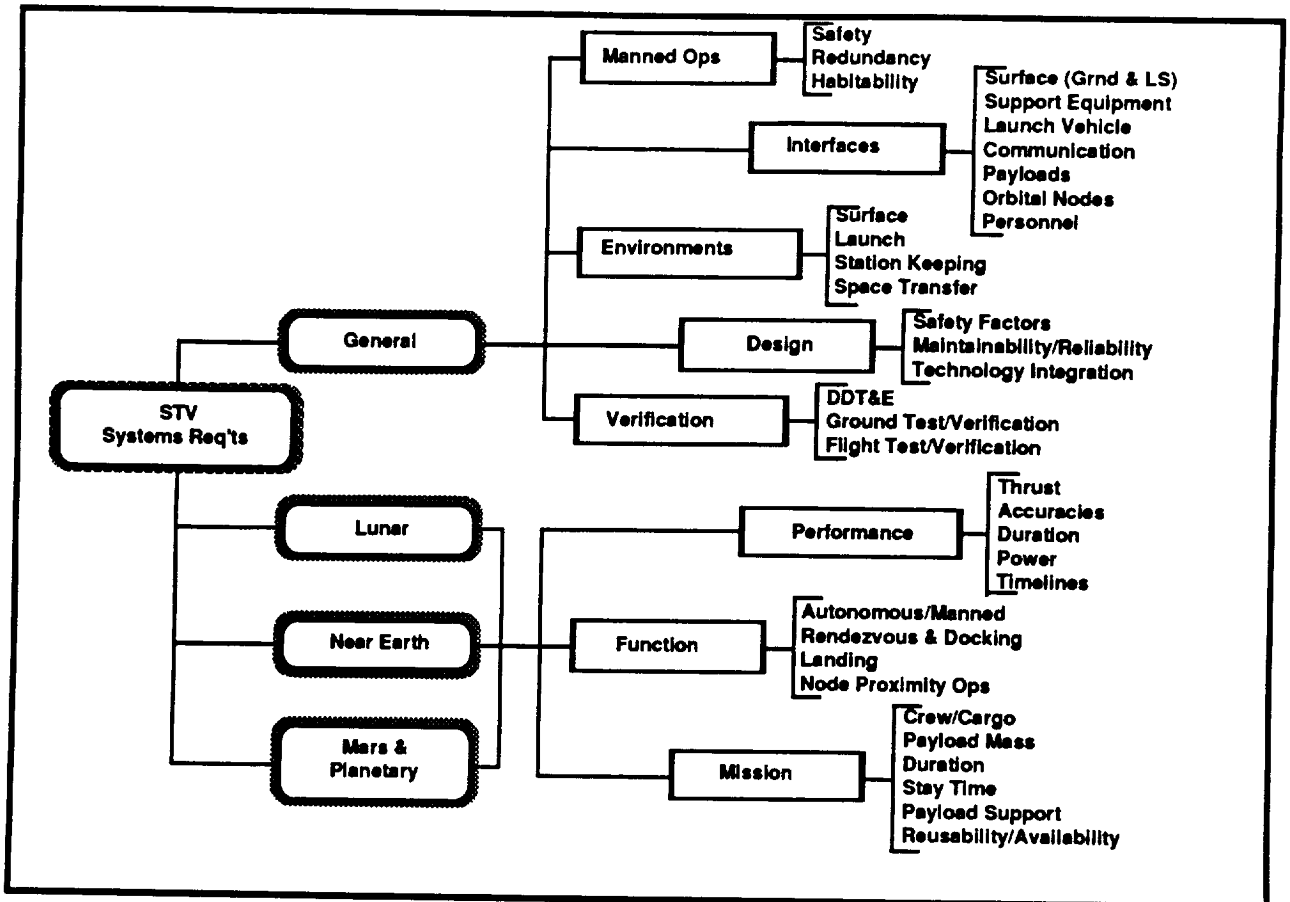


Figure 2.2-6 STV System Requirements

Manned Mission Operational Requirements—The STV shall be capable of transporting personnel (one or more) to a safe haven, with abort trajectories for free return aborts for manned missions and planetary surface impacts for the disposal of unmanned mission hardware.

A minimum of two crew members shall perform each scheduled EVA. Suit pressure/pre-breather combinations for EVA shall achieve an R value of 1.22. In-space and surface EVA provisions shall be made for each crew member. It is not required to provide simultaneous capability for the entire crew.

Interfaces—The Space Transportation system shall interface with earth based facilities, ground transportation systems, power systems, payload handling mechanisms, thermal management systems, and launch elements. The ground operations will provide processing, assembly and checkout, and launch of space transportation elements. The STV crew will be processed as part of an STS (or equivalent) mission launched with existing ground elements.

The Earth-to-Orbit system shall provide the hardware systems and/or support system which provide the capability for transportation elements to be delivered to LEO.

a) STV Elements shall be delivered to a 160 nmi circular by 28-1/2 to 56° inclination orbit
b) Payloads diameters up to and including 10 m will be delivered to LEO
c) A maximum of six ETO flights/year will be allocated to support space transportation missions

The Low Earth Orbit (LEO) transportation node shall provide the hardware systems and/or support systems for assembly, storage, checkout, refurbishment, and control of transportation elements. Propellant management and storage shall be capable of providing a maximum storage time for a quantity not to exceed 174 mt, for 90 days.

Transportation system shall interface with all destination support elements. Manned systems shall interface with power systems, data systems, payload handling mechanisms, thermal and propellant management systems, life support systems, and launch elements. Unmanned systems shall interface with power systems, data systems, and payload handling mechanisms.

Design—Fault detection/fault isolation and reconfigurations of critical systems will be provided (ref. 3: NHB 53000..4 (1d-2) "Safety, reliability, Maintainability and Quality Provisions For The Space Shuttle Program"). Redundancy for man-rated elements shall be Dual-Fault Tolerant (Fail-Op, Fail-Op, Fail-Safe). Critical mission support functions shall be one failure tolerant. Critical functions affecting crew safety and survival shall be two failure tolerant.

The service life of STV systems and subsystems shall be a minimum of five missions. There will be no scheduled in-flight maintenance. All scheduled maintenance shall take place at the Earth transportation and space based nodes. Removal and replacement shall be done at the functional component level. Non-pressurized systems shall be accessible to telerobotic or EVA maintenance.

Technology—First flight shall not be impacted by technology development schedules. System architecture will allow incorporation of new technologies as they become available.

Verification—Overall reliability shall be demonstrated and verified by testing (ref. NHB 53000..4 (1d-2) "Safety, reliability, Maintainability and Quality Provisions For The Space Shuttle Program"). Requirement verification shall be performed, either by analysis or test. System shall be certified for flight only after the requirement verification has been satisfactorily completed. All critical mission elements shall be verified by flight test. All critical mission elements shall be verified by ground test, to the extent practical.

The mission unique requirements shown below, represent those characteristics that support the conduct of a specific mission and should not be imposed on another class of missions.

Lunar Mission Requirements— Transportation system shall deliver to the Lunar surface, 429 tonnes PSS elements between 2002 and 2026. 142.8 tonnes between 2002 to 2007, 106.0 tonnes between 2008 to 2013, and 189.9 tonnes between 2014 to 2030.

Piloted Flights shall deliver a crew of four and a maximum of 14.6 tonnes of cargo to the lunar Surface and return a crew of four and a maximum of 0.5 tonnes of cargo to earth orbit. Cargo flights shall deliver a maximum of 33.0 tonnes of PSS components. The flight rate for the delivery of these payloads shall not exceed one mission per year.

Transportation system shall be capable of autonomous rendezvous and payload propellant transfer. Landing on the lunar surface occurs on a 50 meter diameter pad, level within 2 deg (improved), or on unimproved landing pads level within 15 deg. Landing shall also be accomplished over surface irregularities not in excess of 1 meter in height (unimproved).

Mission operations that shall not exceed a planned duration of 4360 hours (180 days), from earth launch to earth return. All system elements shall remain in lunar proximity during manned occupation. Period includes 48 hours following landing and prior to ascent.

Utilizing the following requirements, the transportation system shall provide performance capabilities of delivering crew and cargo.

a) Propulsion system utilizes cryogenic propellant
b) Two engines out will not abort the mission
c) Total cryogenic boil-off shall not exceed 2% per month
d) 1% reserves for Isp
e) 1.5% residual
f) 5% ullage

Unmanned mission does not require meteoroid/debris protection. In-space propellant transfer shall be performed between the vehicle and LEO node, internal vehicle tankage, and the vehicle and the PSS support equipment on the lunar surface.

First manned flight shall support manned occupation on the lunar surface by 2004. First cargo flight will be to the lunar surface by 2002.

Near Earth Mission— The transportation system shall be capable of delivering payloads to LEO between 2001 and 2030, GEO between 2001 and 2019, and to a polar orbit between 2001 and 2008. Missions shall deliver a maximum of 12.0 tonnes with a flight rate not exceeding two missions per year. System will be capable of autonomous rendezvous, docking, and payload/propellant transfer. Reusable configurations will use an aerobrake return to LEO. Meteoroid/Debris protection shall not be provided for unmanned Near Earth configurations. In-space propellant transfer is performed between the vehicle and LEO node and internal vehicle tankage.

Mars Mission— System shall be capable of supporting the delivery of 20 tonnes of cargo and a crew of four to the Mars surface between 2015 and 2026.

As the description of the LTS/STV configuration matured, eight system requirements were found to be key design drivers. The impacts that these requirements bring to the design of the system are defined below. It should be noted that a change in any one of these requirements has the potential of completely altering the results of the configuration selection activity.

System Shall Deliver 14.6 tonnes of cargo and a crew of four to the surface and return: Delivery of 14.6 tonnes of cargo and a crew of four represents the maximum propellant requirements of the three mission scenarios (piloted, reusable cargo, and expendable cargo). Transforming the piloted system to an expendable cargo configuration provides the capability to delivery 37.4 tonnes of cargo with the same propellant tanks as carried on the piloted mission. Sizing the propellant tanks and vehicle for the 33.0 tonne cargo mission will result in a cargo capability well short of the 14.6 tonne requirement in the piloted mode.

System shall be reusable for a minimum of five missions: Reuse of the system requires more of the vehicle elements to be returned to a LEO node to make the scenario economically feasible. To support this, the IMLEO required for the mission increases to support the return performance requirements. A LEO Node becomes the primary support element for assembly, checkout, and verification. To minimize the assembly requirements at the LEO Node, quick disconnects are required in major system elements, impacting IMLEO as well as driving technology requirements. Within the vehicle itself, system health monitoring and aeroassist become mandatory, to minimize performance requirements and LEO Node maintenance. While reducing LEO Node EVA/IVA requirements, the additional avionics equipment increases the IMLEO.

Manned systems shall be fault tolerant: Increasing the avionics complexity to comply with this dual-fault tolerant requirement adds additional mass second only to the propellant as the major contributor to the IMLEO. Included in this complexity is the additional software that will be required, becoming an enabling technology and thus having a direct impact on system availability.

System shall deliver 429 tonnes to the Lunar surface between 2004 and 2030 as defined by the PSS requirements document (05 Jun 90): Compliance with the manifest delivery schedule defined by PSS, requires the use of a minimum of four expendable cargo missions as shown in Figure 2.2.-7. Minor reallocation of the cargo can significantly reduce the LCC costs of the LTS/STV program by allowing the reuse of three of these four cargo vehicles. The large cargo requirements in these expendable missions translates into major impacts to support systems such as KSC, the LEO Node, and the handling of the cargo once delivered to the surface.

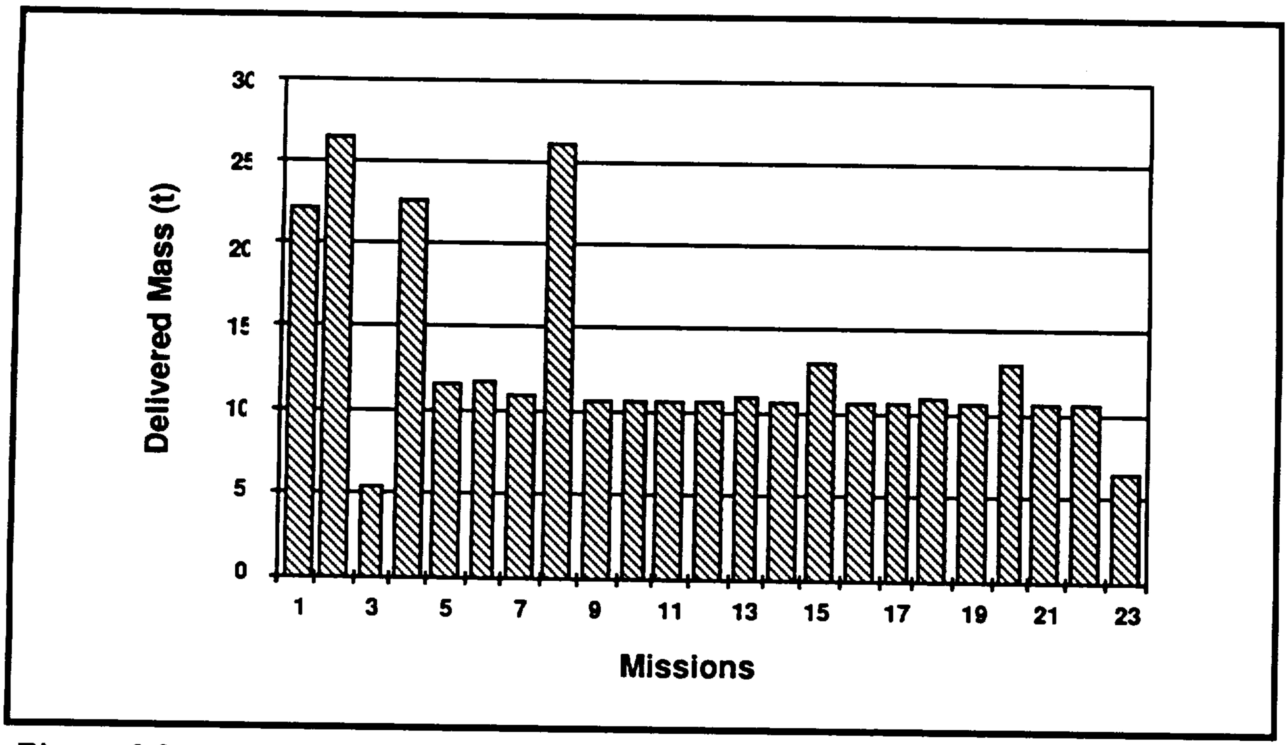


Figure 2.2-7 PSS Manifest Lunar Surface Delivery Requirements

Space Station Freedom shall be utilized as the LEO Transportation Node: With SSF used as the LEO Node, all interfaces with the supporting space infrastructure (KSC, ETO, PSS, and others) and the LTS/STV must be common with those on SSF. This increases the LTS IMLEO since the SSF interfaces have been designed for stationary operations where weight restraints do not pay as much of a penalty as they do on a transportation vehicle. The handling and storage of propellant tanks have a physical and safety impacts. Present data shows that the crew requirements for assembly and servicing of the LTS/STV fleet ranges from 400 to 1200 manhours or at a maximum 70% of the available crew time at SSF, see Figure 2.2-8. Contamination issues must be addressed to ensure that the SSF environment is not adversely affected. If the management and control of contamination falls on the LTS side of the interface, the potential exists for significantly increasing the IMLEO of the system.

System IOC shall be 2001 with initial manned flight in 2006: To support a mission in 2001, necessary technology must be at Level 6 or at PDR maturity by 1996. Based on current technology plans, the potential for the highly advanced systems necessary to meet the requirements of the STV/LTS program is moderate at best.

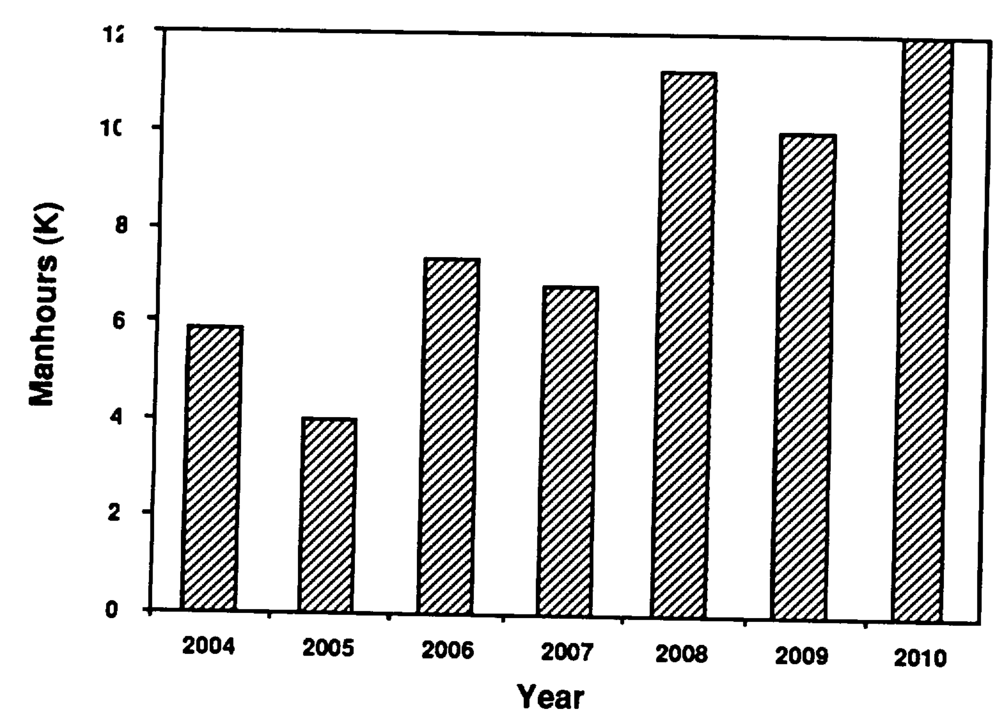


Figure 2.2-8 STV Assembly and Support Manpower Requirement

Propulsion system shall utilize LO_2/LH_2 propellant: Cryogenic propellants require complex and expensive storage equipment both at LEO and the lunar surface. Development and transportation of this equipment directly impacts the STV/LTS economically and physically. Replacement of the cryogenic propulsion system with an advanced propulsion system, such as a nuclear thermal rocket (NTR), can increase the mass capability to the Lunar surface by as much as 100%, as shown in Figure 2.2-9, which translates into a lower IMLEO if the current PSS mass requirements are maintained.

System shall be capable of autonomous operation: Increasing the avionics complexity to provide autonomy adds additional mass second only to the propellant as the major contributor to the IMLEO. Included in this complexity is the required additional software. With this requirement, software becomes an enabling technology having a direct impact on system availability. Training requirements and facilities for the flight crews are reduced by implementing autonomous operations.

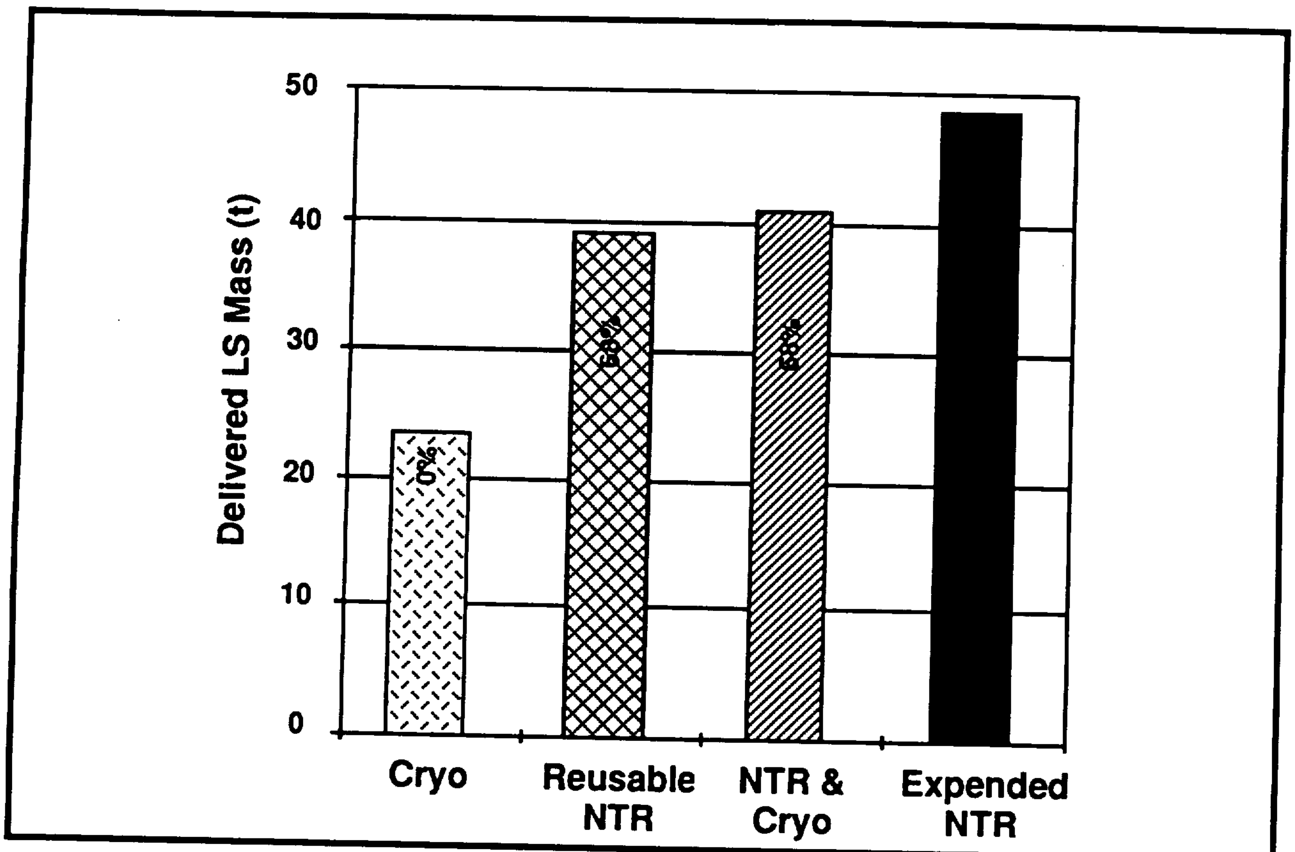


Figure 2.2-9 Propulsion System Capabilities to the Lunar Surface

2.3 SYSTEM TRADE STUDIES & ANALYSES

Top-level systems trades provided results that directly influenced the definition and the selection of the optimum STV concept or family of vehicles. Top-level program decisions were made regarding aeroassist versus all propulsive, vehicle growth options, performance impact of lunar liquid oxygen, direct descent versus lunar orbit, etc. The results of substantiating system trades are included in this section following the description of the STV concept selection process.

The analysis and study activities of the STV Study program were made up of six major areas; systems, mission operations, avionics, aerobrake, propulsion, and interfaces, as defined in Figure 2.3-1 These categories were defined within the original proposal and updated in the initial phases of the program with inputs from our MFSC customer as well as on-going studies. Included in this process was the ability to integrate the top level system results in the definition of the key subsystem.

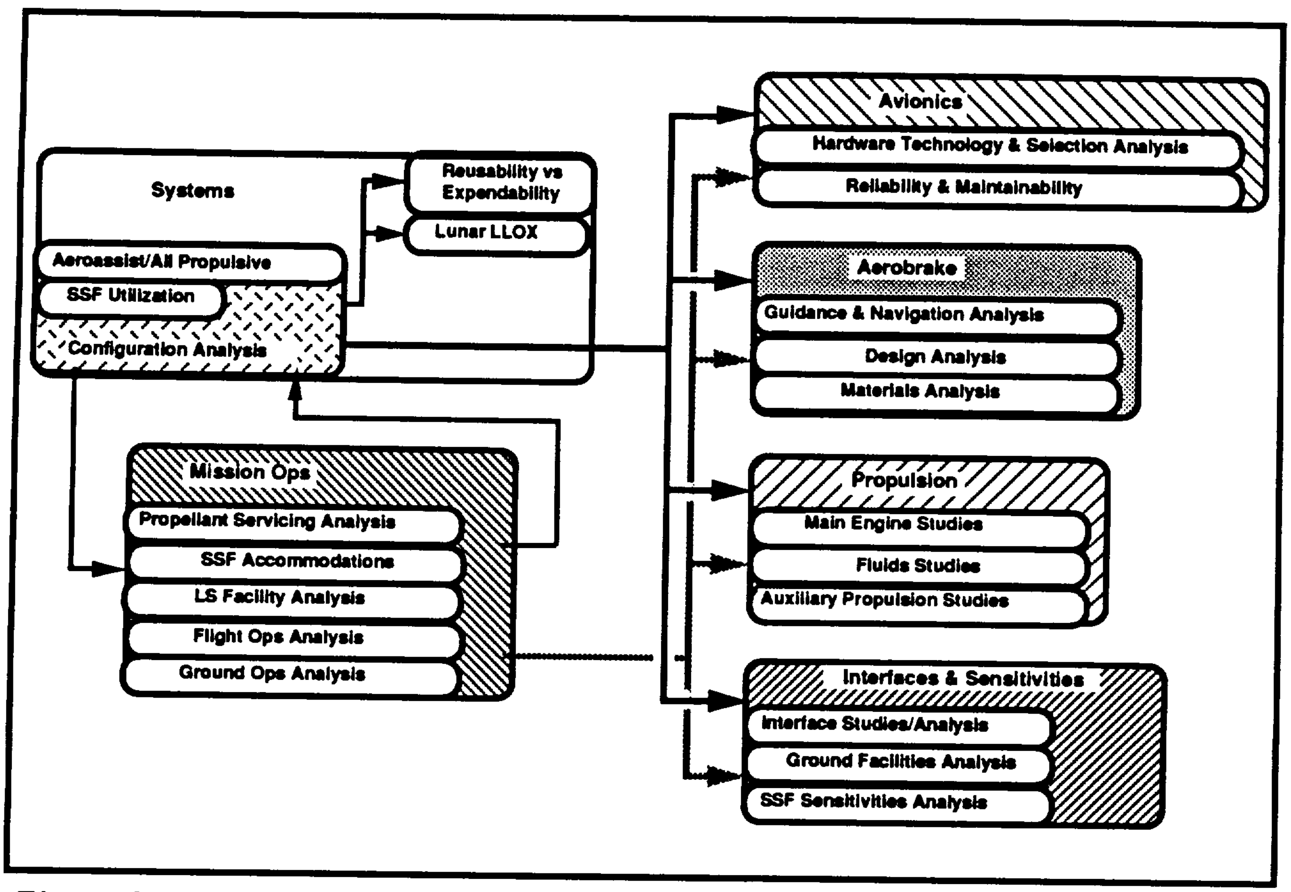


Figure 2.3-1 STV Studies & Analyses Approach

2.3.1 Mission Operations

The mission operations study provided performance, sensitivities, and operations needed before the configuration analysis could be completed. Included in this study were analyses that addressed; ground, orbital, flight , and surface studies, with the emphasis placed on supporting the "Option 5" lunar outpost missions. Results were largely influenced by Martin Marietta's involvement in the MSFC "Skunk Works" effort. Since the primary focus of the "Skunk Works" was the lunar missions the bulk of the data available supported the continuation of the detailed definition and description of a Lunar Transportation System (LTS) with an upward and downward evolution to Mars and Near Earth missions.

Orbital Operations Analysis—Orbital operations analysis assumed the ability of Space Station Freedom to provide support to a spaced-based transportation system. Key areas addressed were the approach to element assembly, with an emphasis on the aerobrake, and the ability of the station crew to provide the necessary support. One of the main Space Station based operations for STV servicing is the assembly of the aerobrake. Along with being intricate, the operational approach has a large impact on the design of the aerobrake. Three criteria areas, crew resources, task time and technology risk were analyzed for two separate aerobrake assembly operations approach.

Option 1 (IVA/Telerobotic Assembly) involves using the crew, inside a Space Station pressurized control center, to direct telerobotic operations to assemble, connect and verify aerobrake assembly. Option 2 utilizes Extravehicular Activity (EVA) crew to directly assemble, connect and verify aerobrake construction. As can be seen in table 2.3.1-1, resource comparisons show equivalent levels of total man-hours to perform the aerobrake assembly, whether accomplished using telerobotics or EVA. However, the use of EVA crewmen imply a substantial operational cost premium over IVA crew usage.

Table 2.3.1-1 Aerobrake Assembly Trade Study Results

Option	Man-Hours (EVA/Total)	Serial Task Hours	Technology Risk	Comments
IVA/Telerobotic	0/280.2	140.1	101/150 (High)	Also Requires EVA Dev't
EVA Assisted	125.8/276.3	91.2	97/150 (Med High)	Uses STV turntable

As a result of studying the aerobrake assembly operations, a set of design recommendations were produced. The significant point involves design of a simply sealing thermal protection system along with positively latching joint mechanisms. If adopted, these recommendations would offer a 28% improvement in assembly time for the telerobotics option, making it comparable to the EVA option. Other key design recommendations relate to latches, adjustable struts, alternate TPS closeout, and the STV turntable. With regards to the latches, recommendations include self-alignment/verification, recycle, and positive latching.

To properly understand the impacts and sensitivities of the Space Station system due to STV servicing operations, a study was conducted that examined each proposed STV configuration and evaluated the complexity of its individual servicing operations.

The study initially defined an exhaustive list of STV servicing tasks against which the complexity of each task were derived. Time estimates were developed for each configuration by multiplying each task complexity by this task duration, and summing for all tasks, the final complexity factor for each configuration was produced.

The complexity factors and crew time estimates were based on a dedicated STV servicing crew size of four, working consecutive two man shifts. For EVA operations, two EVA crewman would be assisted by a regular Space Station crewman to monitor operations. If the tasks are not undertaken by specifically trained STV servicing crewmen, then complexity factors could change. This speaks to the added issue of when additional crew habitation facilities will be needed for these special crewmen.

Results indicate that complexity factors of cargo configurations did not vary significantly. Similarly, the factors of crew configurations did not vary significantly. There was, however, a significant difference when comparing factors of crew and cargo configurations. The crew factors were much higher because of the post-flight inspections and refurbishment

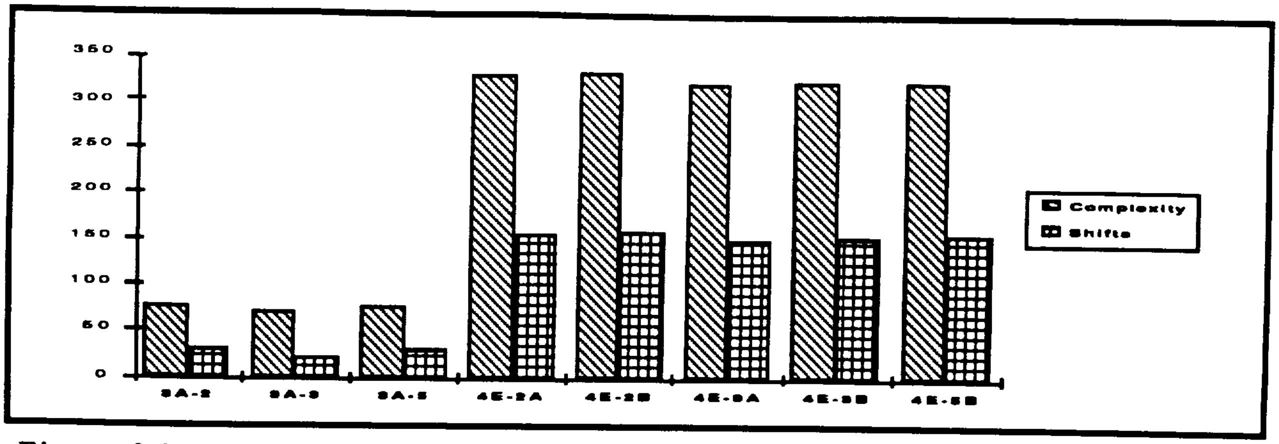


Figure 2.3.1-1 Sensitivity of STV Configurations to Servicing Operations

The final study conducted addressed the utilization of the SSF crew time. The basis of this study was an analysis of candidate tasks and shift times done by MDSE.-KSC (STV Concept Selection - SS Freedom On-Orbit Operations Evaluations - Preliminary Data - 6/2/90 by Don Bryant). The total shift-times in the study were multiplied by eight hours and four crew persons to get the total SSF crew hours for each type of mission. For purposes of comparison, 2800 hours was assumed to comprise a SSF man year to allow an approximate value of 18,000 man hours/year of utilization time (6 man crew). This was derived from currently hypothesized payload manifest scheduling and utilization operations extrapolated over a year.

Figures 2.3.1-2, 2.3.1-3, and 2.3.1-4 represent the results of this analysis. Figure 2.3.1-2 defines the total manpower required at SSF for each year of STV operations, Figure 2.3.1-3 translates these manpower requires into the actual percent of the total available manpower that these hours represent. Figure 2.3.1-4, defines the relationship between IVA and EVA during the coarse of the SSF servicing tasks. Initial assessment of the LEO operations requirements indicate a substantial manpower need. The goal of future studies as well as specific technology/advanced developments will be to drive toward a reduction in this requirements, that in turn reduces operational costs and risks.

Flight Operations Analysis—The flight operations analysis has been separated into two areas. The primary area of activity involved analysis of lunar missions including trajectories, aeroassist maneuvers, and mission times. The secondary area of analysis addressed a ground-based approach involving a high energy stage in support of meeting the STV DRMs.

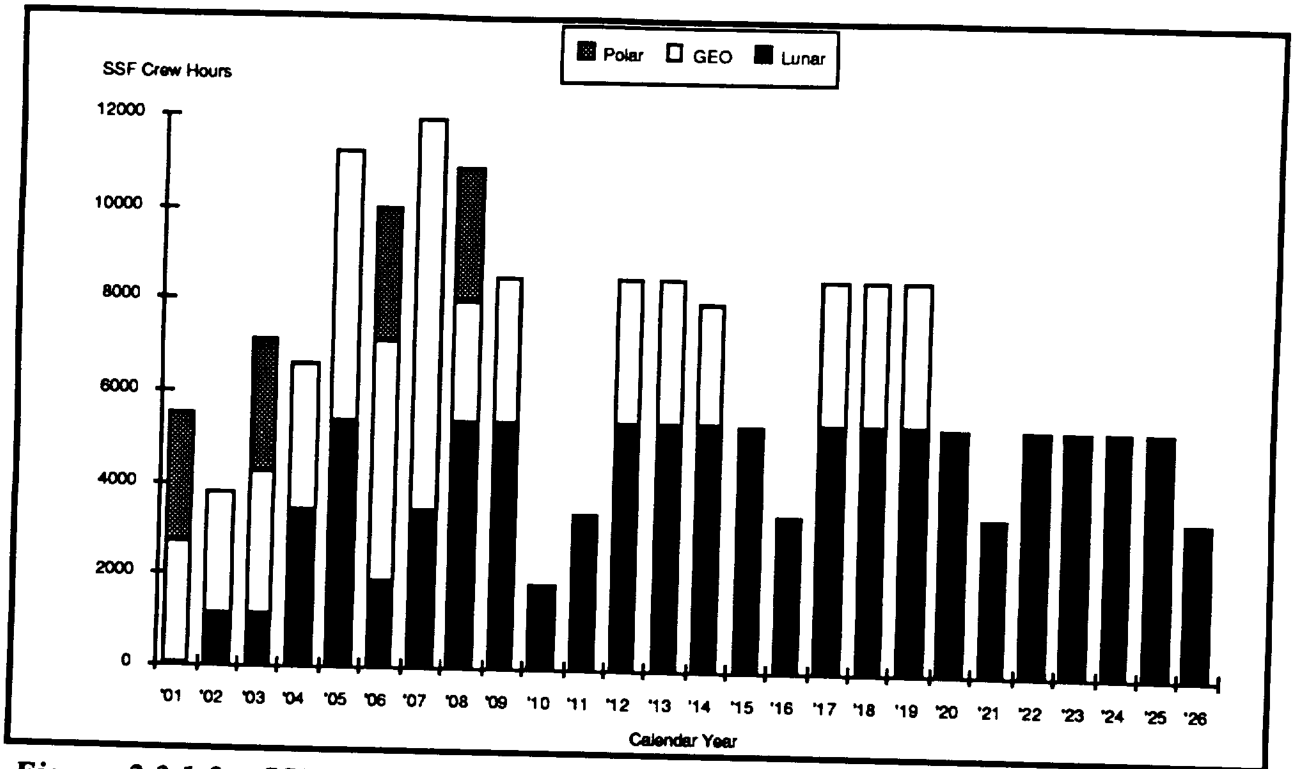


Figure 2.3.1-2 SSF Crewtime to Support STV Operations

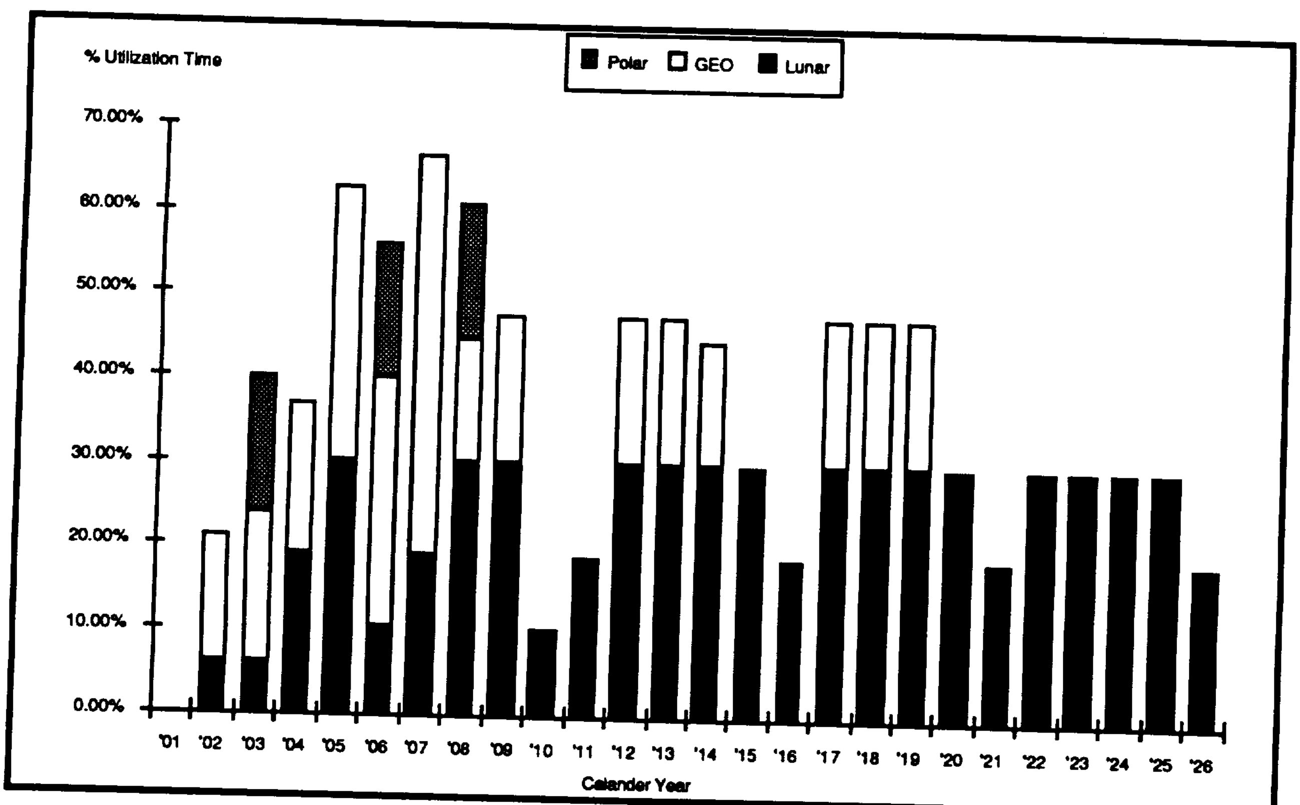


Figure 2.3.1-3 % SSF Utilization Time to Support STV Operations

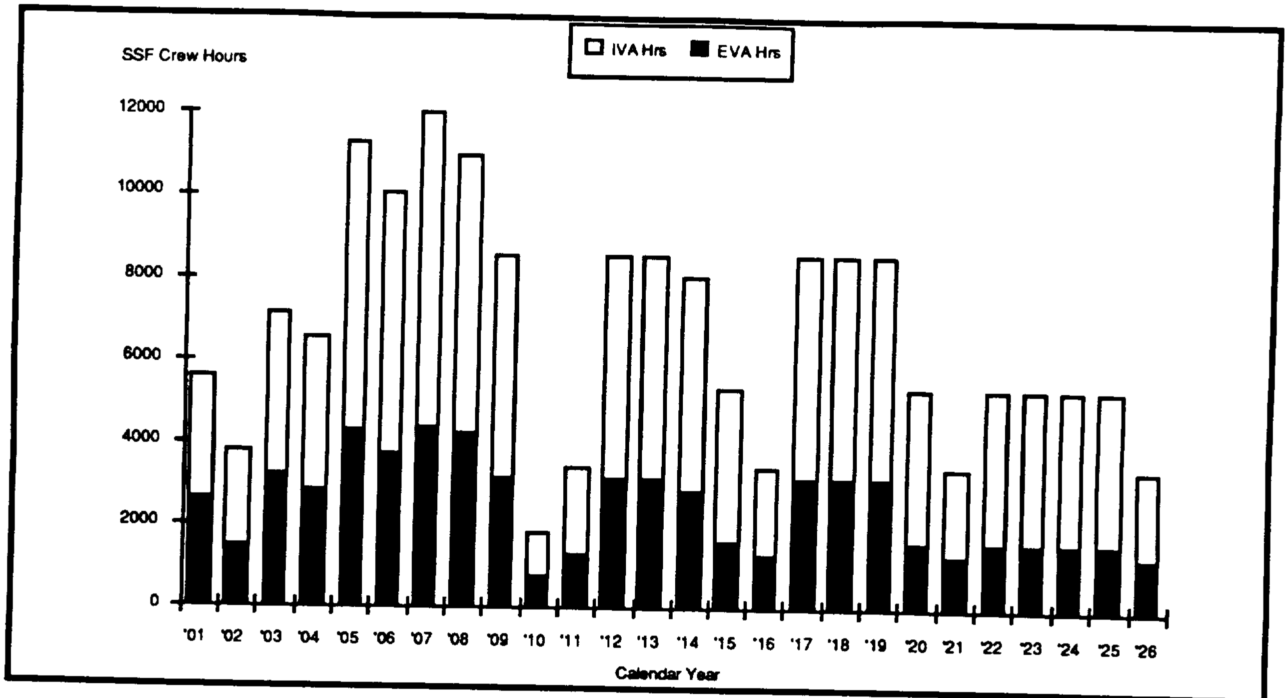


Figure 2.3.1-4 EVA / IVA Time Required to Support STV Operations

The analysis and the recommendation of a baseline and alternative architectures was constrained by several sizing groundrules and assumptions, Table 2.3.1-2.

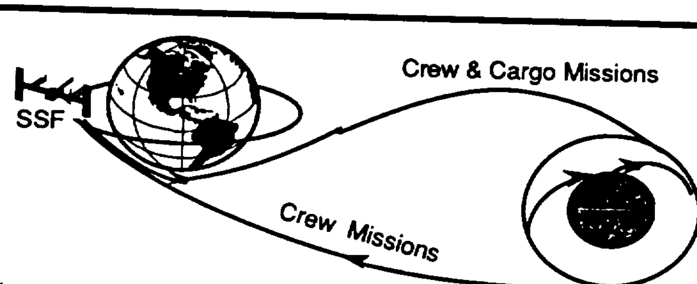
Table 2.3.1-2 Groundrules and Assumptions

Tank Fraction	4% of Propellant
Leg Fraction	2% of Landed Mass
Structure	2% of Gross Vehicle Mass (no P/L)
Aerobrake	20% of Vehicle Gross at Aeroentry
Engine T/W	30
Vehicle T/W	
Earth Escape	0.25
Lunar Surface	0.5
2nd Stage TV	0.1
Flight Performance Reserve	2% by Velocity
Unusable Propellant	1.56% of Total Propellant
Avionics	0 (In the noise)

- TV-Crew Module Mass, including 4 crew, suits and consumables: 9760
- LV-Crew Module Mass, including 4 crew, suits and consumables: 3130
- Single Stage combined Vehicle Expends the Following on the lunar surface: Structure mass and Leg mass
- Multi-Stage vehicles driven to common size
- Drop Tanks always dropped after TLI
- Drop tanks sized for Entire Propellant load
- Engine Performance Based on RL-10B-2 (Isp = 460 sec)

The configuration analysis task evaluated the five primary mission architectures shown in Figure 2.3.1-5. The recommendation to use the LEO Transportation Node as the baseline lunar mission architecture, see Figure 2.3.1-6, was based on, cost, risk, operations, and mission adaptability. It should be recognized that this decision is dependent on the assumptions that were made, as well as the relative weighting of the various selection criteria. Once the baseline mission architecture and trajectory were defined, a detailed analysis was conducted to optimize the effect of one-way transfer time on the total propellant load, assuming that both legs of the round-trip mission had the same one-way time. A free return trajectory with a lunar fly-by altitude of 300 km would have a one-way transfer time of ~71 hours, with transfer time increasing (up to ~120 hours) with increasing lunar fly-by altitude. The minimum one-way transfer time for a free return is ~68 hours (0 km lunar fly-by). The left border on the graph represents a parabolic Earth departure and is not a physical boundary, i.e., hyperbolic earth departures and lunar orbital captures are possible. However, the right border on the graph is a physical boundary and represents the lowest energy elliptical transfer possible.

To supplement this trajectory analysis, a strategy was developed that would allow for the return of the vehicle, and crew to SSF without the use of a separate rescue vehicle.



Baseline Option

1: LEO Transportation Node
Cargo - LEO to Lunar Surface
Crew - LEO to Lunar Surface, Return to LEO

2: LEO Crew Node
Cargo - Earth to Lunar Surface
Crew - LEO to Lunar Surface, Return to LEO

3: No Transportation Node (Apollo)
Cargo - Earth to Lunar Surface
Crew - Earth to Lunar Surface, Return to Earth

4: LEO Crew Return Node
Cargo - Earth to Lunar Surface
Crew - Earth to Lunar Surface, Return to LEO

5: LEO Crew Node/Earth Return
Cargo - Earth to Lunar Surface
Crew - LEO to Lunar Surface, Return to Earth

Figure 2.3.1-5 Lunar Mission Architectures

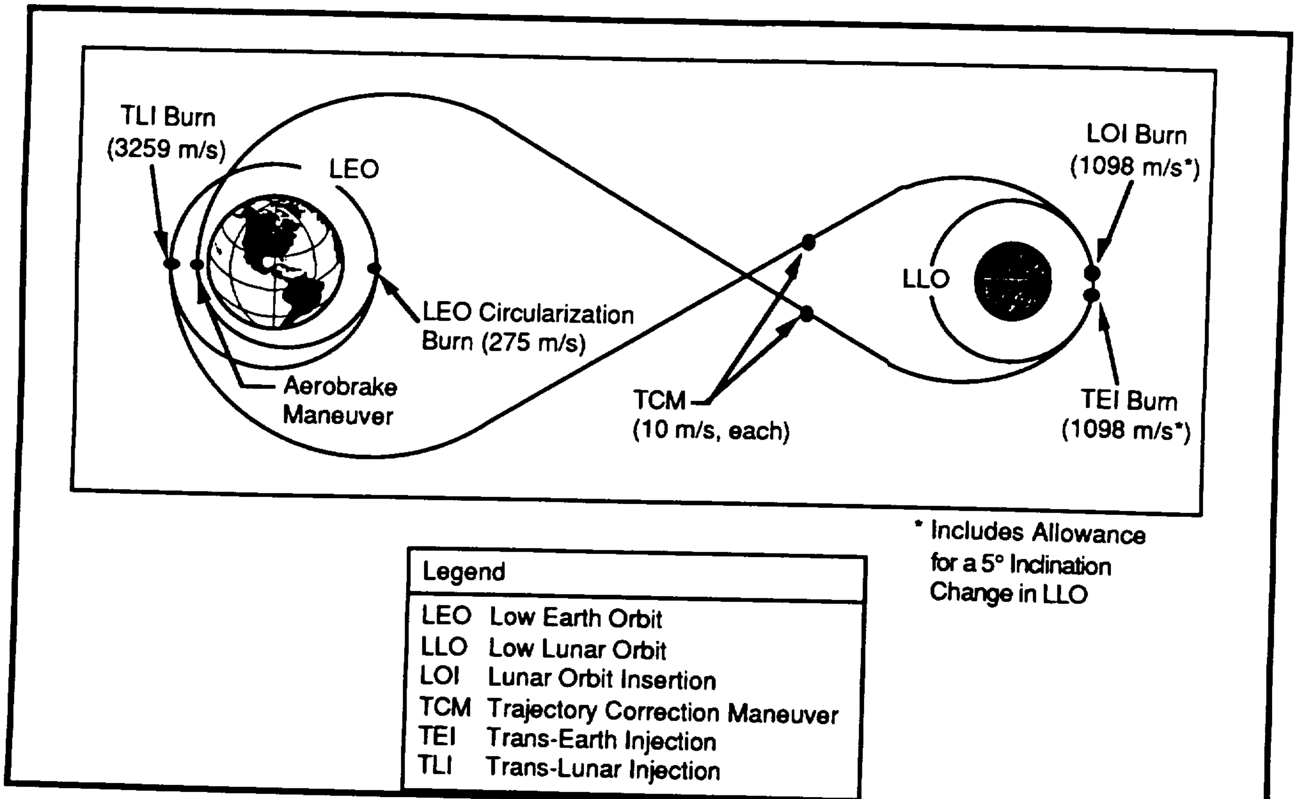


Figure 2.3.1-6 Baseline Earth-Lunar Trajectory

The initial step begins with the vehicle departing on a 71.1 hour free return trajectory to the moon, with a lunar fly-by altitude of 300 kilometer. Once the decision has been made to execute the free return for some reason, the vehicle would perform the 300 km lunar fly-by and embark on the 71.1 hour return to Earth. Once at Earth, the vehicle would begin the second step, performing a 102 meter/second retro-burn at periapsis to change the vehicle's orbit from a 407 x 518814 kilometer, 15.8-day orbit to a 407 x 202800 kilometer, 4.1-day orbit. The vehicle would then remain in that holding orbit for 11 complete orbits (~45 days), allowing SSF's orbit to precess into the plane of the elliptical orbit. After the orbital planes are realigned, the vehicle would make the final 3003 meter/second retro-burn to insert into SSF's orbit and then rendezvous with SSF. Our baseline vehicle would employ its aerobrake to achieve both the 102 meter/second and 3003 meter/second Δ-velocities if its main propulsion had failed. Because the vehicle would pass through the Van Allen radiation belts several times while waiting for SSF rendezvous, it might seem that the crew would be exposed to an inordinate amount of radiation. However, a separate study has determined that the crew's exposure to radiation while in a 4-day orbit is actually less than it would be for the same amount of time spent in LEO.

Since a direct free return to SSF is generally not possible due to the plane in which the vehicle returns to Earth not aligning with SSF's orbital plane, this strategy uses two steps to achieve the recovery of the vehicle at SSF.

Alternative HLLV Upper Stage Configuration—Since the baseline STV presented in the rest of this document is dominated by requirements that came from the 1989 90-Day Report (Skunk Works), it is important to assess what requirements could be generated without the emphasis on space-basing and reusability. Figure 2.3.1-7 shows how three important mission classes all require about 4 kilometer/second Δ-velocity from LEO, bringing a capability forward that for the commercial GSO market and the two objectives of SEI - the moon and Mars - a common stage is possible.

The study assumed the use of a circular park orbit at 185 kilometers and 28.5 degrees. This park orbit was used because most high energy missions use LEO to minimize their total mission Δ-velocity by selecting the optimum time to start the transfer burn, i.e., nodal crossing. LEO is also used for final targeting and improves mission flexibility by increasing the width of the ETO launch window. In all cases, the booster vehicle consisted of two Advanced Solid Rocket Motors

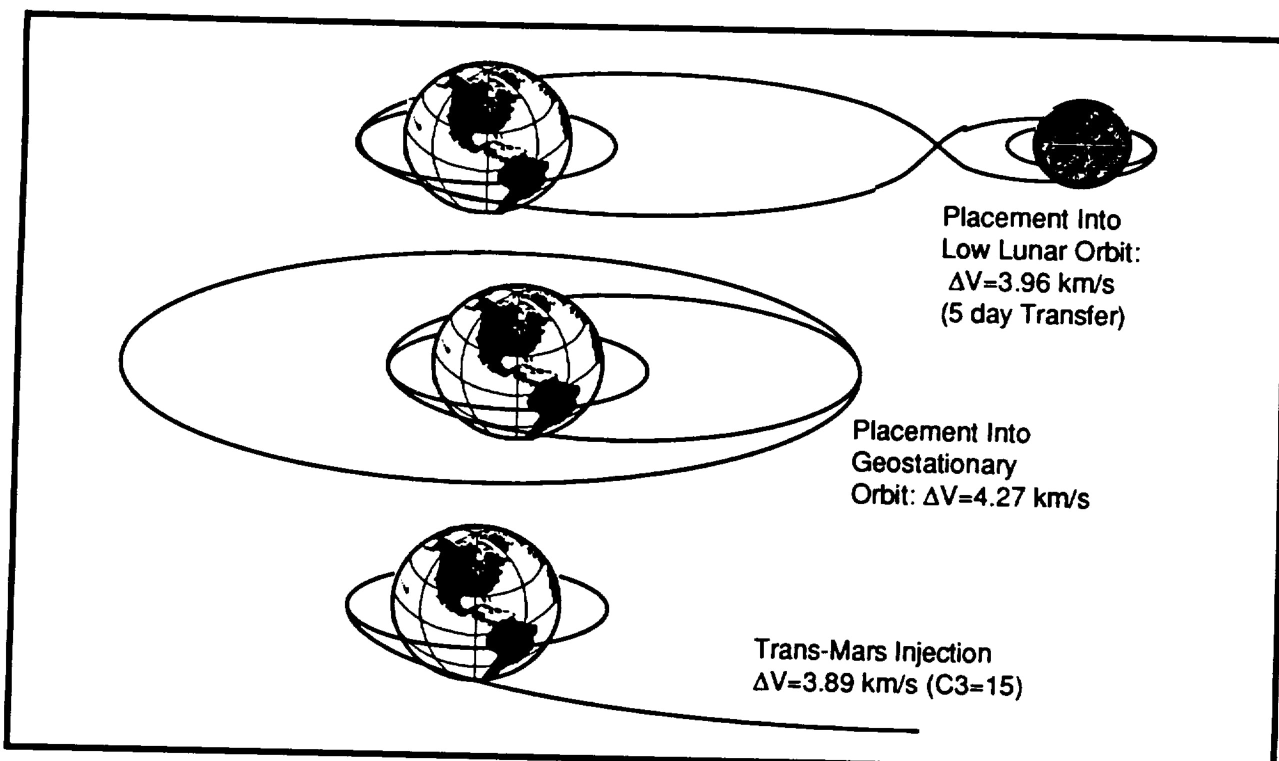


Figure 2.3.1-7 Similar Δ-Velocity Mission Requirements

(ASRMs), an External Tank (ET) derived core, and a payload shroud based on our Advanced Launch System work. The differences lie in the type and number of engines used, and the manner in which they were mounted on the core. The two engines considered were the Space Shuttle Main Engine (SSME) and the Space Transportation Main Engine (STME). These engines were used in sets of three and four and were mounted in either a side-mount or in-line fashion.

The characteristics for each of these engines are shown. The upper stages were sized parametrically, but all were based on the assumptions listed on the chart. The upper stages had thrust levels ranging from 444 kilonewtons (100 kilopounds) to 1332 kilonewtons (300 kilopounds) and propellant loads ranging from 45 tonnes (100 kilopounds) to 160 tonnes (350 kilopounds).

The performance advantages that this stage offers are shown in Figure 2.3.1-8. By going to three ASRMs and extending the length of the ET, the 1.5 stage HLLV has been sized to match the LEO capability of one of the eight 2.5 stage vehicles evaluated. But as the Δ-velocity increases, the capability of the 1.5 stage HLLV falls off much more rapidly than does the capability of the 2.5 stage vehicle.

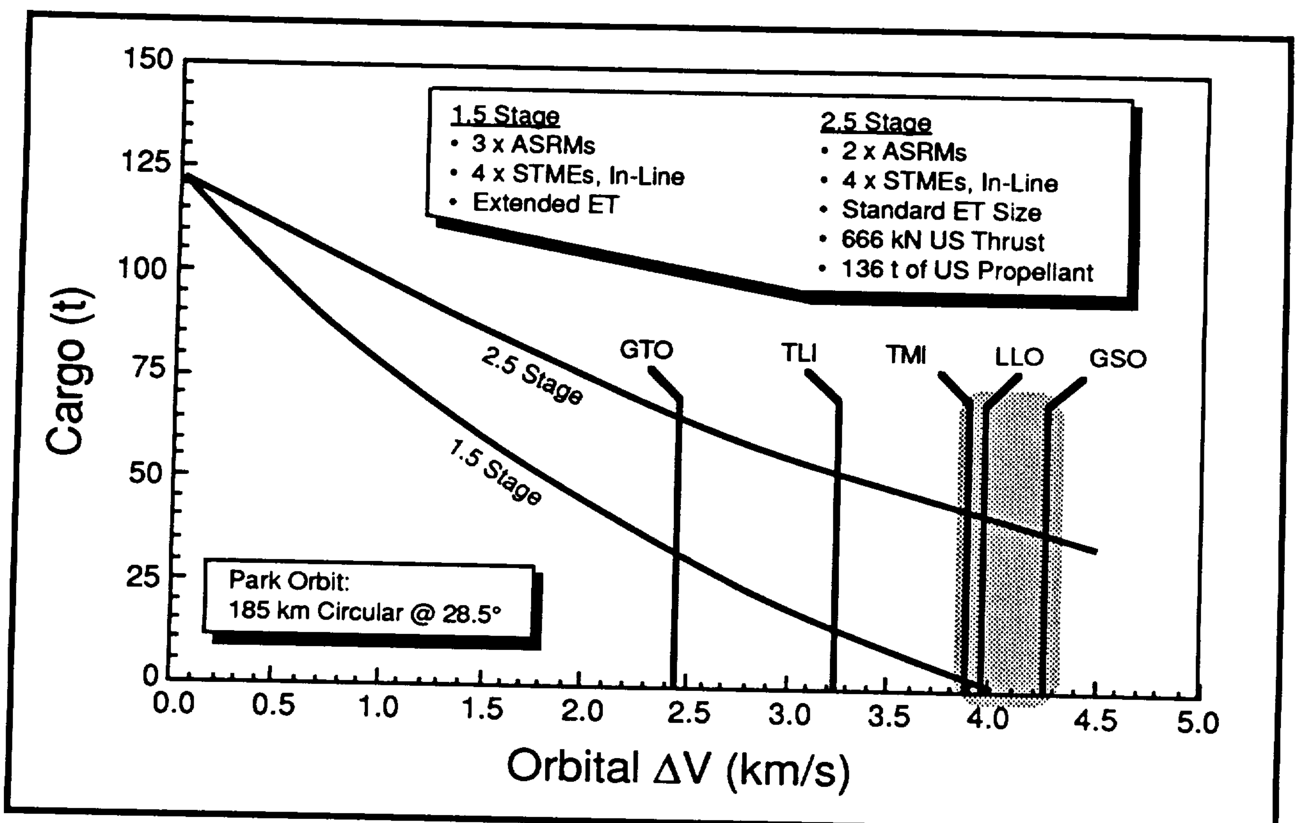


Figure 2.3.1-8 Cargo versus Orbital Δ-Velocity With/Without an Upper Stage

For example, the Geostationary Transfer Orbit (GTO) capability of the 2.5 stage vehicle is roughly twice that of the 1.5 stage HLLV. Furthermore, at 4 km/s the 1.5 stage HLLVs capability drops to zero while the 2.5 stage vehicle gets ~45 tonnes. The three missions previously mentioned as having a Δ-velocity of approximately 4 km/s have been highlighted.

Analysis of the 4 kilometer/second stage was conducted over a range of potential HLLV systems since the exact configuration and capabilities of the HLLV have not been formulated. Even with the fluctuations in the defines of an HLLV system, the results of this analysis show a clear requirement to consider the integration of a high energy upper stage into the HLLV configuration for both a near earth as well as planetary exploration and manned missions.

2.3.2 Systems Analysis

Following the definition of the STV requirements base and in conjunction with the mission analysis effort, three major system studies were conducted. These studies included basing, aeroassist, and design. Within these analysis the implementation of man-rating on the transportation system was evaluated along with the systems programmatics that included test, cost, and schedules.

Aeroassist vs All-propulsive Analysis —The objective of the aeroassist versus all-propulsive study was to determine relative life cycle cost (LCC) benefits as a function of the aerobrake mass fraction, ETO specific costs ($/mass), and the costs associated with development of the aerobrake. The study showed that even if greater aerobrake mass fractions are required than currently estimated (11% to 15%), the life cycle cost benefits are still substantial, see Figure 2.3.2-1. One of the more critical elements in establishing aerobrake and total system development cost is the question of the need for subscale flight testing. Preliminary studies have shown that flight testing an approximately half scale prototype aerobrake could be accomplished using the existing STS as the launch vehicle. However, such a test or tests would add significantly to the cost of aerobrake development. Further assessment of the pros and cons of such testing is required. Relative to the issue of aerobrake reusability, the LCC cost study results suggest that, depending on development costs, the cost advantage the aerobrake affords should not disappear even if it is only used one time. (Complications in ETO manifesting associated with replacement of the aerobrake more frequently than other subsystems have not been evaluated). Another concern, afterbody heat protection during the aerobrake maneuver, also has not been evaluated sufficiently due to wake heating uncertainties. There appears to be room to increase system mass for this purpose without significantly eroding the cost advantages of the aerobrake approach, although adding heat protection to the core vehicle has a two to three times greater impact on IMLEO mass as does adding mass to the aerobrake since the core vehicle descends to the lunar surface.

Space versus Ground Basing Analysis—The objective of the space versus ground basing analysis was to provide a means of course screening for the large configuration selection analysis. The configurations that space-based system and a ground-based systems were based on, had been defined as a result of information derived from the 1989 Skunk Works activities.

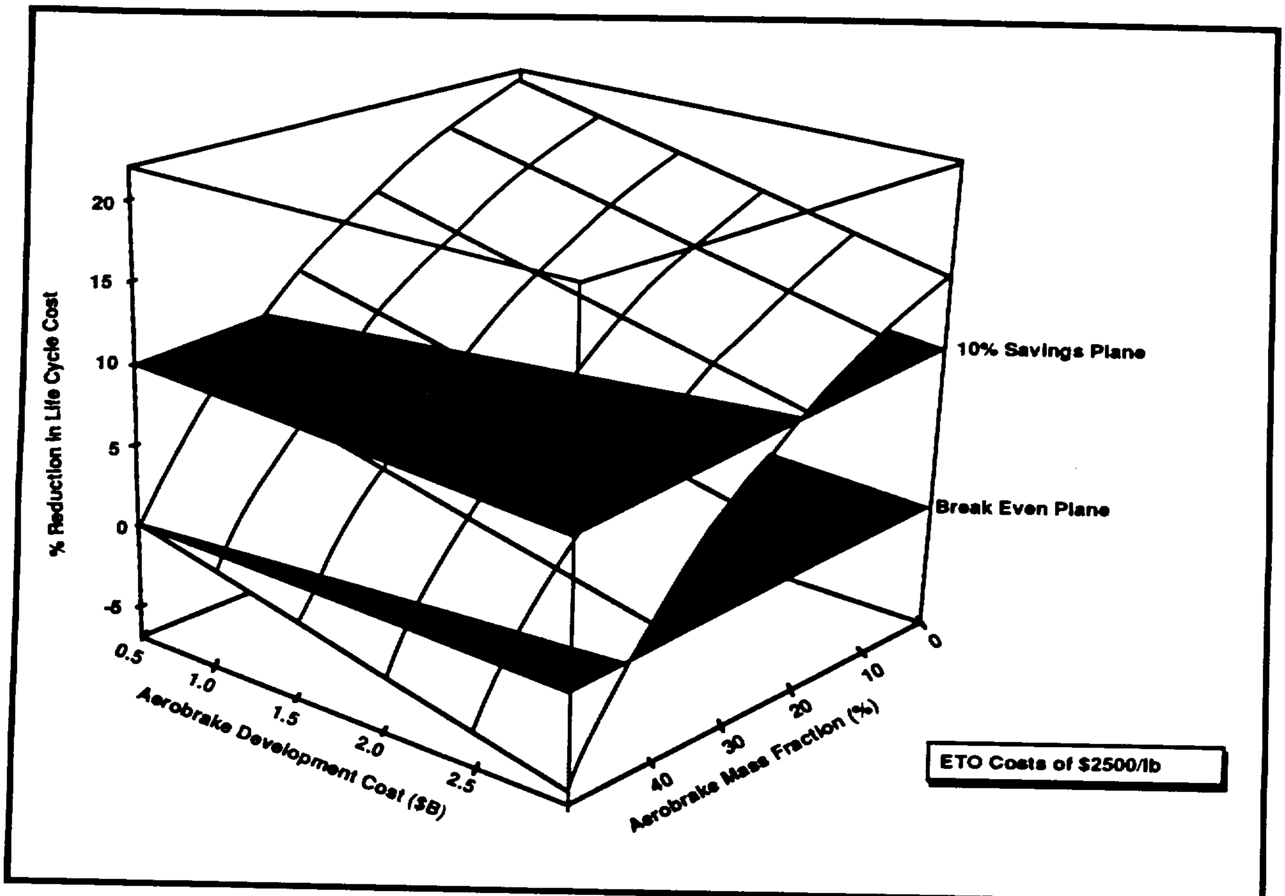


Figure 2.3.2-1 Aerobrake LCC Saving Relative to All Propulsive

The ground-based system is comprised of an expendable transfer stage with a ballistic return lander. Details of the configurations used to assess these criteria are shown in Figure 2.3.2-2.

The spaced-based configuration is comprised of a multiple stage system with drop tanks for propellant storage and crew module. At the initiation of this analysis it was determined that cost and operations were the most important of the four primary analysis criteria under which the STV studies have been performed. Program cost defines the total cost to acquire and operate the

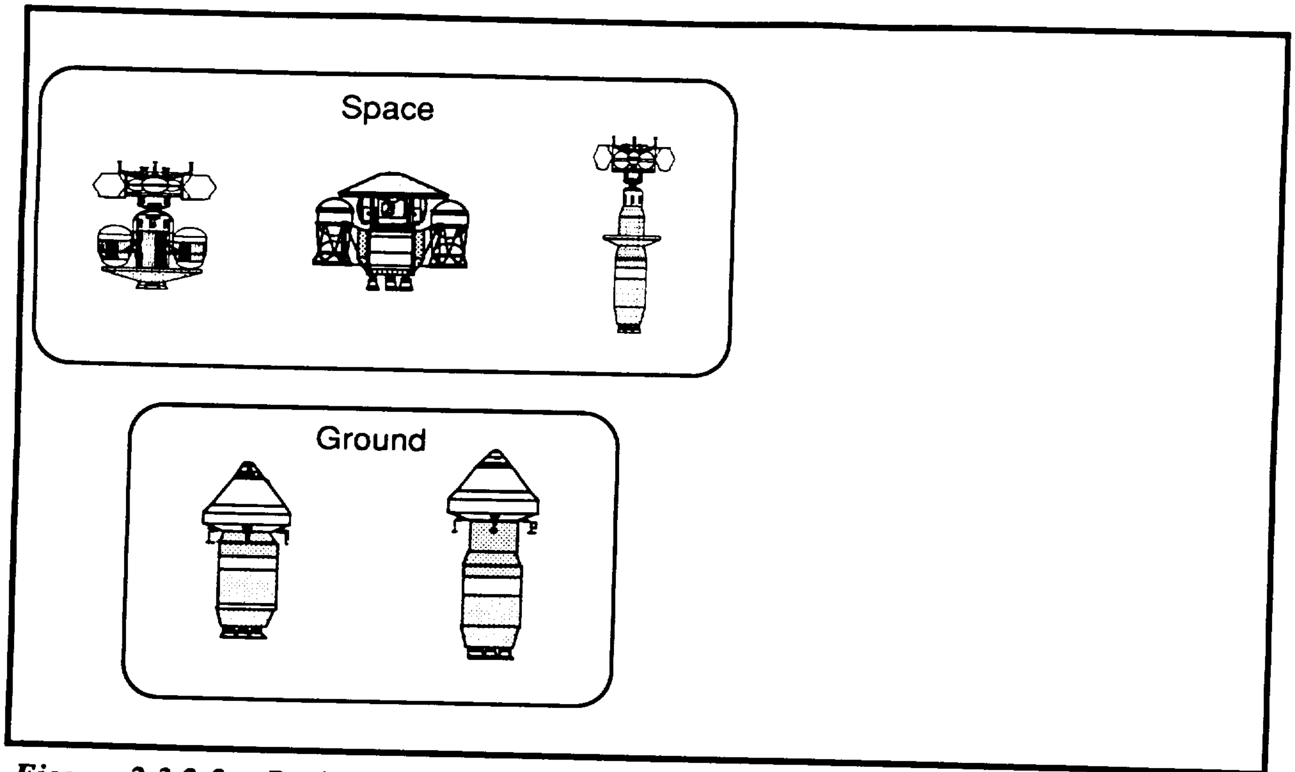


Figure 2.3.2-2 Basing Configuration Candidates

system. This total cost includes: Full Scale Development (FSD), verification, production, operations and support, and disposal. The operations analyses included both space and ground functions. The operational functions included rendezvous and docking both at Low Lunar Orbit (LLO) and Low Earth Orbit (LEO); Engine Burns at Trans-Lunar Injection (TLI), LLO, lunar landing, ascent, and Trans-Earth Injection (TEI); system element separations including stages and drop tanks; crew, cargo and propellant transfers; and critical maneuvers including aerobrake preparation and operation and a ballistics return. Each of these functions was assigned either a Crit 1 or 2 rating, which provided a quantitative value to the criticality of the operation. A Crit 1 operation is defined as an operation which if not successfully completed results in loss of life or failure to deliver mission critical cargo. Crit 2 is defined as an operation which if not successfully completed allows the crew to return safely or leaves the cargo in a position where it can be salvaged.

The following groundrules were observed in conducting this analysis:

- Propellant shall be cryogenic
- Earth return shall be aeroassisted (derived from results of the Aeroassist vs All-propulsive Return Study, 2.3.3.1)
- ASE engine shall be used on transfer vehicle (Isp - 476) and transfer/landing vehicle

(Isp - 460)
- ETO transportation system cost shall be $2500/lb
- LCC shall include design, development, test hardware and operations
- System life shall be 30 years
- Space basing shall utilize SSF, requiring $2.0 billion for modifications

The results of the cost evaluations are shown in Figure 2.3.2-3. This data shows that in three of the four cost categories the space-based systems represent a lower cost, including LCC. The only

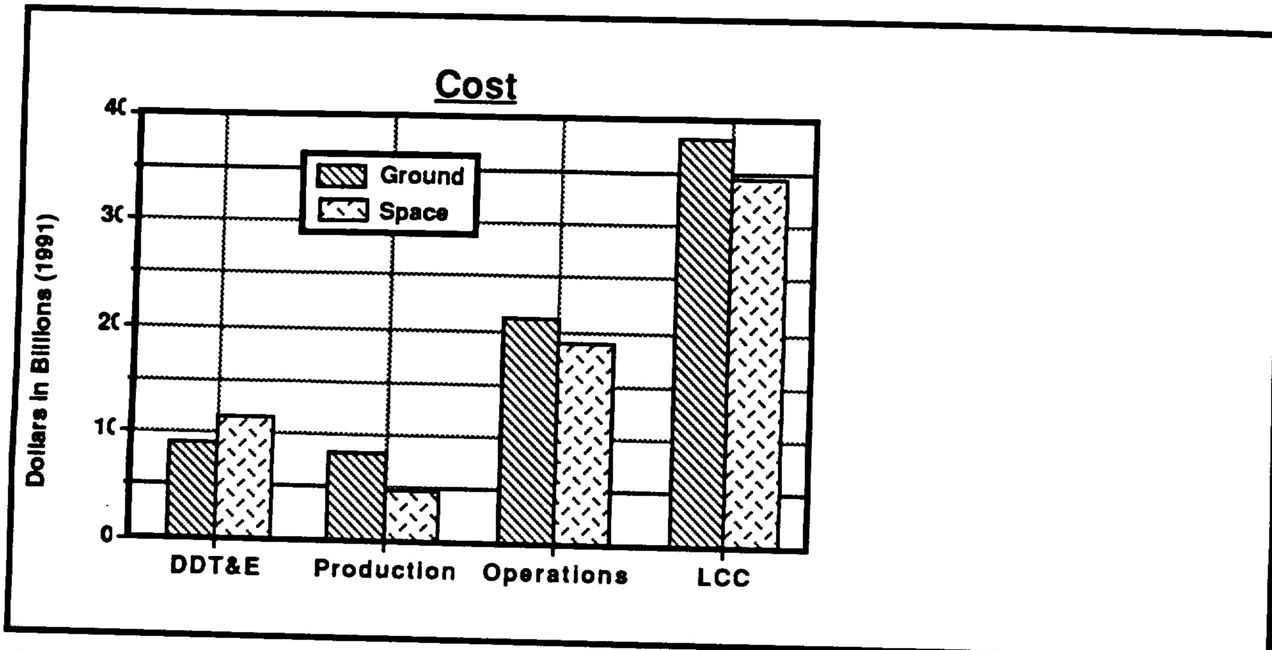


Figure 2.3.2-3 Basing Cost Analysis Results

category in which the ground-based system rated better in cost was in DDT&E since the ground-based system utilizes fewer technology/advanced development items that require extensive development costs. The results of the operations evaluation, shown in Figure 2.3.2-4, show the opposite trend, with the ground-based system representing an approach with fewer critical failure modes during the conduct of the transfer missions. This can be attributed to fewer rendezvous and docking operations and the elimination of the aerobrake and the aeroassist maneuver. Further assessment of the operational complexity based on ground processing operations was conducted to cast a deciding vote in providing a recommendation from this analysis. This additional work indicated that the ground-based system greatly increased the processing requirements at KSC.

The results of this basing evaluation provided significant data to recommend a basing approach that utilizes a LEO transportation node and space-basing the LTS. This provides an overall reduction in the system LCC of 9% and a similar approach to ground processing and launch at KSC. It should be noted that although this approach provides a lower cost, it does represent a system with more potential failure modes, that must be accounted for in the final design.

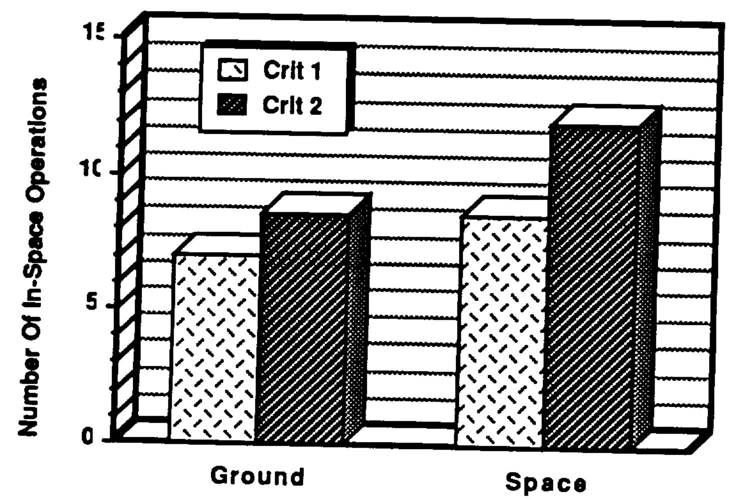


Figure 2.3.2-4 ***Basing Operations Analysis Results***

STV Concept Selection Analysis—There are two basic STV concept selection philosophies. The first is to start with a ground-based initial STV, proceed to space-based reusable concepts, and continue to utilize the STV or family of STV vehicles for lunar missions and eventually Mars missions. A second philosophy starts with the most mission driven STV concept — the lunar mission — and evolves backwards and forwards to satisfy the other missions. These two philosophies are illustrated in Figure 2.3.2-5. Since the Lunar missions represent the most stringent drivers for vehicle definition, the concept selection philosophy of starting with the lunar STV family and evolving to the other design reference missions (DRMs) was utilized for this top level systems trade.

The concept selection process chosen for this analysis, Figure 2.3.2-6, was established to systematically evaluate and down select STV concepts into a single concept or family of concepts. The process began with the development of a concept selection methodology and was followed by a concept identification task. Once concepts were defined, simple configurations, operational scenarios, performance data and relative cost data were generated for each concept. Concepts were evaluated against top level selection criteria — performance, relative cost, and operational complexity. Top scoring concepts for each selection criteria were recommended for additional

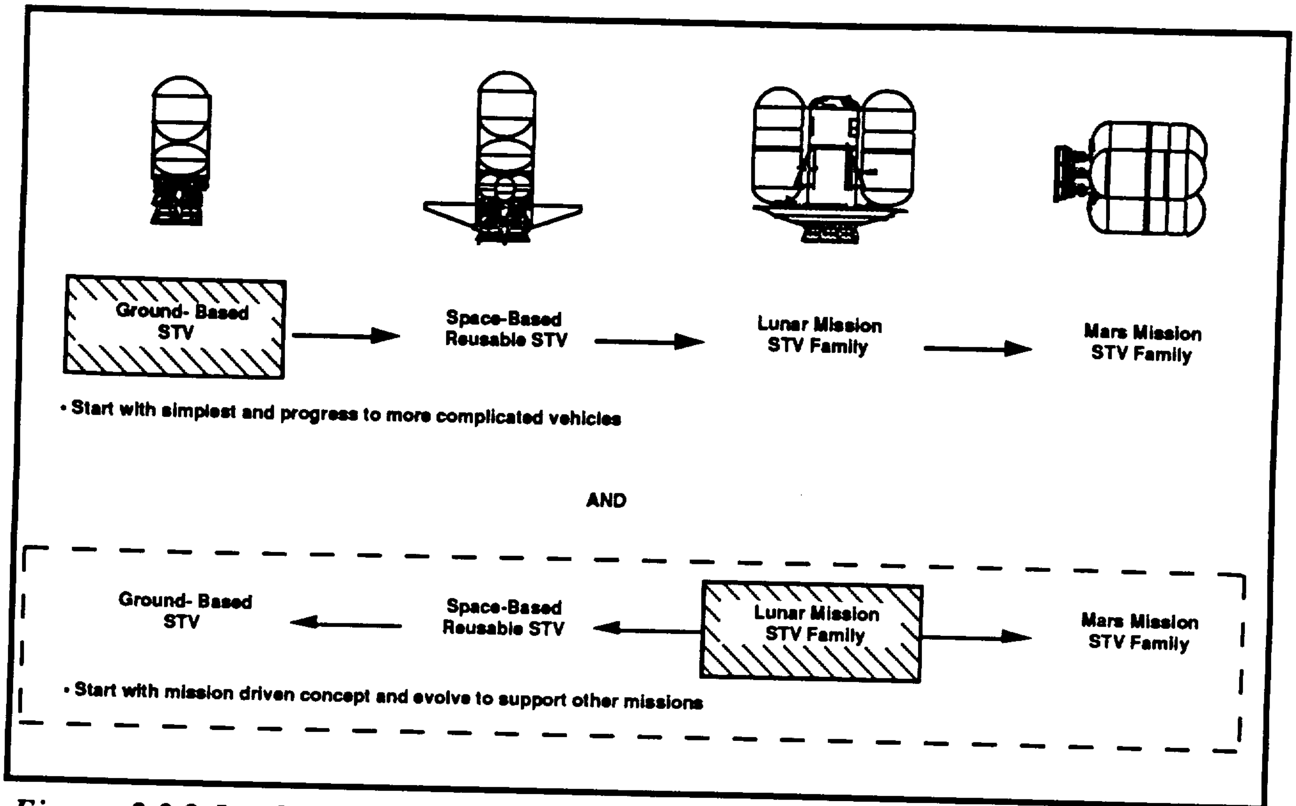


Figure 2.3.2-5 Concept Selection Philosophy

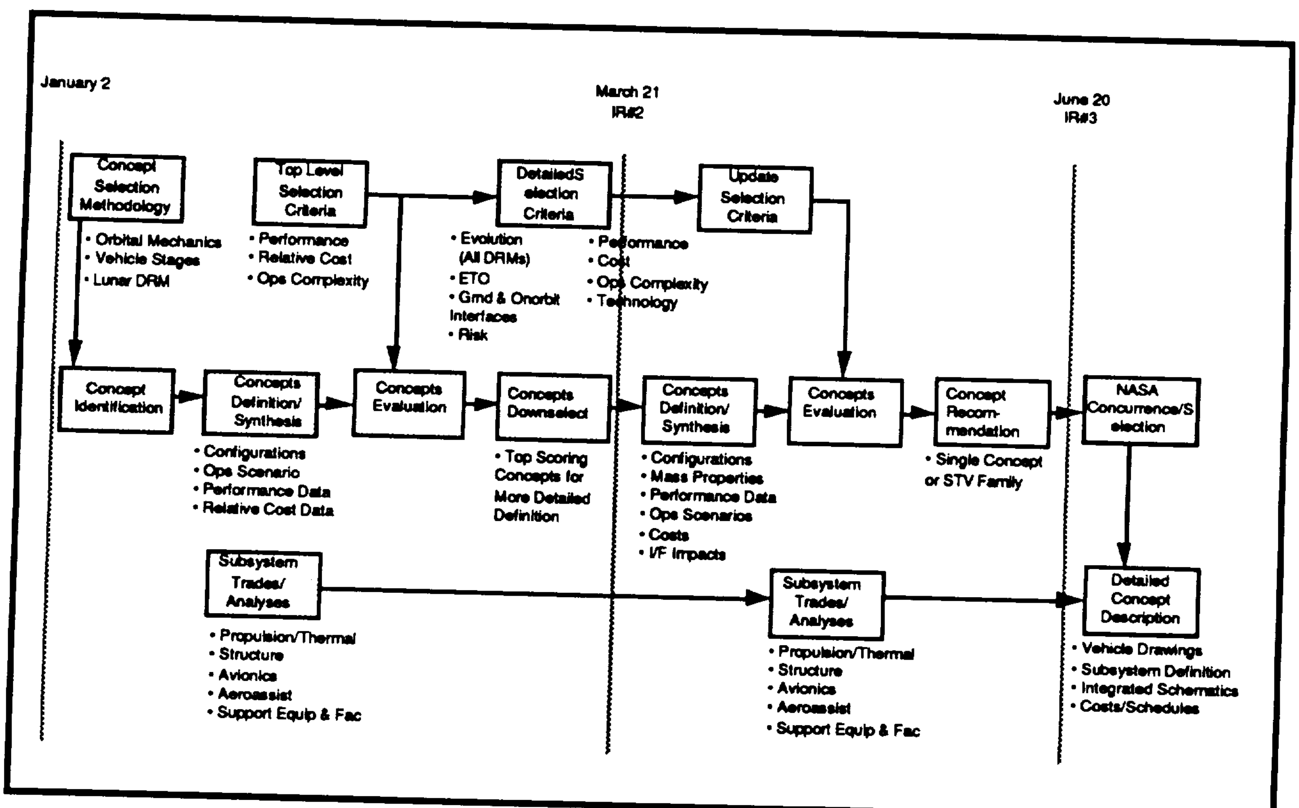


Figure 2.3.2-6 Concept Selection Process

evaluation during the second downselect process. Downselected concepts were further defined and evaluated to determine interface impacts, real costs, evolution to other missions, ETO transportation methods, etc.

After the first downselect, lunar architectures were developed and concepts were allocated to these architectures. More detailed data consisting of configurations, mass properties, performance results, flight operational scenarios, interface impacts and programmatic costs were generated for each concept. Cost, operations, adaptability to meet other DRMs, and risk were used as evaluation criteria to recommend criteria driven concepts for additional study during the final downselect.

The criteria driven concepts were further studied to define a common family of vehicles and assess abort scenarios. Results from these final studies were evaluated, and a final STV family of vehicles was selected. Once NASA concurred with the final STV selection, results from subsystems trades were incorporated and detailed concept description of the selected concept and detailed programmatics were conducted.

The first step in the downselect process was to identify orbital mechanics solutions for delivering crew and/or cargo to the Moon. Figure 2.3.2-7 is a pictorial overview of the node options available for lunar transfer and return. Nodes, which were defined as locations where two vehicles

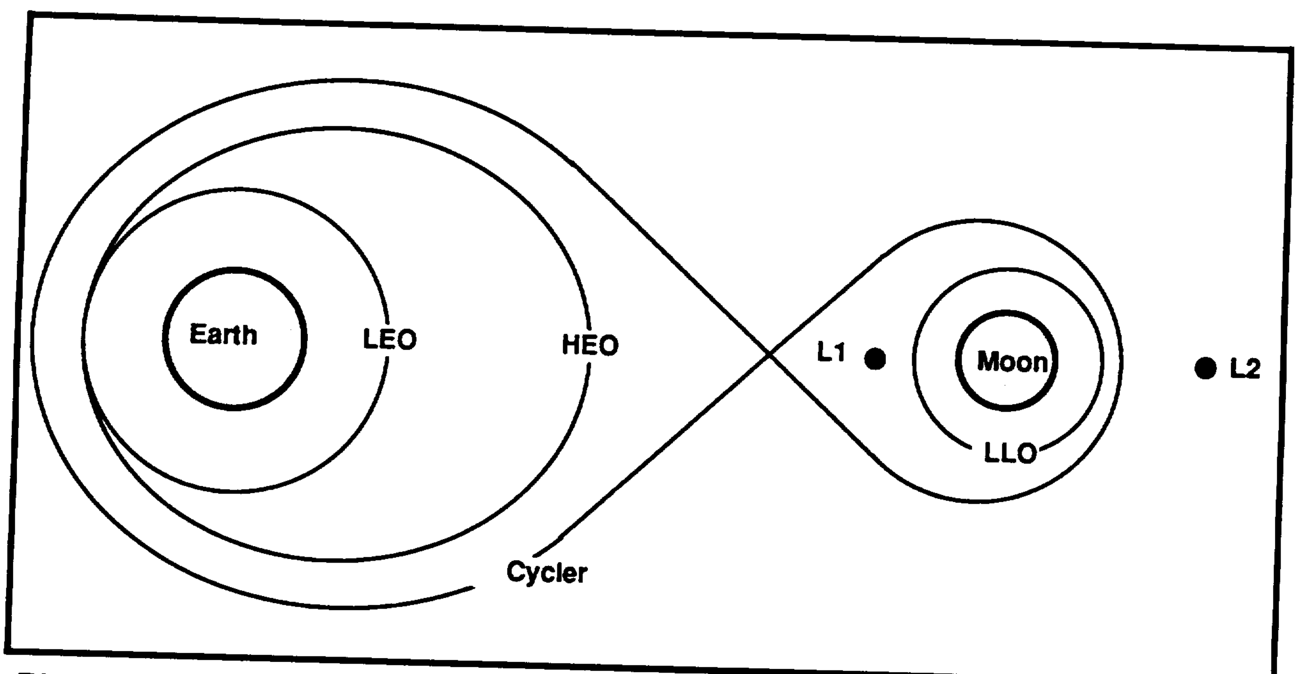


Figure 2.3.2-7 Lunar Mission Orbital Mechanics Options

can meet to transfer people, cargo, and propellant, included Low Earth Orbit (LEO), a Highly Elliptic [Earth] Orbit (HEO), an orbit with a perigee near Space Station Freedom (SSF) altitude and a period that is resonant with the sidereal rate of the moon. L1 and L2 were also evaluated as nodes. L1 is the libration point on a line between the Earth and moon. L2 is a similar point, but located on the far side of the moon, still on the Earth-moon line. A cycler which is a continually moving node that is placed in a resonate, free-return trajectory between the Earth and Moon was also defined.

The reverse process is followed for getting back to Earth. The final node considered was Low Lunar Orbit (LLO), typically a 300 kilometer circular orbit with an inclination of less than 30 degrees.

Using these node options the all possible orbital mechanics solutions to launch and/or return cargo and/or crew from the Earth to the Moon were developed and are listed below:

Launch - Up Leg from Earth or Low Earth Orbit (LEO) to Lunar Surface

1 Earth to Lunar Surface
2 Earth to Low Lunar Orbit (LLO) to Lunar Surface
3 LEO to Lunar Surface
4 LEO to LLO to Lunar Surface
5 Earth to Libration Point to Lunar Surface
6 Earth to Highly Elliptic [Earth] Orbit (HEO) to Lunar Surface
7 Earth to Cycler to Lunar Surface
8 LEO to Libration Point to Lunar Surface
9 LEO to HEO to Lunar Surface
10 LEO to Cycler to Lunar Surface

Return - Down Leg from Lunar Surface

A No Return
B Direct Return from Lunar Surface to Earth
C Direct Return from Lunar Surface to LEO
D From Lunar Surface to LLO to Earth
E From Lunar Surface to LLO to LEO
F From Lunar Surface to Libration Point to Earth
G From Lunar Surface to Libration Point to LEO

H From Lunar Surface to HEO to Earth
I From Lunar Surface to HEO to LEO
J From Lunar Surface to Cycler to Earth
K From Lunar Surface to Cycler to LEO

Using these orbital mechanics launch/return options a reasonable orbital solutions matrix used to populate and develop a matrix of for the lunar mission shown in Figure 2.3.2-8.

Launch Pt - Up Leg From Earth / Leo

1 Earth To Lunar Surface
2 Earth To LLO Ops To Lunar Surface
3 LEO To Lunar Surface
4 LEO To LLO Ops To Lunar Surface
5 Earth To Libration Pt. To Lunar Surface
6 Earth To HEO To Lunar Surface
7 Earth To Cycler To Lunar Surface
8 LEO To Libration Pt. To Lunar Surface
9 LEO To HEO To Lunar Surface
10 LEO To Cycler To Lunar Surface

Return - Down Leg From Lunar Surface

A: No Return
B: Direct Return From Lunar Surface To Earth
C: Direct Return From Lunar Surface To LEO
D: From Lunar Surface To LLO Ops To Earth
E From Lunar Surface To LLO Ops To LEO
F From Lunar Surface To Libration Point To Earth
G: From Lunar Surface To Libration Point To LEO
H: From Lunar Surface To HEO To Earth
I From Lunar Surface To HEO To LEO
J: From Lunar Surface To Cycler To Earth
K: From Lunar Surface To Cycler To LEO

LAUNCH POINT	*	RETURN									
	A	B	C	D	E	F	G	H	I	J	K
1	X	X	X								
2	X			X	X						
3	X	X	X								
4	X			X	X						
5	X					X	X				
6	X							X	X		
7	X									X	X
8	X					X	X				
9								X	X		
10										X	X

* No Return

Figure 2.3.2-8 Orbital Solution Matrix

In order to reduce the number of orbit mechanics approaches, the Δ-velocities to complete either a one-way or round-trip mission to the Moon were calculated. All node options were considered except the cycler option which was eliminated on assumed cost grounds and operational complexities associated with lunar-to-Earth return and abort scenarios. Because L1 and L2 required more delta-V, they were eliminated as viable options. The HEO node scenario offers some advantages over using LLO — namely reduced Δ-velocity budget for the lunar transfer vehicle (LTV), however, from an opportunity point of view, HEO has distinct disadvantages over direct transfers and was therefore eliminated from further evaluation.

Figure 2.3.2-9, presents a downscaled orbital mechanics matrix of reasonable orbital mechanics solutions for the lunar mission after the libration point, HEO, and cycler options were removed. A vehicle stage matrix, Figure 2.3.2-10, was created based on the orbital mechanics matrix. Options for configuration candidates now consisted of 10 cargo only options were identified and 48 crew/cargo were identified.

Launch Pt. - Up Leg From Earth / Leo

1: Earth To Lunar Surface
2: Earth To LLO Ops To Lunar Surface
3: LEO To Lunar Surface
4: LEO To LLO Ops To Lunar Surface

Return - Down Leg From Lunar Surface

A: No Return
B: Direct Return From Lunar Surface To Earth
C: Direct Return From Lunar Surface To LEO
D: From Lunar Surface To LLO Ops To Earth
D: From Lunar Surface To LLO Ops To LEO

		*	RETURN			
		A	B	C	D	E
LAUNCH PT	1	X	X	X		
	2				X	X
	3	X	X	X		
	4				X	X

* No Return

HEO, Cycler, and Libration Point Solutions Deleted from Matrix

Figure 2.3.2.-9 Downscaled Orbital Mechanics Matrix

Preliminary operational scenarios and vehicle configurations were developed for each possible concept solution in the matrix. Performance analyses were run to determine vehicle propellant quantities required to deliver 33 tonnes of cargo for the no return concepts and 14.6 tonnes of crew/cargo for the manned return concepts. Each concept was also evaluated for operational complexity by determining number of elements, operations/maneuvers, transfers, matings, separations, etc. Relative cost data was generated for each concept by determining number of elements, ETO transportation requirements based on using a 150 klb launch vehicle, and SSF operations. This analysis data was input to an evaluation sheet , Figure 2.3.2-11, where trends were identified and candidate concepts were selected for additional study. From a detailed evaluation of the cost and operations data and trends, the three cargo configurations shown

		•	RETURN			
		A	B	C	D	E
LAUNCH PT	1	X	X	X		
	2				X	X
	3	X	X	X		
	4				X	X

• No Return

Launch Pt. - Up Leg From Earth / Leo

1: Earth To Lunar Surface
2: Earth To LLO Ops.. To Lunar Surface
3: LEO To Lunar Surface
4: LEO To LLO Ops.. To Lunar Surface

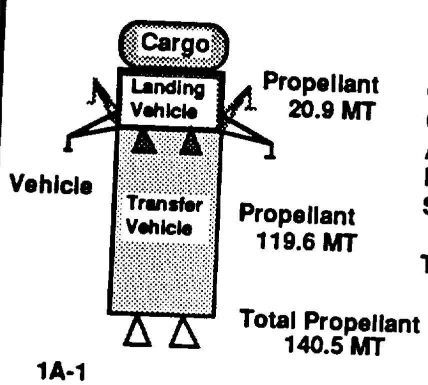
Return - Down Leg From Lunar Surface

A: No Return
B: Direct Return From Lunar Surface To Earth
C: Direct Return From Lunar Surface To LEO
D: From Lunar Surface To LLO Ops.. To Earth
E: From Lunar Surface To LLO Ops.. To LEO

		Separate Transfer Vehicle Separate Landing Vehicle (Multiple Propulsion Systems			Combined Transfer & Landing Vehicle (Single Propulsion System	
		Single Stage TV & LV 1	Single Stage TV & LV w/ Drop Tanks 2	Multistage TV & LV 3	Single Stage Combined Vehicle 4	Single Stage Combined Vehicle w/Drop Tks. 5
No Return From Lunar Surface	1A	x	x	x	x	x
	3A	x	x	x	x	x
Launch Point / Return	1B 1C	x	x	x	x	x
	2D	x	x	x	x	x
	2E	x x	x x	x x	N/A	x
	3B	x x	x x	x x	N/A	x
	3C	x	x	x	x	x
	4D	x	x	x	x	x
	4E	x x	x x	x x	N/A	x
		x x	x x	x x	N/A	x

Note: Two X's Indicate Single and Dual Crew Cab Concepts

Figure 2.3.2-10 Vehicle Stage Matrix



Relative Cost

Landing Vehicle	1
Transfer Vehicle	1
Drop Tanks	0
Combined Vehicle	0
Crew Capsule	0
Aerobrake	0
ETO Transp.	3.6
SSF Ops	0
Total Rel Cost	5.6

Ops Complexity

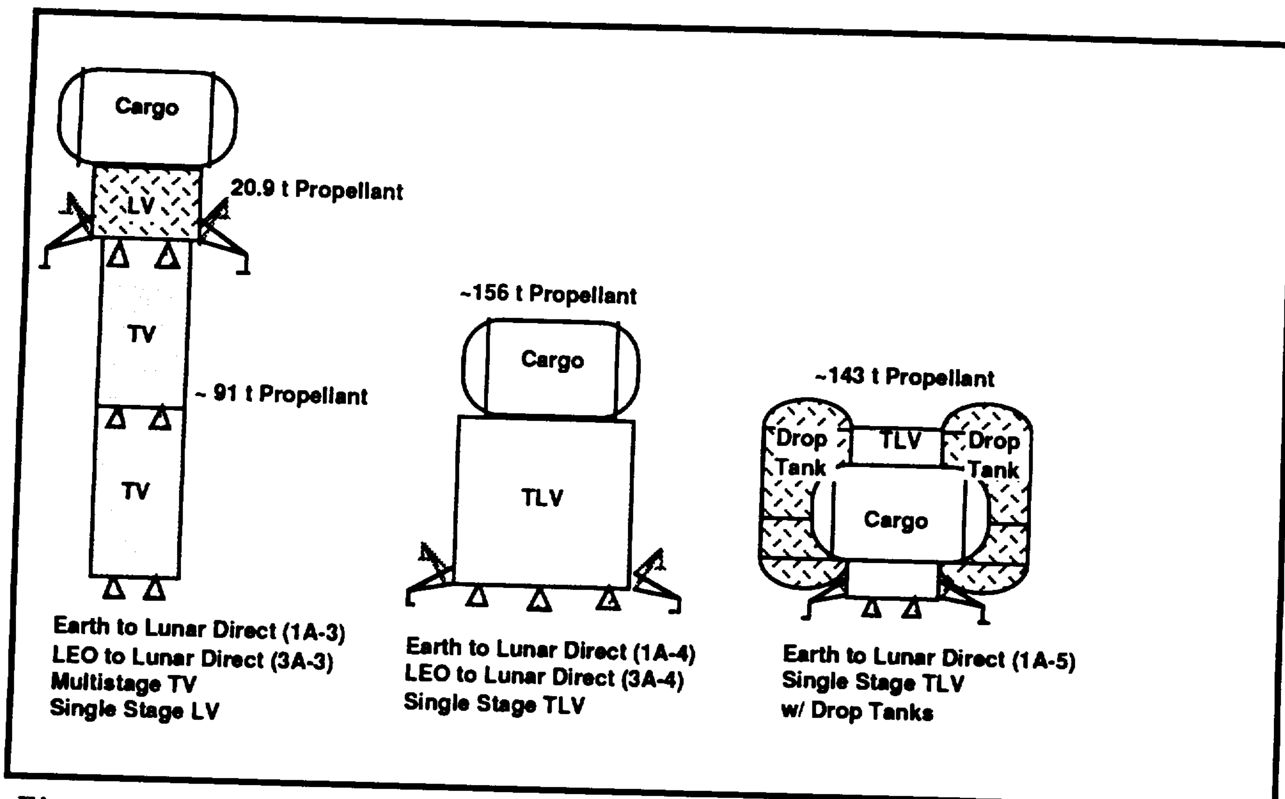
No. of Concept Elements	2
No of Rendez/ Dockings	0
No. Engine Burns	2
No. of Crew Transfers	0
No. of Cargo Transfers	0
Propellant Transfer	0
No. of Aerobrakes	0
No. of Propulsion Systems	2
No. of Element/Tank Separations/Deployments	1
On Orbit Assembly / Mating	0
Total Ops Complexity	7

No Return from Lunar Surface
Separate Transfer Vehicle and Separate Landing Vehicle
Single Stage TV & LV

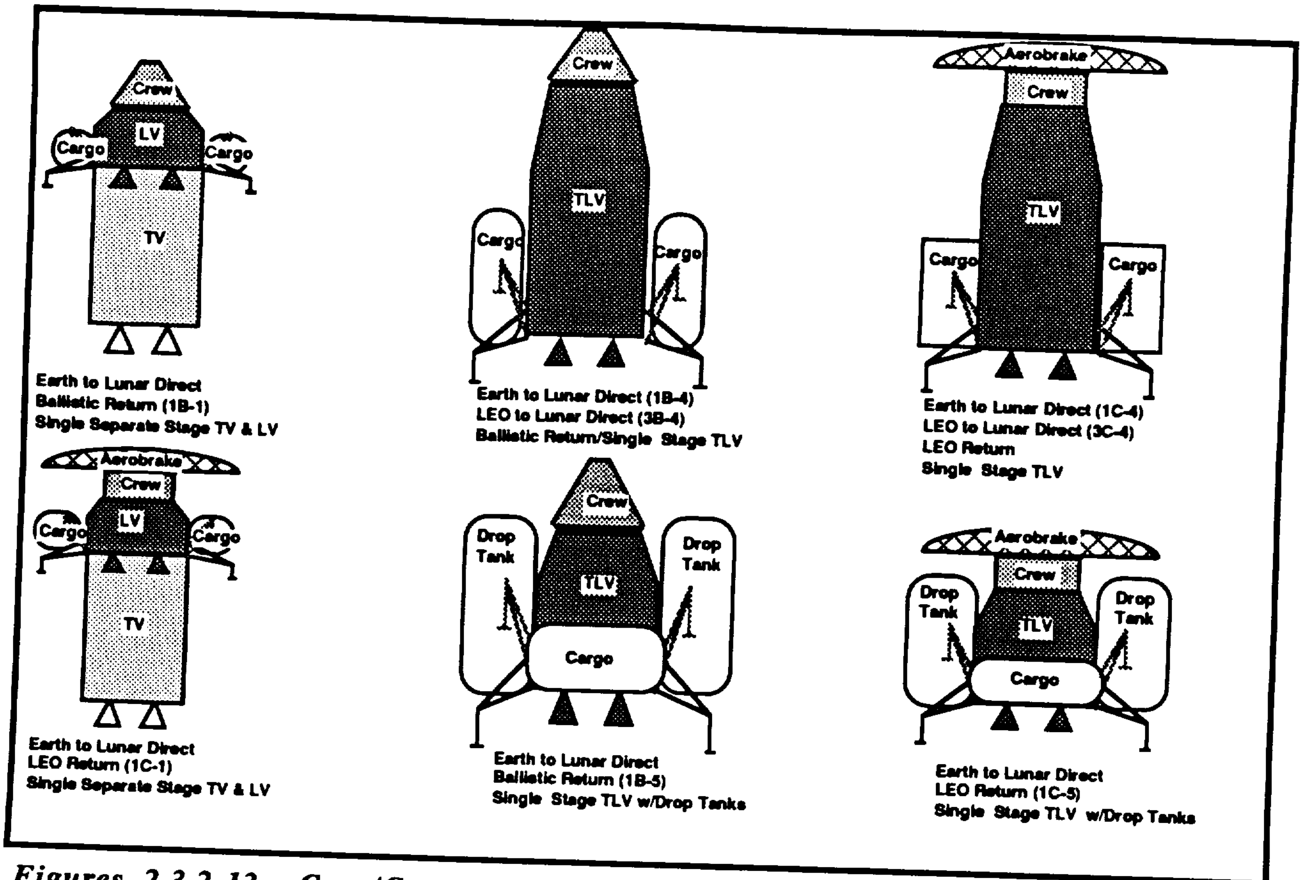
Figure 2.3.2-11 Typical Evaluation Sheet in Figure 2.3.2-12, and thirteen piloted configurations, shown in Figures 2.3.2-13, 2.3.2-14, and 2.3.2-15, were selected to be carried forward for additional study.

A preliminary screening was performed of concepts recommended from the first downselect, some new concepts, and some concepts added back from the initial downselect. Twelve concepts— five cargo only and seven crew/cargo concepts went through detailed concept definition during the second downselect phase. These concepts were evaluated against selection criteria - cost, operations, mission adaptability, and risk. Five criteria driven concepts — two cargo and three crew concepts — were recommended for additional study during the final selection process.

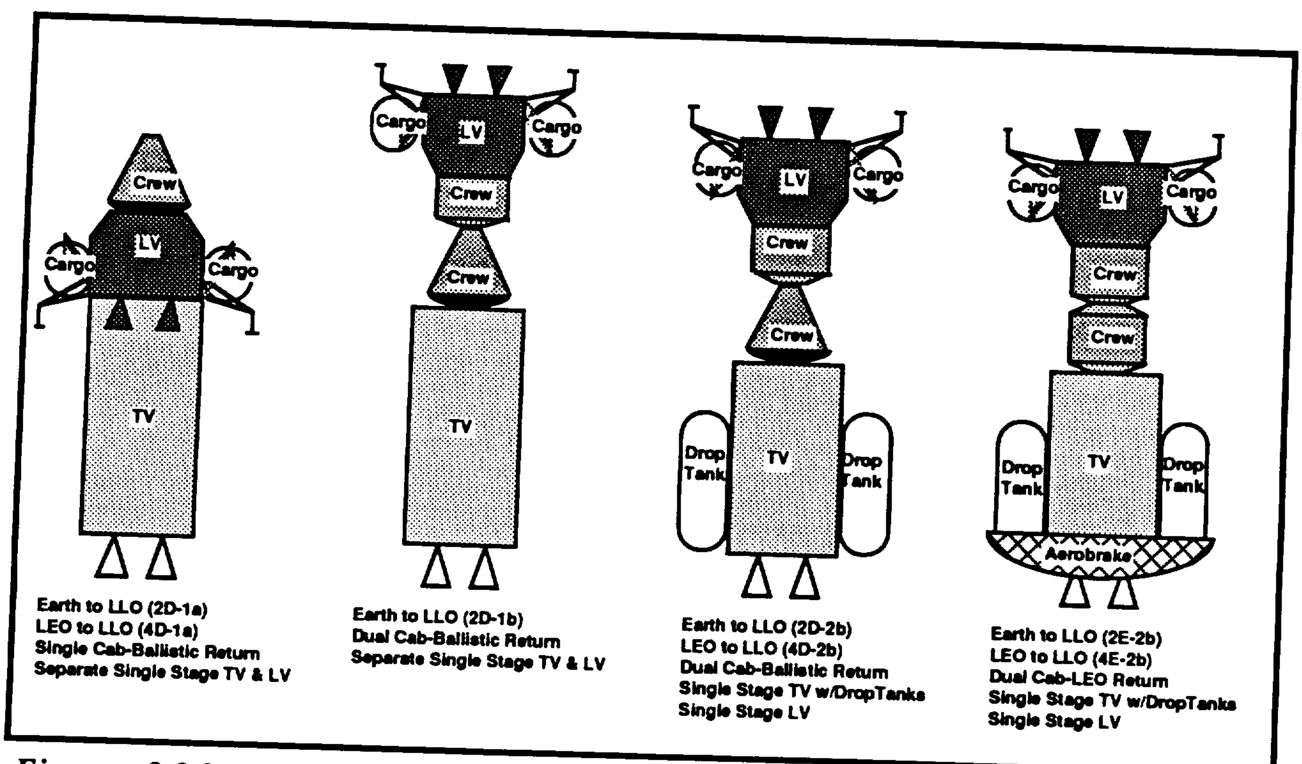
The first step of this phase, screened each configuration through the lunar architectures shown in Figure 2.3.2-16 against top level criteria such as LEO requirements and operations, technical risk, cost drivers, ground operations, etc.



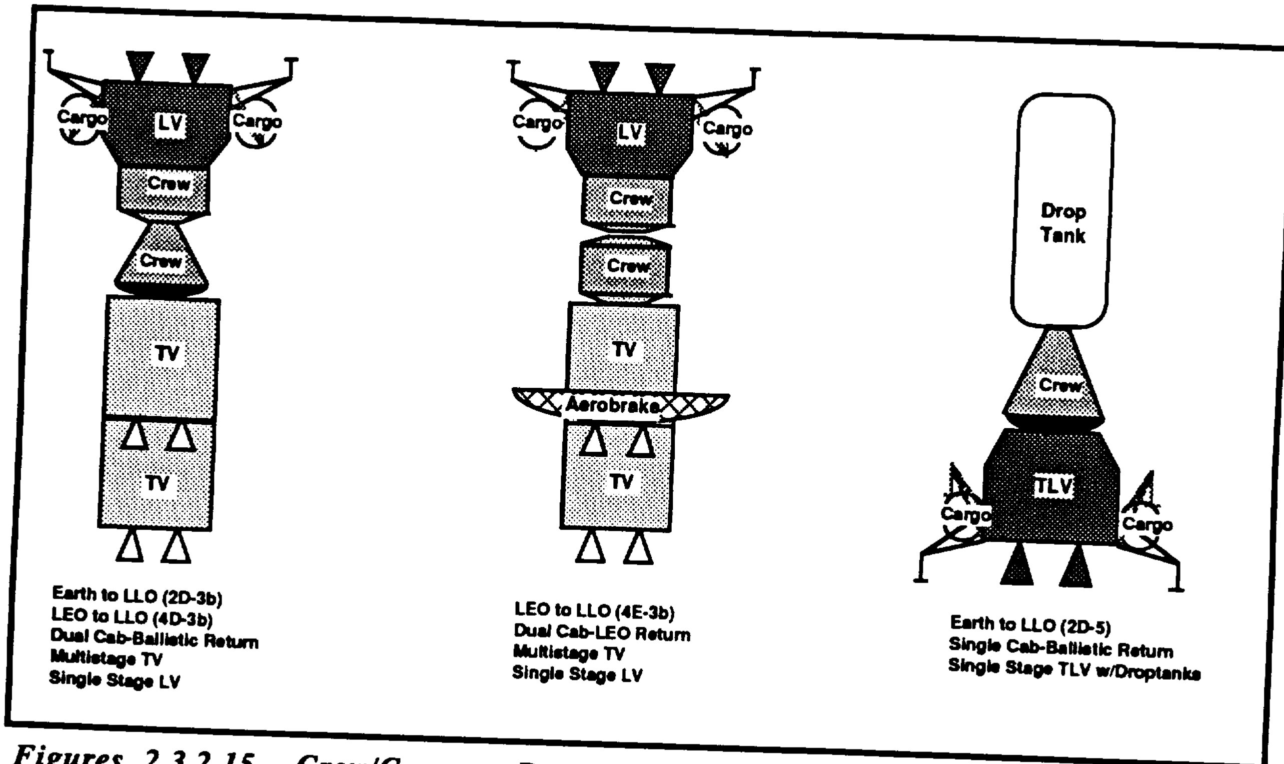
Figures 2.3.2-12 Cargo Only - Recommended Concepts



Figures 2.3.2-13 Crew/Cargo - Recommended Concepts



Figures 2.3.2.14 Crew/Cargo - Recommended Concepts



Figures 2.3.2.15 Crew/Cargo - Recommended Concepts

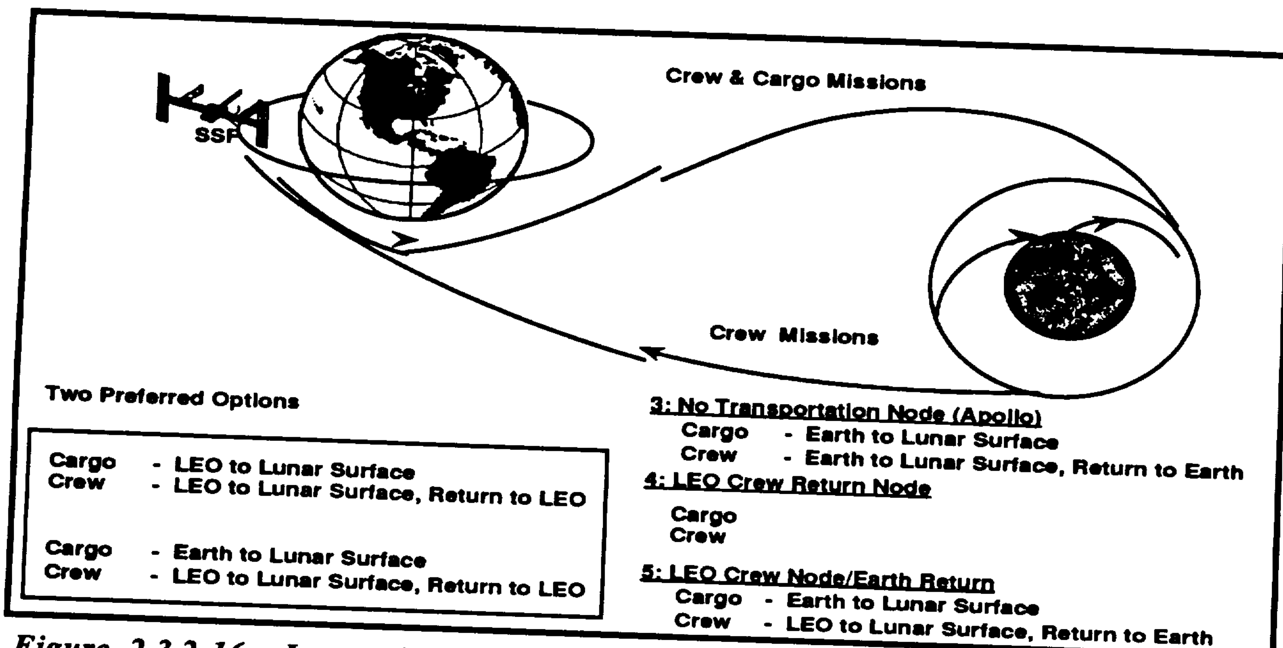


Figure 2.3.2-16 Lunar Architecture Assessment

As a result of this screening effort five cargo concepts, (1A-1, 1A-3, 3A-2, 3A-3, and 3A-5), shown in Figure 2.3.2-17, were retained for additional study and definition. Five crew concepts, (4E-2A, 4E-2B, 4E-3A, 4E-3B, and 4E-5B), shown in Figure 2.3.2.18 were retained after the preliminary screening for lunar architectures options 1 & 2. Two crew concepts, (2D-1A and 2D-

3A), shown in Figure 2.3.2.19, were retained after the preliminary screening for lunar architecture option 3.

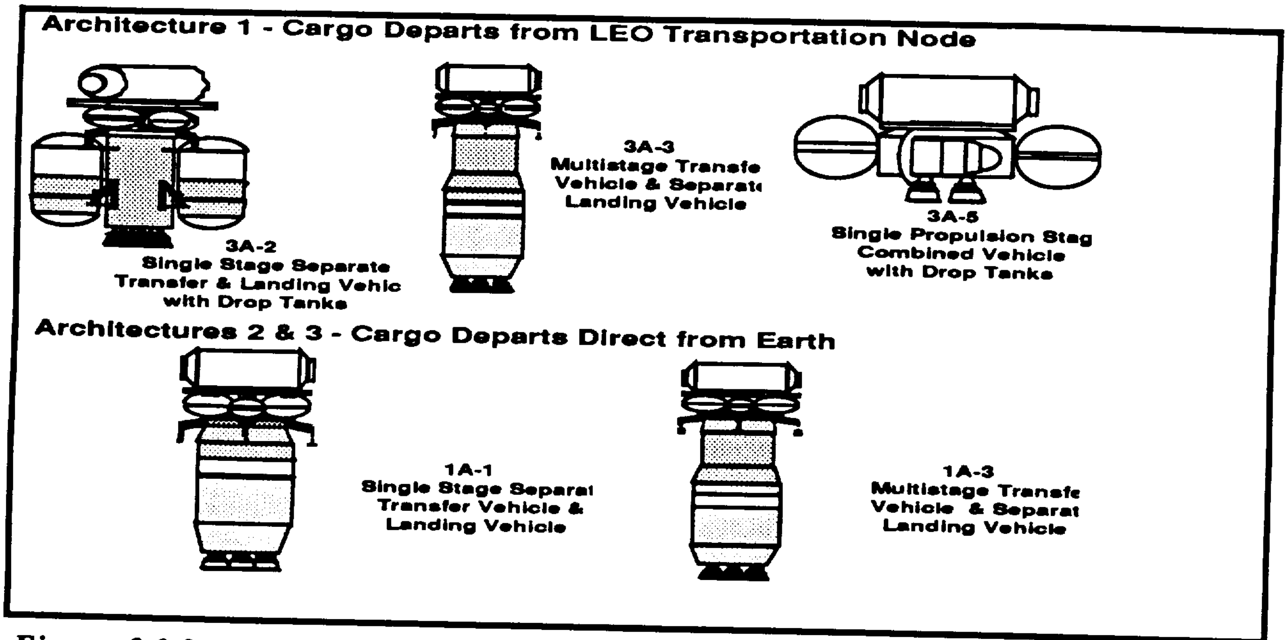


Figure 2.3.2-17 Cargo Concepts Retained for Additional Study

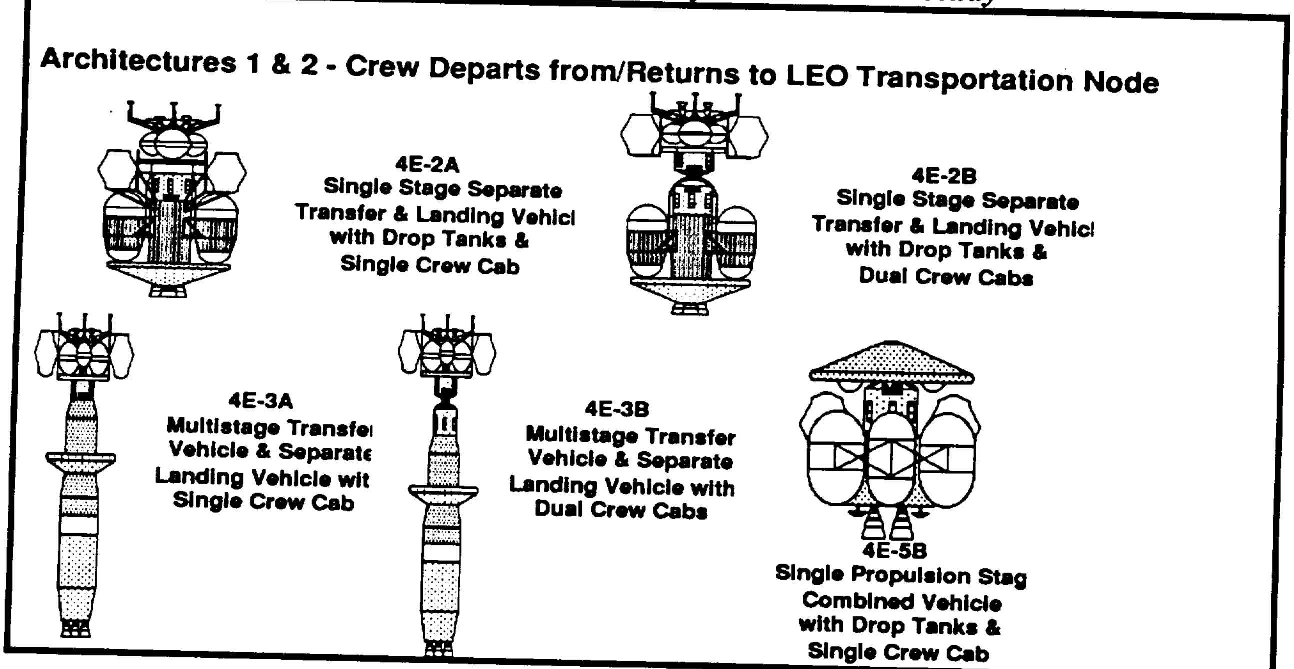


Figure 2.3.2-18 Piloted Concepts Retained for Additional Study

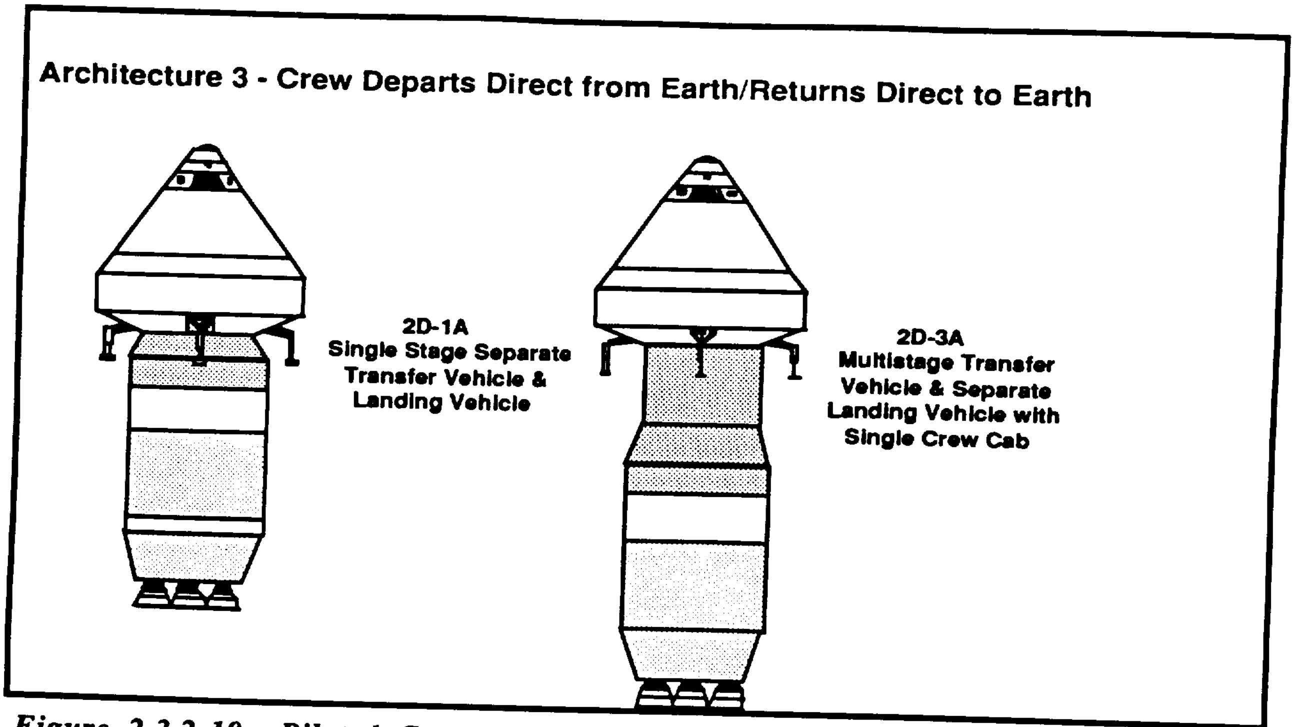


Figure 2.3.2-19 Piloted Concepts Retained for Additional Study

These five cargo concepts and seven crew concepts were then subjected to more detailed concept definition. Top level missions scenarios (outbound and inbound legs) were generated for each concept. An assessment of critical mission operations during each mission phase were evaluated for criticality 1 operations (loss of crew or loss of mission critical hardware) and criticality 2 operations (loss of mission - crew returns safely and cargo can be salvaged). Detailed configuration definitions for each concept were developed that included preliminary sizing, dimensions, and mass properties. In addition, manifest layouts were generated for each concept to show typical flight manifesting in heavy lift launch vehicles. The ability of each concept to adapt to other design reference missions was assessed by addressing vehicle element interchangeability and performance capability to perform other missions. Operational timelines were generated for each concept to determine workshifts required at Space Station Freedom for the initial vehicle assembly and steady state refurbishment operations. New ground operations facilities for each concept were also determined. Cost data generated for each concept was broken up into DDT&E, production, operations, and total life cycle costs by vehicle element. Figures 2.3.2-20 illustrates the typical detailed data generated for each concept (crew concept 4E-5B is shown as an example). Selection criteria and their associated weighting factors were then developed prior to conducting the detailed evaluation for each configuration. Four selection criteria were utilized in support of the second downselect process—program cost, operational complexity, mission adaptability, and risk. These criteria are defined as listed below:

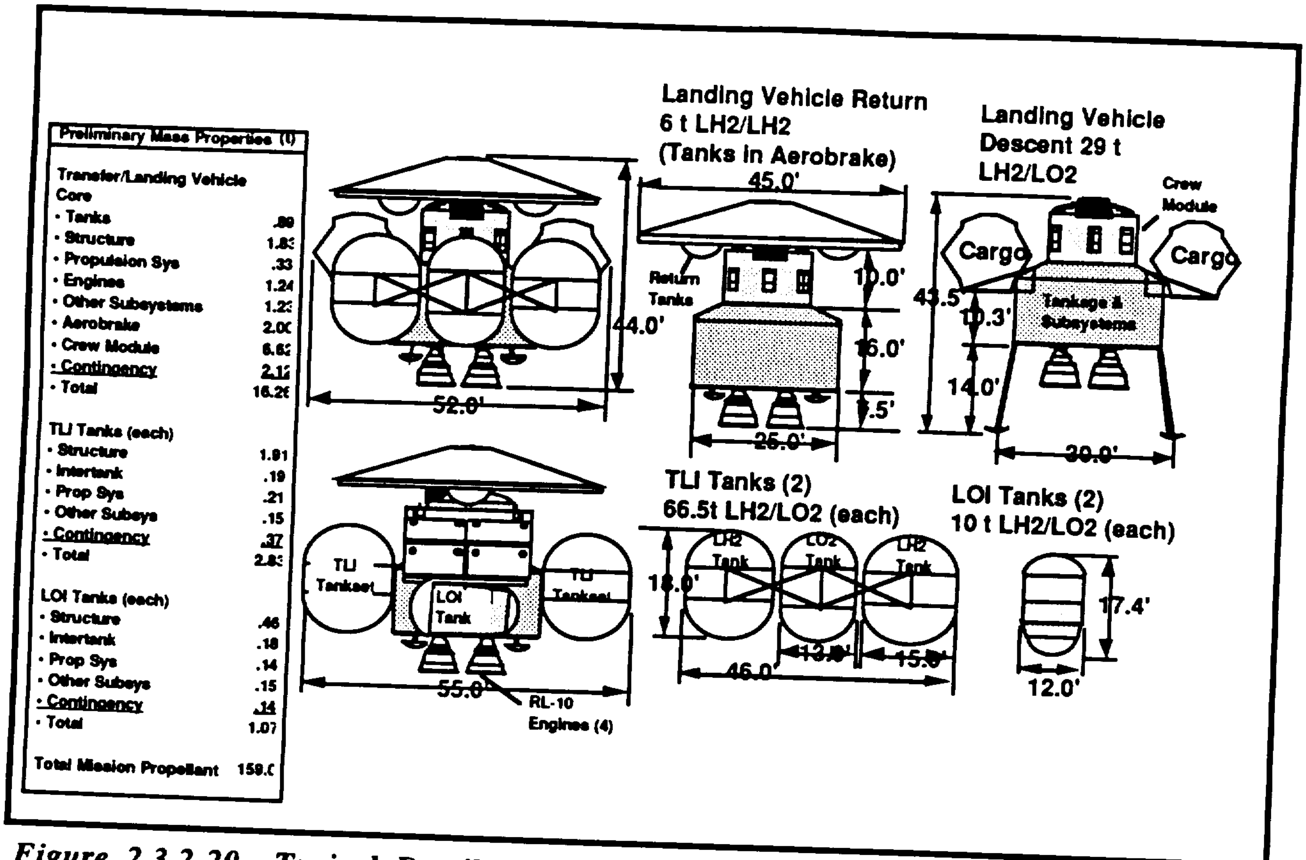


Figure 2.3.2-20 Typical Detail Data - Configuration Definition

- Program Cost: The total cost to acquire and own the system including full scale development, verification, production, operations, support, performance, and disposal.
- Operational Complexity: Addressed the number and complexity of the STV mission phases with the emphasis on safety and mission success.
- Mission Adaptability: Determined the capability of a configuration to capture all or some of the STV design reference missions either with existing elements or the reconfiguration of an element.
- Risk: The probability of not meeting a technical, schedule, or cost requirement and the effect on the program if the requirement was not met.

The data from the detailed concept definition was consolidated into four separate selection models—one for each criteria (one model emphasized cost as the primary driver, another emphasized operations, etc.). The evaluation values were then ranked in order of their value with the lowest value representing the best overall evaluation score. Selection of the final configurations were based on the best selection value from each criteria model.

The amount of influence that the results of the criteria/configuration evaluations had on the overall selection ranking of a configuration was determined by defining the weight that each criteria would carry during the selection analysis. These weight factors would be derived first as dictated by programs wants, and second by assigning a set value to a criteria and allowing the remaining criteria factors to shift according to program wants. A quality function deployment (QFD) analysis was used to develop both the derived set of weighting factors as well as the fixed values shown below:

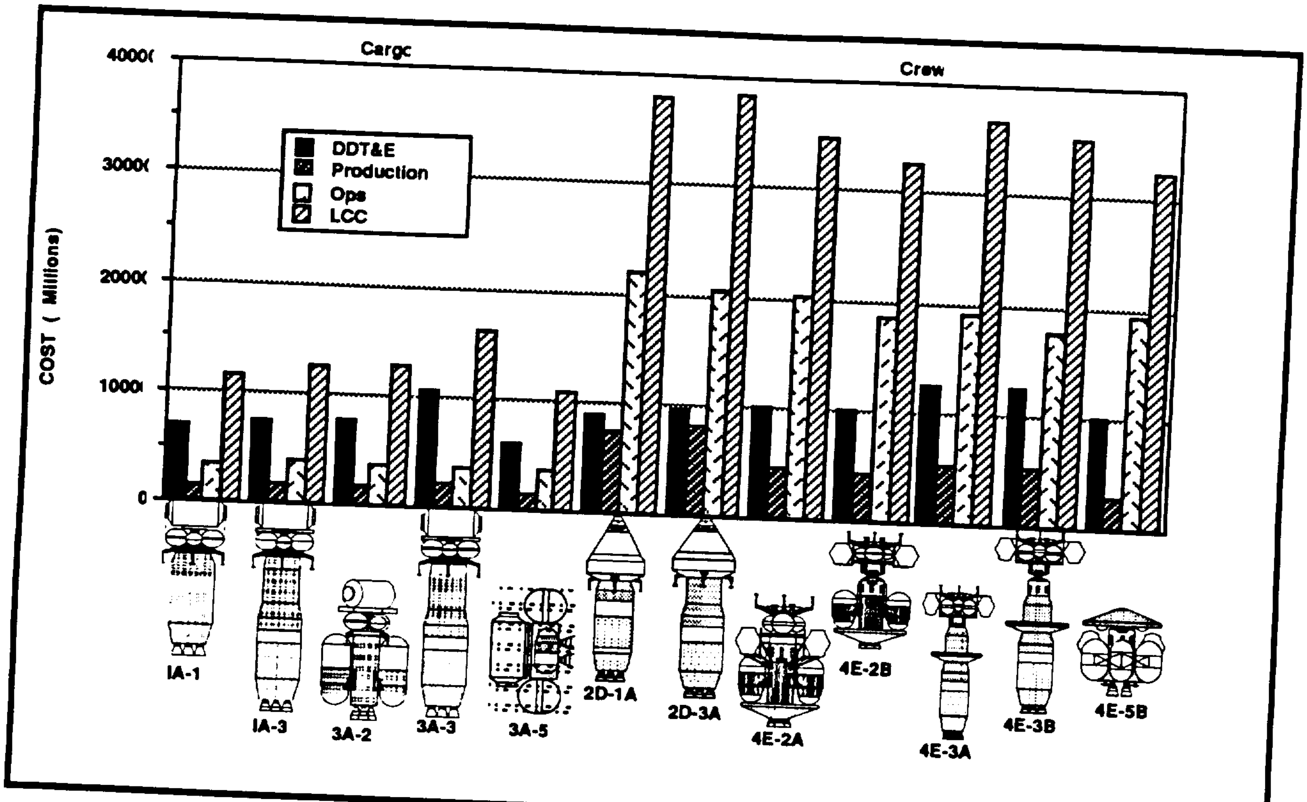
Derived:	Cost = 50%, Ops = 30%, Mission Adapt = 2%, Risk = 18%
Fixed:	Ops = 50%, Cost = 25%, Mission Adapt = 5%, Risk = 20%
Fixed:	Risk = 50%, Cost = 20%, Ops = 25%, Mission Adapt = 5%
Fixed:	Mission Adapt = 50%, Cost = 15%, Ops = 20%, Risk = 15%

Following completion of this analysis, a review of the NASA criteria and their associated weighting factors showed a very close correlation.

The results of the detailed evaluation effort provided an extensive database from which the final recommendation could be made. With this database was a summary the mass properties for the seven crew concepts and five cargo concepts evaluated during the second downselect process. A summary of the cost data for all twelve concepts is shown in Figure 2.3.2-21.

Ground processing operations analyses were based on the quantity of facility modifications and additions required to support the STV configuration as summarized as well as LEO node operations analyses as summarized, and a summary of the risk evaluation analysis which was based on a qualitative assessment of the probability of not meeting a technical, schedule, or cost requirement and the overall program effect of not meeting that requirement.

Using the quantitative values produced from the criteria-based selection models, each of the configurations were ranked in order of lowest selection value to highest (lowest being the best). For the piloted configurations, this produced a ranking from one to seven and in the cargo configurations, a ranking of one to five. This was done for each of the four selection criteria, producing the relative selection ranking chart as illustrated in Figure 2.3.2-22. Based on the results of the second downselect process, five vehicle configurations were recommended for additional



Figures 2.3.2-21 Cost Evaluation Summary

	Cargo Evaluation Rankil					Crew Evaluation Rankir						
Cost Driver	2	5	3	4	1	6	5	7	3	4	2	1
Operations Driven	2	5	3	4	1	6	4	7	3	5	2	1
Adaptability Driven	2	5	3	4	1	4	5	6	7	3	2	1
Risk Driven	2	5	3	4	1	6	5	7	3	4	2	1

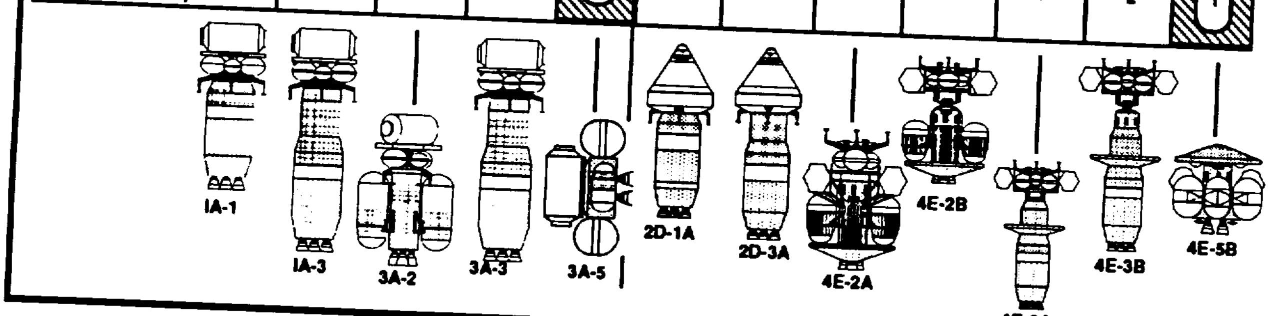


Figure 2.3.2-22 Configuration Selection Evaluation Summary

study during the final downselect process. These five configurations (two cargo and three piloted/cargo configurations) are shown in Figure 2.3.2-23.

The final phase of the concept selection trade started with determining the feasibility of combining the piloted and cargo versions of the configurations recommended from the second downselect into common vehicles. Following the commonality evaluation, a final configuration analysis was performed to select the final recommended configuration. This final evaluation was based on an operational contingency analysis and a detailed cost/operations analysis. After the selection of the recommended STV for the lunar transportation mission, a configuration based reusability trade was conducted.

The first phase of the final downselect process was to determine the feasibility of combining the piloted and cargo versions of a configuration into common vehicle families. The five configurations (2 cargo and 3 piloted) recommended for additional study from the second downselect were evaluated to determine commonality between the vehicle elements.

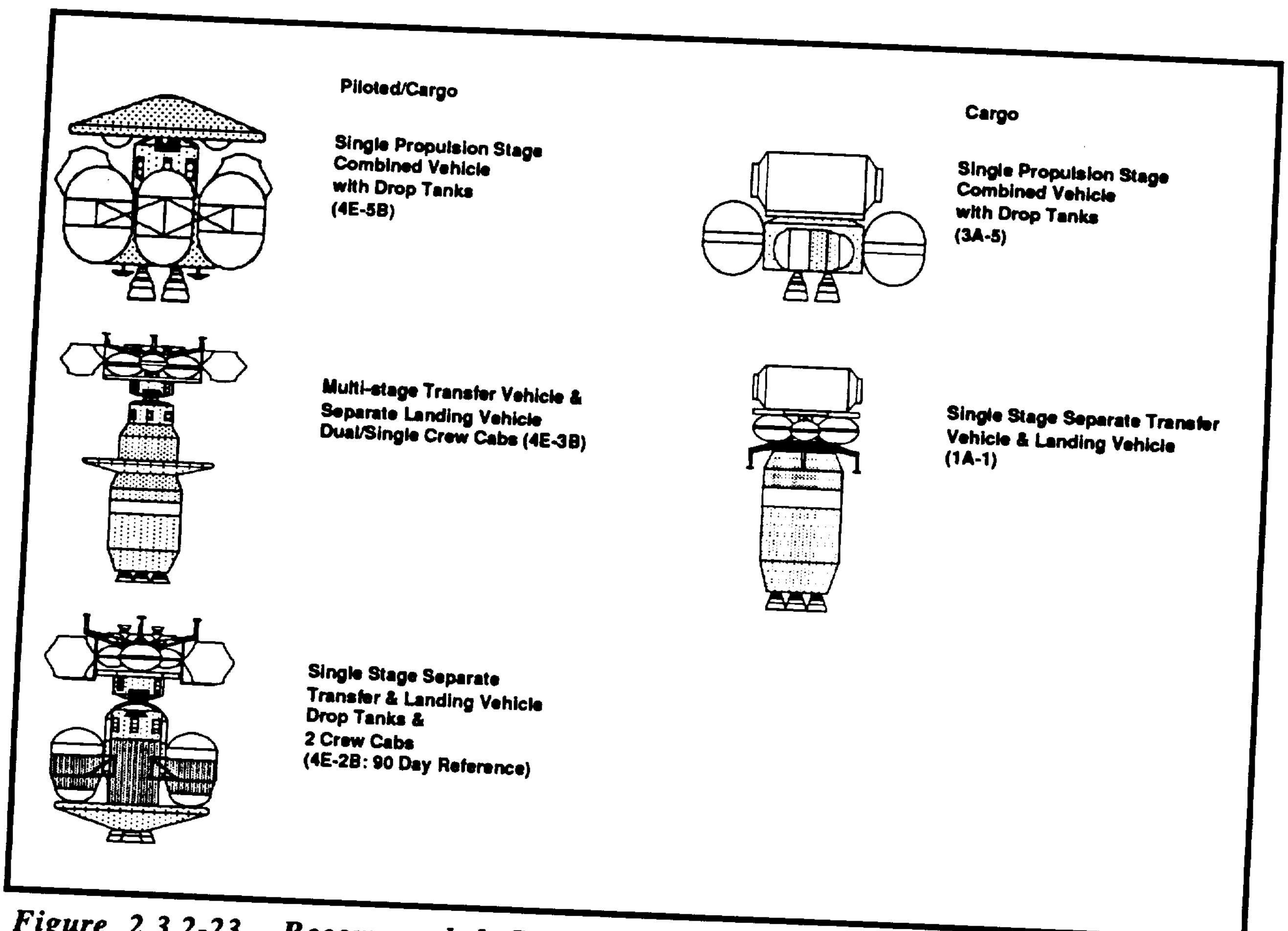


Figure 2.3.2-23 Recommended Configurations

The thrust of this assessment was to breakdown each cargo and piloted configuration into similar components and evaluate the commonality between them. The results recommended that the piloted and cargo concepts from the second downselect could be combined to form three common families of vehicles. Figure 2.3.2-24 illustrates the three common families and their required propellant quantities.

The next phase of the final downselect process was to conduct an operational contingency analysis. This analysis addressed each lunar mission phase, determined possible contingencies for system failures, and provided a recommendation on which of the configurations tended to have the fewest mission anomalies. Results of the contingency analysis showed no clear discriminators between the candidates. Since each of the configurations has advantages and disadvantages, there was no configuration that stood out as being better than the others.

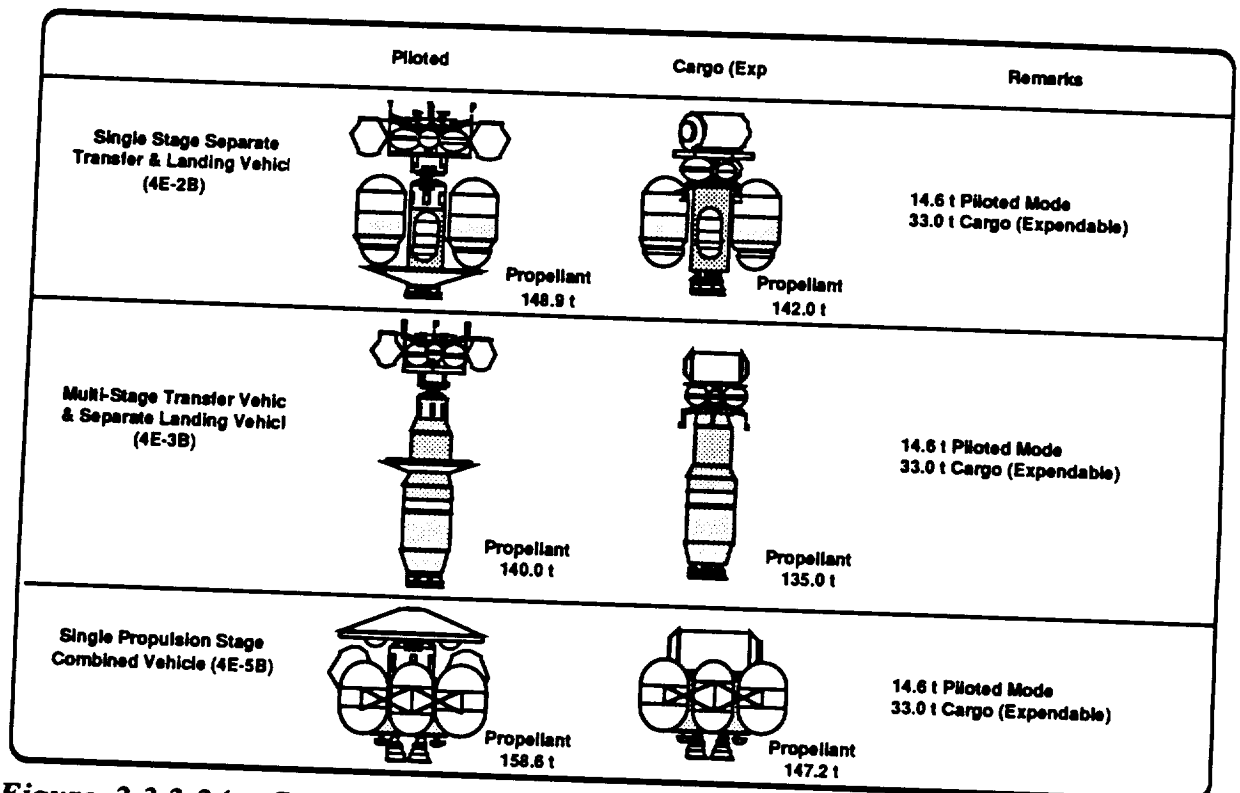
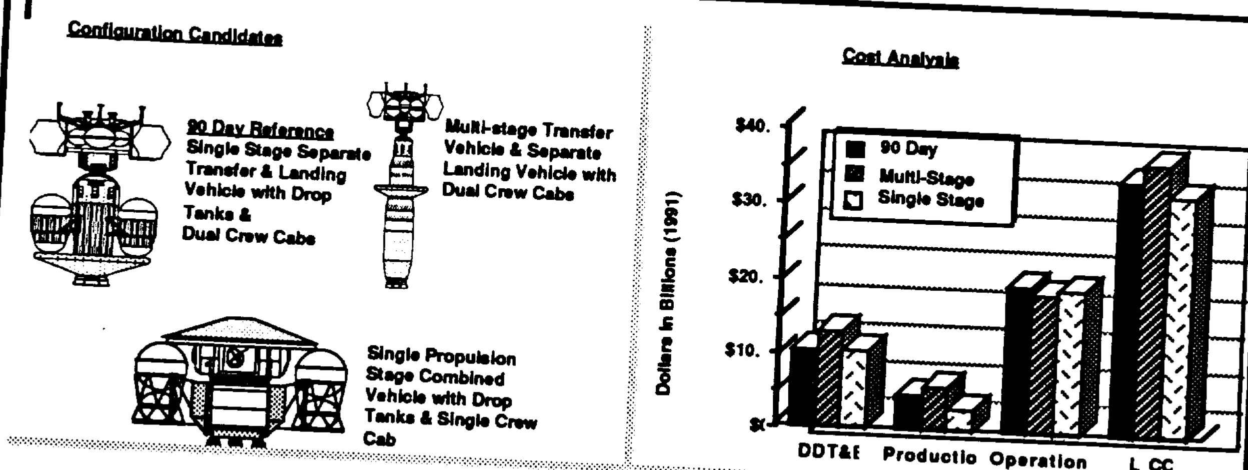


Figure 2.3.2-24 Common Families

The last phase of the final downselect process was to perform a detailed analysis of system costs and operations. The cost evaluation was based on DDT&E, production, operations, and life cycle costs. As shown in Figure 2.3.2-25, the single propulsion family (4E-5B) had the lowest life cycle costs, while also exhibited the lowest number of shifts required for initial flight assembly and checkout at Space Station Freedom. When analyzing the cost and operations data for each configuration, the weighting factors that were developed during the preliminary configuration analysis, were incorporated. Based on the weighted values determined during the study, the single propulsion system family was the clear winner.

After the final configuration selection was complete, a configuration reusability trade was conducted. The configuration reusability trade addressed the feasibility of reusing vehicles for the cargo missions. Performance data defined a cargo capacity range of 37.4 t for expendable missions, to 25.9 t for a reusable cargo mission, to 14.6 t for a piloted mission. Because the 25.9 t does not comply with the 33.0 t cargo requirement, an evaluation of the actual payload support systems manifested cargo indicated that the 25.9 t capability is within the noise range of the actual mass requirements of 26.46 t.



Operations Analysis

		90-Da	Single Stag	Multi-Stag
Ground	Processing Facilities Mods And/Or Additions	9	9	9
LEO	Initial Flight A&CO Complexity Steady State Flight A&CO Complexity	125 Shifts 150 Shifts	100 Shifts 160 Shifts	115 Shifts 150 Shifts
Mission Failure Modes	Crit 1 Crit 2	15 16	10 12	17 12

Figure 2.3.2-25 Life Cycle Cost/Operations Data

Based on this, the recommendation to reuse the cargo vehicles based on performance is a valid one. The final piece of data that was required to complete the reusability study was the economic impact of reusing the cargo vehicle. With the reuse of one of the four cargo only vehicles that are currently manifested in Option 5, the total lunar transfer system vehicle requirement is reduced from nine to eight. The cost saving associated with these reduction in a vehicle is $0.8 billion. By reallocating, to one or more piloted missions, a small portion of cargo, the two remaining cargo missions can be reused. With all three cargo missions flown in the reusable configuration, the vehicle cost savings increases from $0.8 to $2.4 billion. Reusing the cargo vehicles also provides the means for a final systems checkout prior to committing a crew to lunar launch.

Figure 2.3.2-26 illustrates the configuration selected as a result of the final downselect process. The Single Propulsion System Family represents the best STV configuration that supports the Lunar design reference missions. Key attributes of this family include:

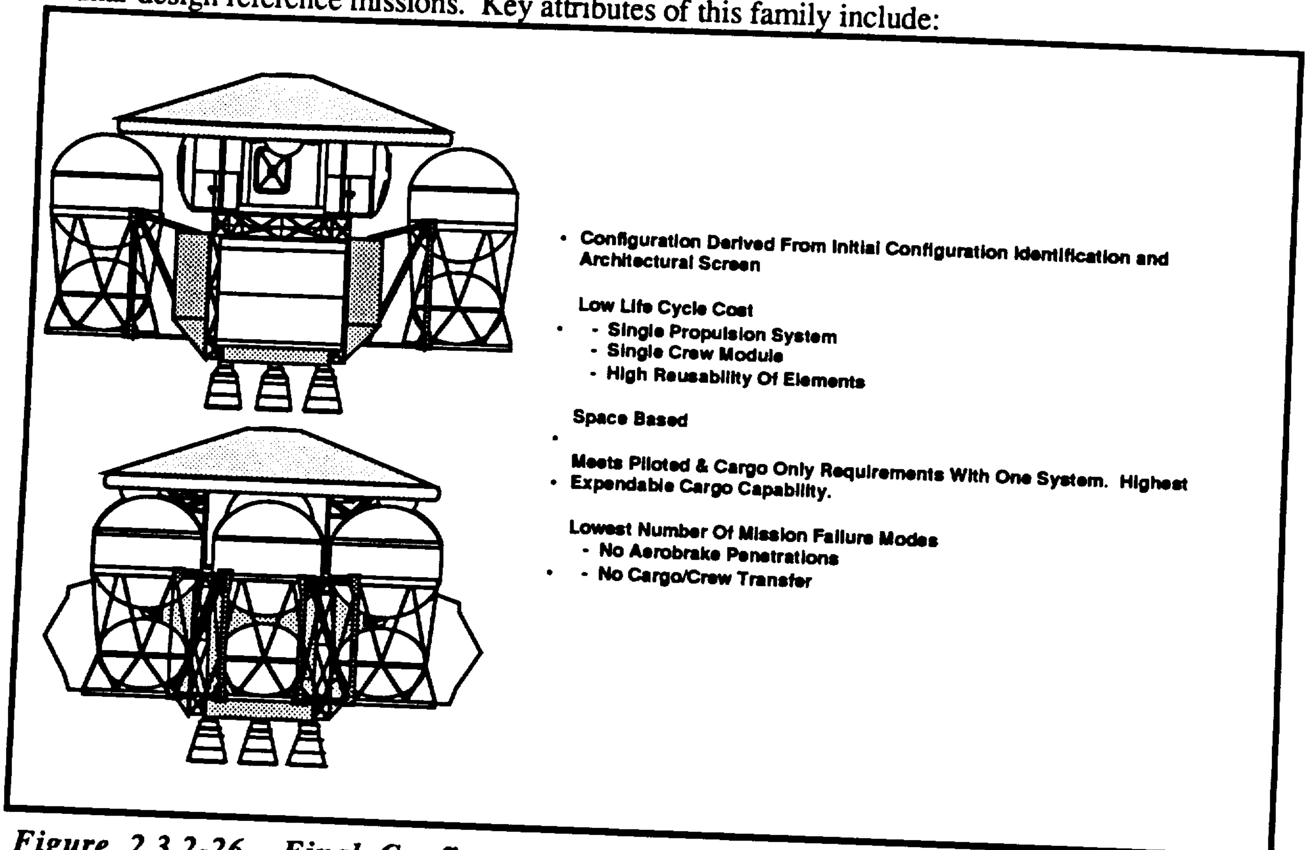


Figure 2.3.2-26 Final Configuration Recommendation

- Lowest LCC
- Lowest number of critical operational failure modes
- Meets all piloted and cargo only requirements, while featuring the highest cargo expendable capabilities.

Two addition system level trades were addressed during the STV study, LLOX Utilization and SSF Sensitivities. This resulted in a recommendation that the LLOX trade study be suspended until two key pieces of data are firmly in hand. One is the cost per kilogram of launching mass to low Earth orbit using an HLLV; and the second is the production cost of a pilot LLOX plant operating on the lunar surface. LLOX is a second generation surface activity and, therefore, should not be addressed until the first generation is implemented, or at least well underway. Key inputs into whether LLOX would be profitable are the cost of goods in LEO and the cost of LOX production on the moon. Trade studies at this point in time can assume many factors biasing the results to support a desired position. It is essential that actual data be inserted into the equation before investing billions of dollars in second generation activities on the lunar surface.

The other addressed the sensitivities that included impacts to SSF Guidance, Navigation & Control, Impact on Micro g Users, Impact on Reboost Logistics, Enclosure Size & Location, of SSF to supporting the LTS/STV.

The impacts to SSF Guidance, Navigation & Control analysis assumed that a high-mass LTS is supported in a 15.3 x15.3 m servicing enclosure positioned on a lower keel of the Space Station. This configuration, derived from the November 1989 NASA 90-day study on Human Exploration, recommended the addition of a lower keel to support lunar operations. Space Station Freedom flies at Torque Equilibrium Attitude (TEA), where aerodynamic and gravity gradient torques cancel. Current analysis indicates that the TEA of the Assembly Complete Station has a large negative pitch angle and will not meet the requirement to fly within +/- 5 degrees of Local Vertical, Local Horizontal (LVLH). The addition of a lower keel will significantly improve the pitch attitude. As the mass of the LTS is increased, pitch and yaw attitudes are further reduced toward LVLH. Roll TEA attitude increases with additional LTS mass, but over the range of potential LTS mass to be supported, Station TEA will remain within the +/- 5 degree requirement.

Analysis indicated the addition of the LTS mass on a lower keel has a severe impact to SSF microgravity environment. Even with an empty servicing enclosure, Station cg would be below the desired centerline for the laboratory modules.

Reboost propellant required during a low solar cycle year is shown as a function of LTS mass. The addition of the lower keel and servicing enclosure increases Station propellant use by about 5000 lb Hydrazine. After this initial increase, the entire range of LTS mass will not require more than one additional propulsion module (8000 lb Hydrazine) for the low solar cycle year.

The size to which an LTS could grow within the constraints of the Space Station system is governed by limits applied to the size of its enclosure. The two dimensional constraints are in the

Y (or latitudinal) dimension and the Z (or radial) dimension of the Station configuration. The LTS enclosure is assumed to be placed in a location bounded by a "lower keel", or two downward pointing extensions of the truss structure connected by a cross boom. The boom dimensions are governed by the physical space available on the main truss structure as well as constraints in station controllability which govern the extent to which the truss can grow downward.

Space Station Sensitivities—The sensitivities identified and addressed between the space station and the LTS, consisted of mass, size, propellant management, and LTS handling.

Mass impacts were assessed for Guidance, Navigation & Control, Mirco-g, and Reboost. Analysis of the guidance, navigation & control functions for SSF showed that as the mass of the LTS is increased, pitch and yaw attitudes are further reduced toward LVLH, Figure 2.3.2-27. Roll TEA attitude increases with additional LTS mass, but over the range of potential LTS mass to be supported, Station TEA will remain within the +/- 5 degree requirement.

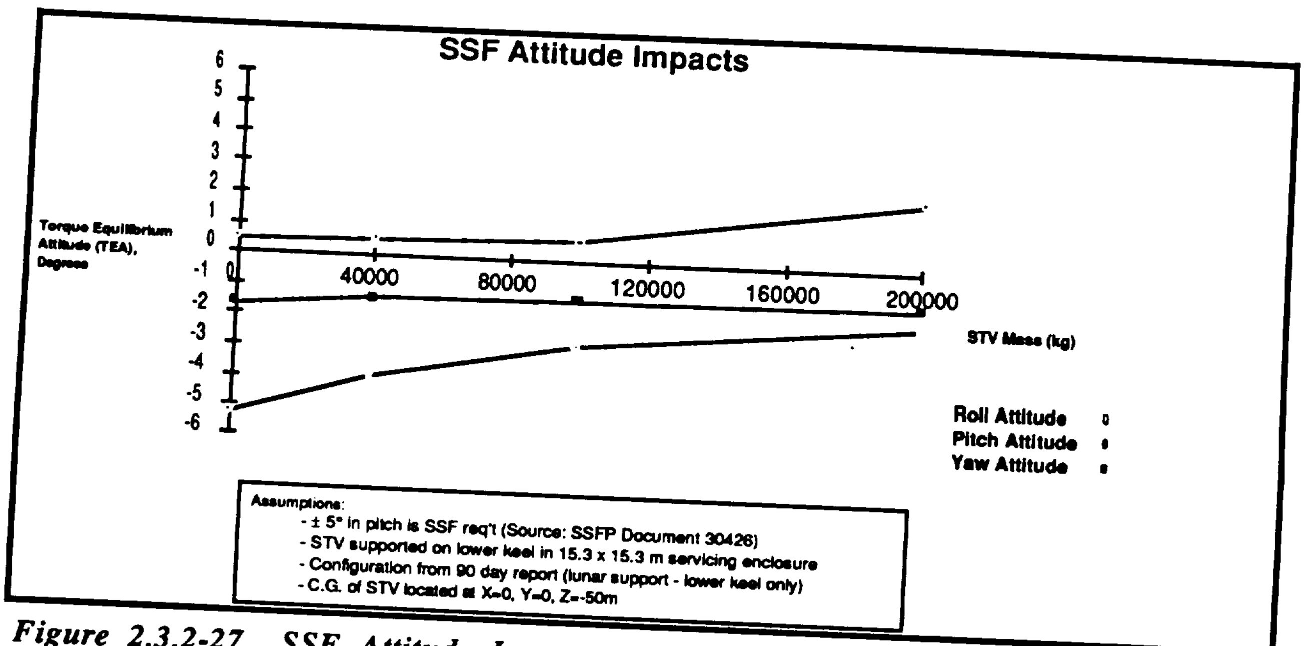


Figure 2.3.2-27 SSF Attitude Impacts

Part of this same analysis determined that the required momentum storage capacity is a function of many variables, including specific configuration and momentum management scheme during flight. Analysis using a momentum-management simulation indicates that increased LTS mass will have low impact on Station control, as shown in Figure 2.3.2-28

Micro-g analysis showed that the addition of the LTS mass on a lower keel had a severe impact on the SSF microgravity environment, including those periods when the LTS is gone and the servicing enclosure is empty. A Level II directive (BB000610A) was recently issued, changed the previous requirement of 10 micro-g in the laboratory modules to a quasi-steady acceleration levels not to exceed 1 mg for at least 50% of the user accommodation locations in each of the pressurized laboratories (US Lab, ESA and JEM PM at AC)". As shown in the plot of % total laboratory volume within 1 and 10 microgravity levels Figure 2.3.2-29, any appreciable mass LTS supported on a lower keel will not be able to meet this directive.

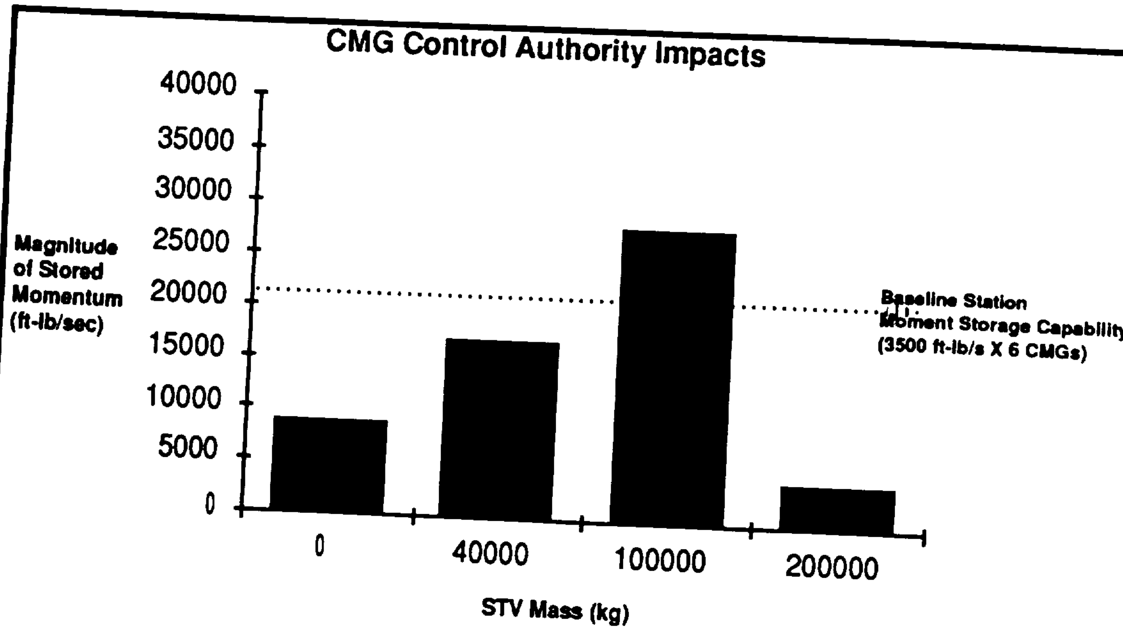


Figure 2.3.2-28 CMG Control Authority Impacts

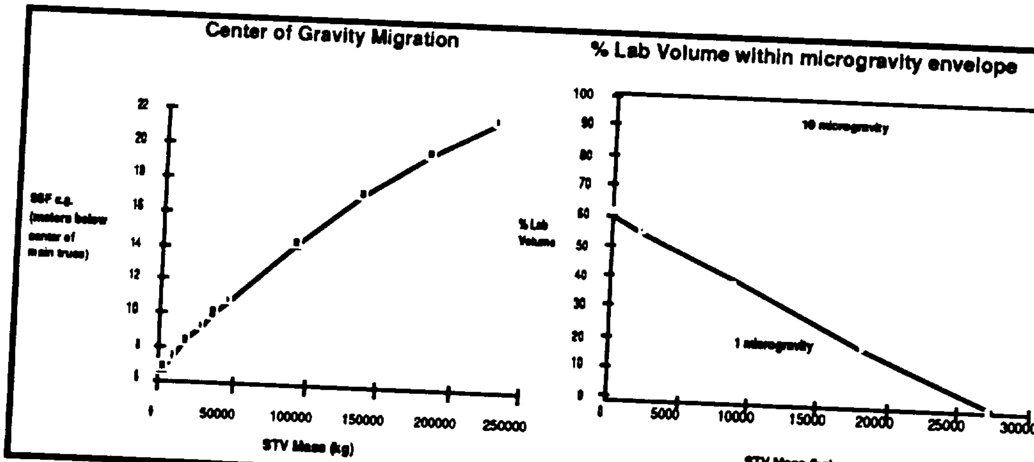


Figure 2.3.2-29 SSF Microgravity Environment Sensitivity

Reboost propellant required during a low solar cycle year was found to be a function of LTS mass. The addition of the lower keel and servicing enclosure increases Station propellant use by about 5000 pounds of Hydrazine, but does not require more than one additional propulsion module (8000 pounds of Hydrazine) for the low solar cycle year. Yearly required reboost Hydrazine is shown in Figure 2.3.2-30, for both low and high solar cycle years over the range of LTS mass on a lower keel.

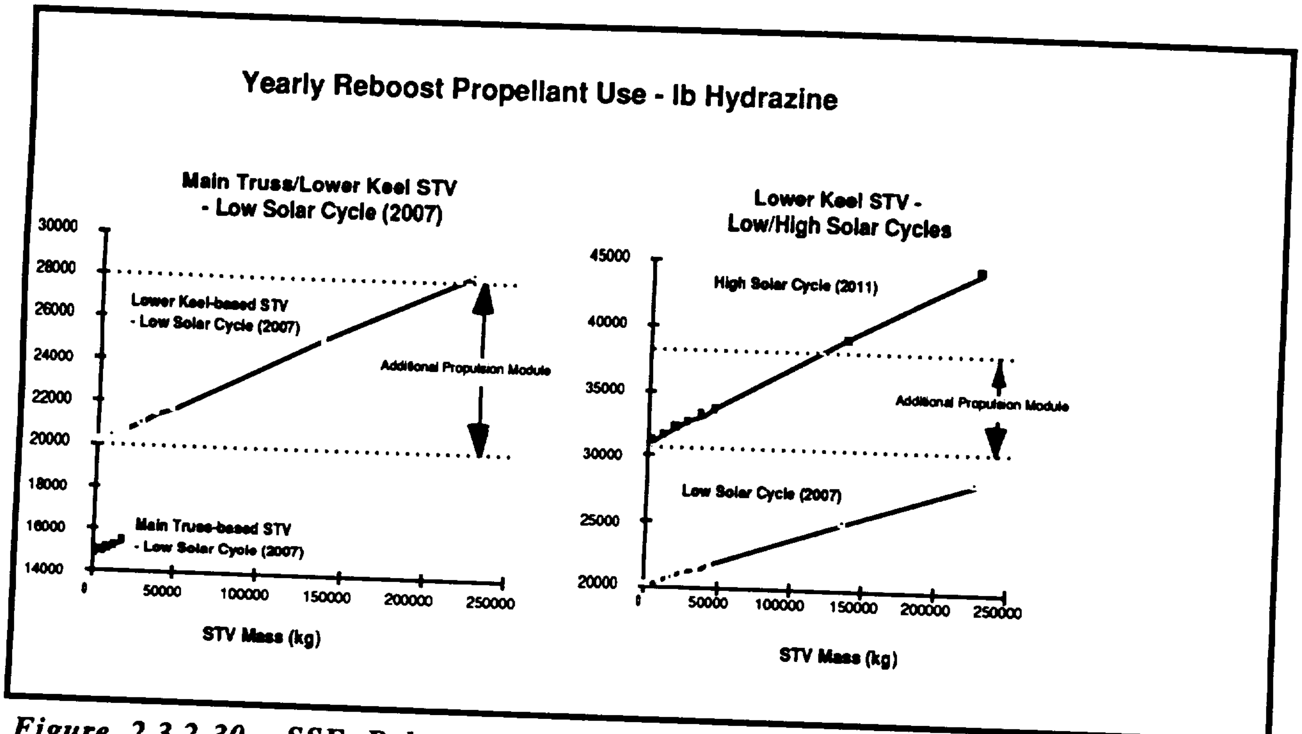


Figure 2.3.2-30 SSF Reboost Logistics

The size to which an LTS could grow within the constraints of the Space Station system is governed by limits applied to the size of its enclosure. The two dimensional constraints are in the Y (or latitudinal) dimension and the Z (or radial) dimension of the Station configuration. The LTS enclosure is assumed to be placed in a location bounded by a "lower keel", or two downward pointing extensions of the truss structure connected by a cross boom. The boom dimensions are governed by the physical space available on the main truss structure as well as constraints in station controllability which govern the extent to which the truss can grow downward. The maximum amount by which the enclosure can grow along the Y axis is 35 meters. Thus the maximum LTS diameter within the enclosure will be 31-33 meters, depending on safety factors. In the Z dimension, the limit, as shown, has two components. Forward of the lower keel truss structure plane, the maximum enclosure growth limit is 26.6 meters due to clearance requirements for LTS docking to the Space Station. Aft of the truss structure plane, the limit is relaxed to 43.8 meters, which is bounded by the envelope for a pressurized logistics module attached to a min-node. However, as the size of the LTS enclosure increases, there are also impacts to Space Station reboost logistics planning and the Station microgravity environment. As the frontal area of the enclosure grows, the drag coefficient increases, and extra propellant must be provided to the Space Station for altitude maintenance. As the enclosure size grows, added drag and mass cause the Station center of gravity (and microgravity ellipses) to move lower relative to the experiment module section. This movement, less than three meters from minimum to maximum enclosure size, can be considered a minimum impact.

One of the key concerns with LTS accommodations at Space Station Freedom is the storage of the LTS propellant tanks after they are received at SSF and prior to assembly with the LTS . As part of the propellant storage study, three options were identified as potential locations for the LTS propellant tanks (Figure 2.3.2-31): (1) Mount the propellant tanks within the SSF servicing enclosure, (2) Mount the propellant tanks on a tether away from SSF, Figure 2.3.2-32 shows the relationship between the length of the tether and the CG of SSF, (3) Mount the propellant tanks elsewhere on the lower keel outside of the servicing enclosure.

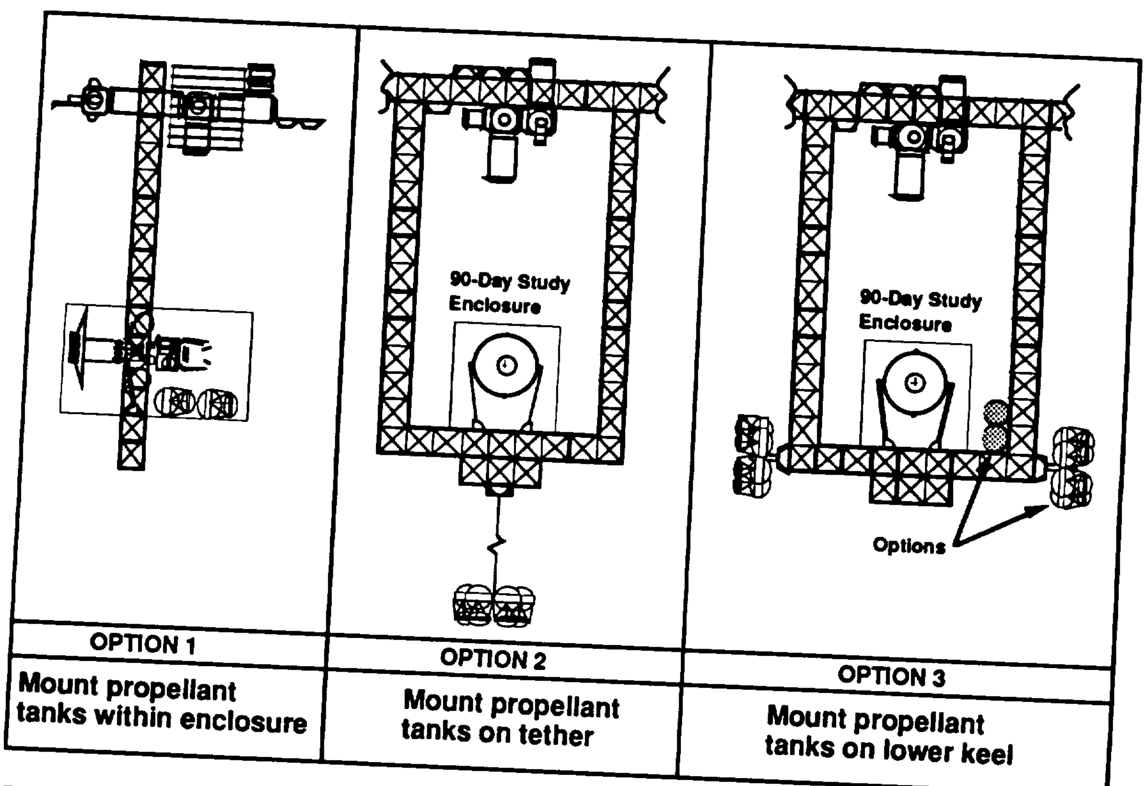


Figure 2.3.2-31 Propellant Storage Location Options

As depicted in Figure 2.3.2-33 the baseline Space Station has at least three mechanical systems that may be adapted to the LTS program. These devices are the mobile servicing system (mobile transporter and space station remote manipulator system), the unpressurized docking adapter, and the capture latches used for attaching the unpressurized logistics carriers and propulsion modules to the baseline space station integrated truss assembly. The unpressurized docking adapter may be modified to allow the LTS or portions of it to dock with the station. The capture latches, which are sized to accommodate either 3.00 inches or 3.25 inches STS payload trunnion pins, may be well suited for mounting LTS cargo elements to the truss structure prior to and during the assembly process. The mobile servicing center or some derivative of it is necessary for the performance of the LTS assembly functions. Although a number of SSF mechanical systems can be adapted for use in the LTS program, there are still several mechanical systems required for the LEO servicing facility that will be unique to the LTS program. These include an LTS core stage handling fixture, engine removal support hardware, LTS stack deployment device, and enclosure opening and closing mechanism. These devices will have to be more clearly defined so that their functions and operational complexity may be better determined.

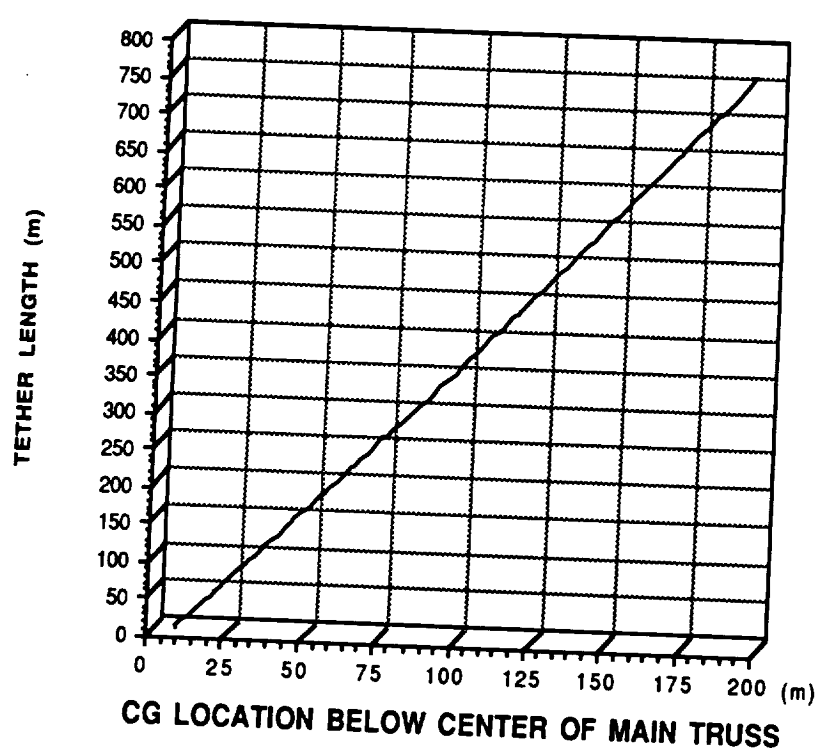
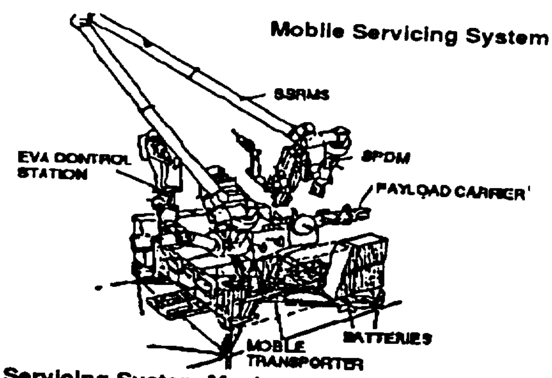


Figure 2.3.2-32 Tether Length Versus Station Center of Gravity



Mobile Servicing System Maximum Payload Capabilities

Station Configuration	Mass (kg)	Diameter (m)	Length (m)
Assembly & Baseline	20,900	4.5	17.0
Growth **	128,000	13.4	44.7

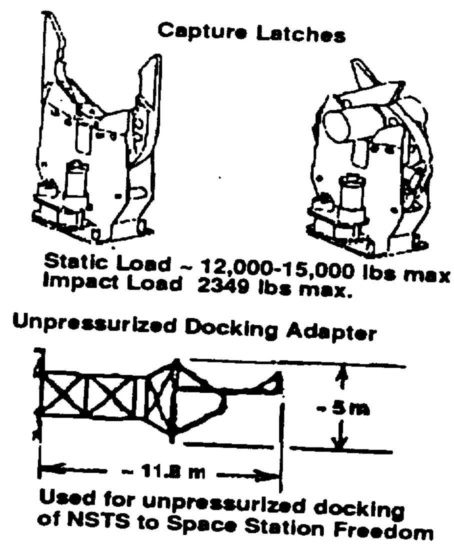


Figure 2.3.2-33 Space Station Mechanical Devices to Support STV Assembly

The baselined assembly complete space station provides a maximum of 75 kW from four photovoltaic power modules. This 75 kW of power is split between station housekeeping and station user payload power. With no surplus of power in the phase I SSF an LTS servicing facility will require additional modules to supply additional power. The current growth path for SSF utilizes pairs of Solar Dynamic (SD) power modules for 50 kW increments. Power to support the presence of an LTS and LEO servicing facility can be accommodated with only 37.5 kW additional for LTS powers up to 12 kW. This includes approximately 10 kW for the servicing facility and 10 kW if additional crew facilities are required (Figure 2.3.2-34).

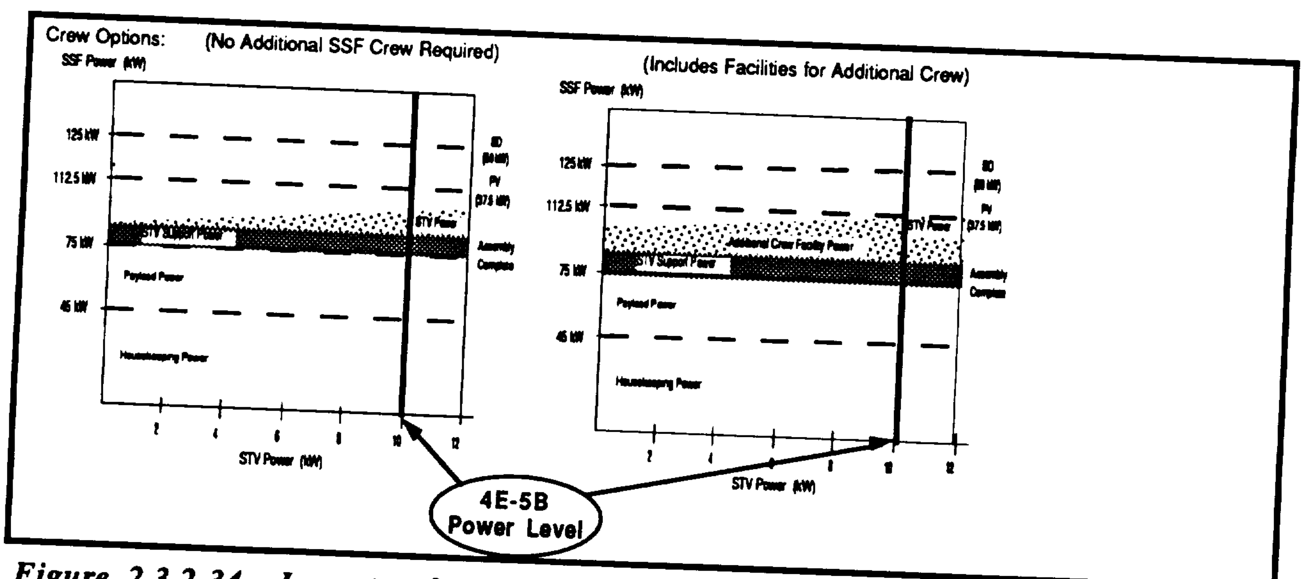


Figure 2.3.2-34 Impacts of Providing Power to the LTS Configuration

2.4 Subsystem Analysis

With the completion of the system level and mission studies as well as a LTS configuration recommendation, there existed sufficient data to support a detailed and comprehensive study and analysis activity in the subsystems that make up the LTS. Through the configuration analysis effort, three key subsystems were identified; avionics, propulsion, and aerobrake. The avionics subsystem analysis addressed power, weight, built-in-test equipment, and technology issues with a goal to provide significant program pay-offs. Propulsion studies addressed primary propulsion and reaction control issues as well as utilization of the propellant to support power and life support systems. Aerobrake studies focused around materials, design, and operational issues. Figure 2.4-1 shows the relationship the subsystem analysis activities have with the overall study and analysis task.

Avionics Analysis—Three distinct classes of requirements were defined as a result of this task analysis: 1) cargo type, 2) mission duration, and 3) reusability, providing two primary areas of analysis: 1) reliability and maintenance: and 2) guidance, navigation, and control.

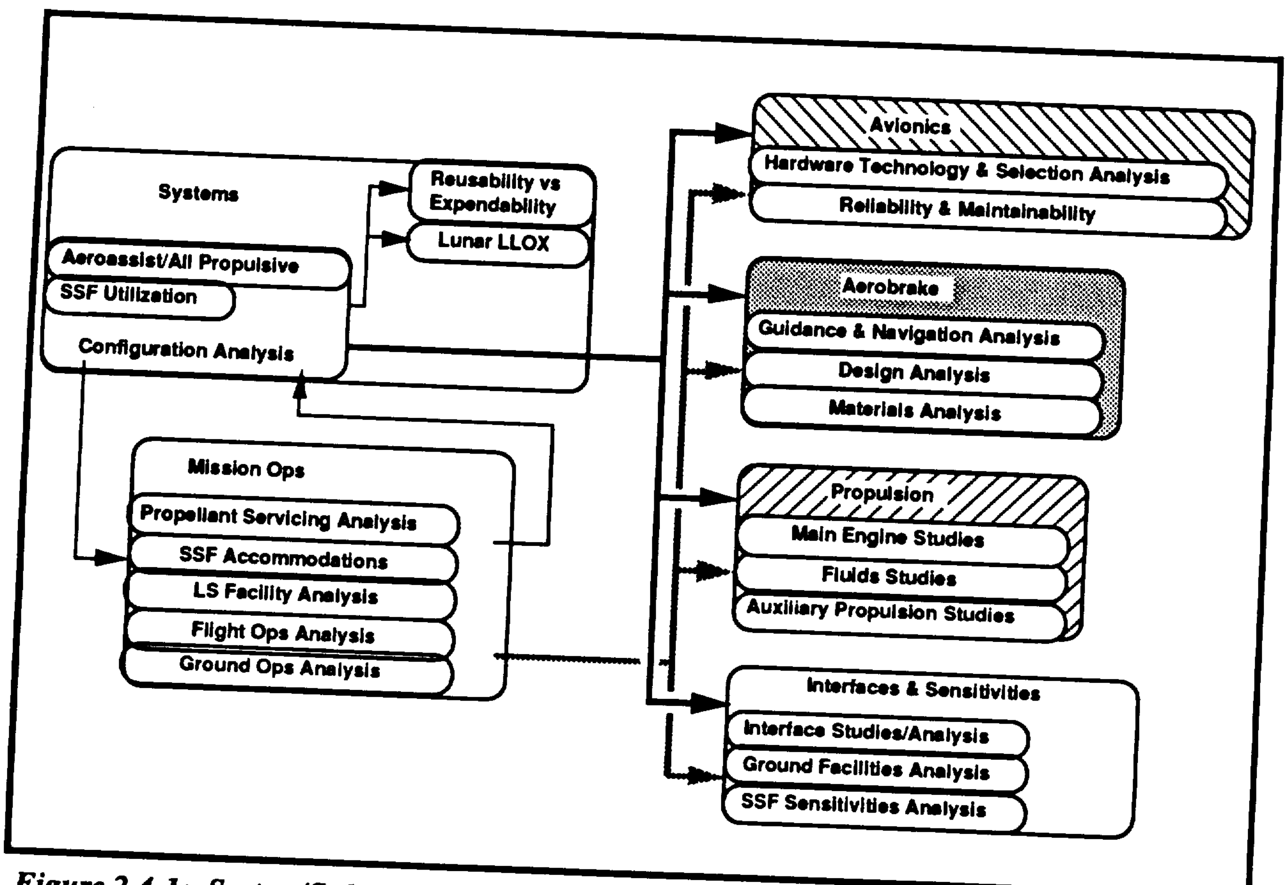


Figure 2.4-1: System/Subsystem Study and Analysis Relationship

Items excluded from these analyses were primary structures and passive subsystems which have no credible failure modes.

Reliability & Maintenance—Two missions were derived from the requirement for reusability; the expendable system and a system which requires periodic servicing. Achieving a reliability above 96 percent for electrical devices requires the use of redundant 'black boxes'. Electronic subsystem designs for space vehicles have achieved outstanding reliability results, although the cost to maintain them is considerable. For example, during the design phase of the shuttle program long term maintenance requirements received little emphasis. The recommendation from the STATS conference was that future systems maintainability should be addressed beginning in Phase A in order to achieve the required reliability levels. Analysis identified two types of navigation systems required for deep space missions, 1) short term navigation using an inertial navigation unit, and 2) and navigation updates using either ground-based ranging or on-board autonomous navigators. The LINS (Laser Inertial Navigation System) used by the Transfer Orbit Stage (TOS) program as a short duration navigation system, represents the most modern, qualified navigation system available for use in space vehicles. A second generation of laser navigation systems is presently under development for the Titan IV/Centaur program, while on the horizon a new set of inertial sensors configured in a hex-head configuration are being pursued.

Several disadvantages were found to be paramount in the use of the Deep Space Communication Network for long duration navigation. The complexity of current communication systems to meet the FO/FO/FS requirements of a manned systems are substantial. The development of on-board optical navigators, Figure 2.4-2, represents a new approach for long duration autonomous navigation.

Guidance requirements for the LTS/STV missions were found to be similar to those employed during the Apollo missions with the exception of the aerobrake deceleration system. Martin Marietta proposes that the LTS/STV program baseline Lambert guidance, currently implemented in NASA's manned space systems, for long duration main propulsion maneuvers, cross product guidance for short duration maneuvers, and explicit guidance for lunar landing.

<u>**Propulsion Analysis**</u>—The propulsion study was broken into 3 areas of interest; engines (type, quantity, evolution), fluid management (transfer, settling, pressurization), insulation (boiloff, type, thickness), and RCS (type, size, location). Although cryogenic propellant was the primary baseline for the STV study, three types of engines were initially evaluated as candidates for use

on an LTV and LEV vehicle to become familiar with some of the system performance parametrics associated with different engine types. Engine analysis involved primarily cryogenic Advanced Space Engine (ASE) and RL-10 derivatives, storable, and Nuclear Thermal Rocket (NTR). The storable engines fall into two categories -

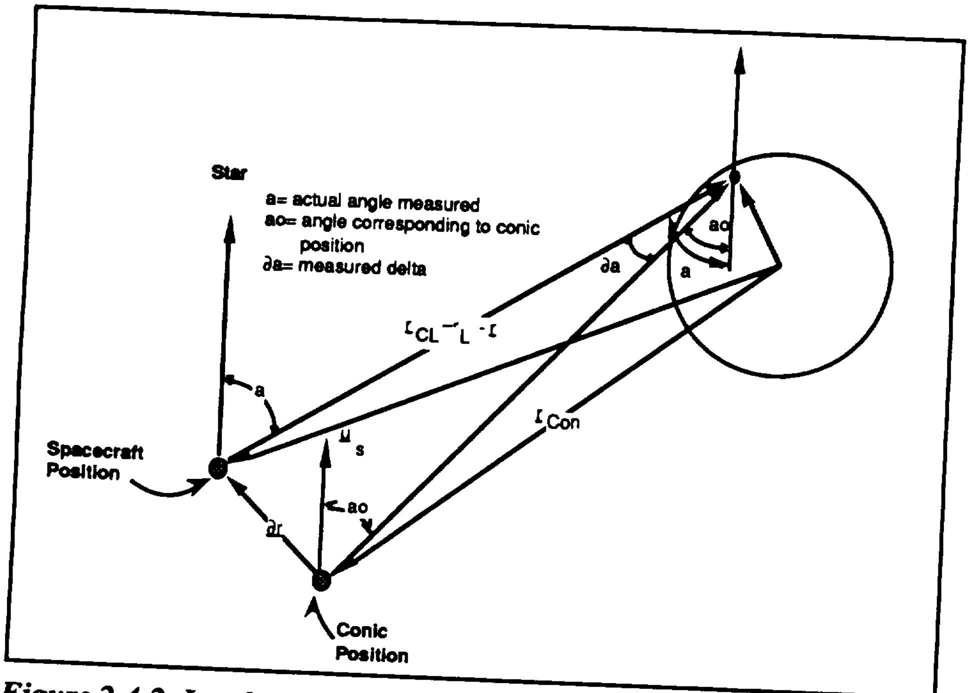


Figure 2.4-2 Landmark Navigation Approach

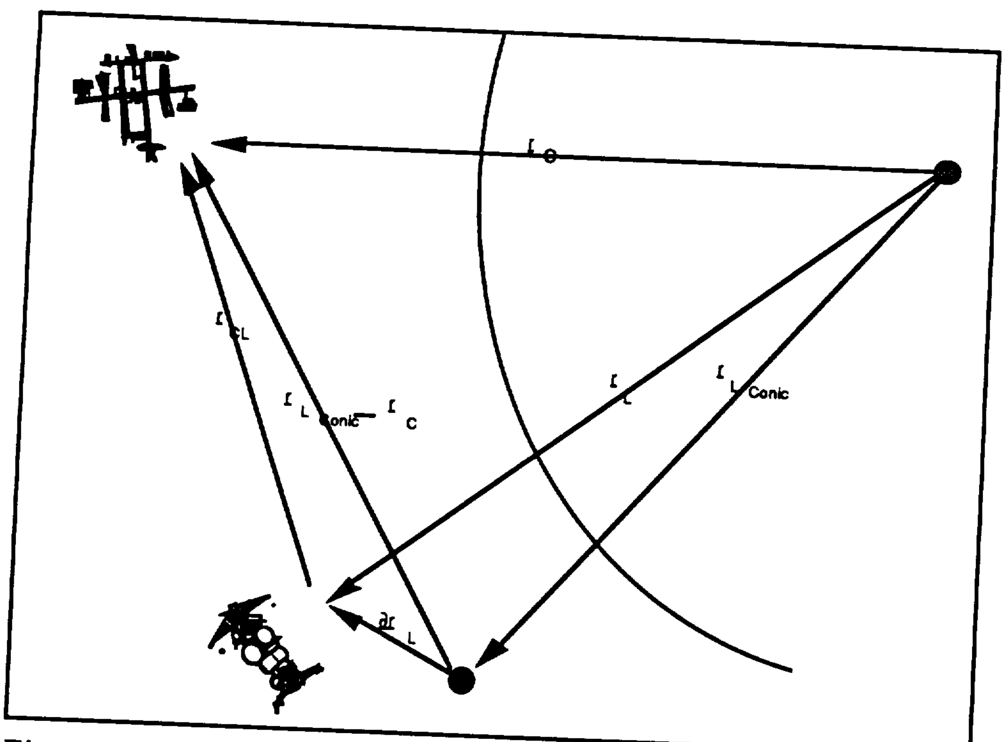


Figure 2.4-3 Optical Navigation As Used for Rendezvous

pump-fed, such as the XLR-132 engine and pressure-fed, most likely the Apollo Lunar Excursion Module Descent Engine. The Nuclear Thermal Rocket engine were based on NERVA (Nuclear Engine for Rocket Vehicle Applications) technology developed in the 1960's. The comparison of these different engine combinations is shown in Figure 2.4-4.

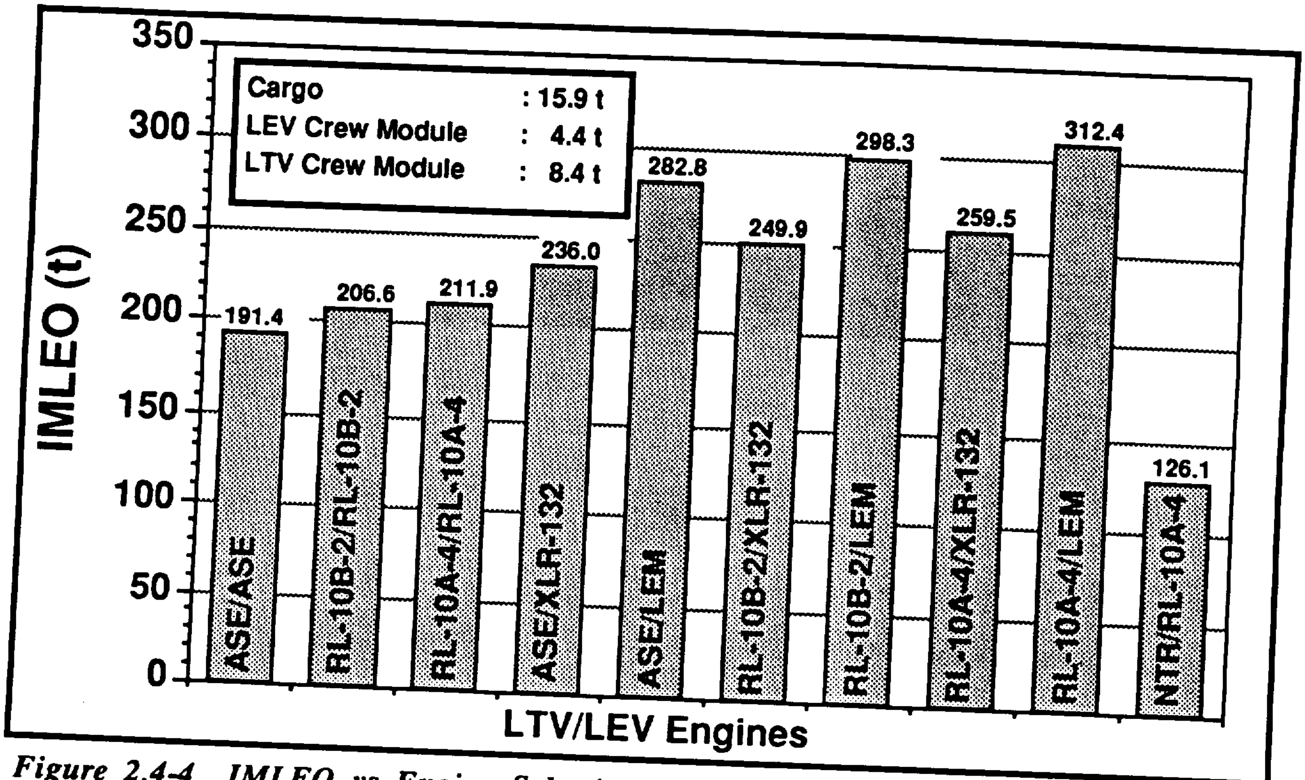


Figure 2.4-4 IMLEO vs Engine Selection For First Flight

A major concern of space basing a system is the effort required to perform engine changeout. Figure 2.4-6 is a representation of a potential engine changeout scenario. Based on a detailed engine analysis, several issues associated with selecting the number of engines for a LTS vehicle were identified. These issues include number of engines for a lander, landing control, engine out strategy, engine system reliability, and LTS aborts relative to number of engines. Evaluation of these issues generated several key engine parametrics including; single engine thrust for LTV / LEV, optimum thrust and throttling requirements vs. no. of engines for 4 and 6 engine systems, single propulsion engine thrust optimization lunar throttling range required, and ASE vs RL-10 Analysis.

A detailed evaluation was made of the fluids required by the LTS/STV vehicle system during the various phases of its operational life. This evaluation considered those fluids required at a launch facility, and also at SSF or other LEO node. It considered first the initial mission, which is likely to be expendable and which may not integrate all of the eventual technologies. It may also

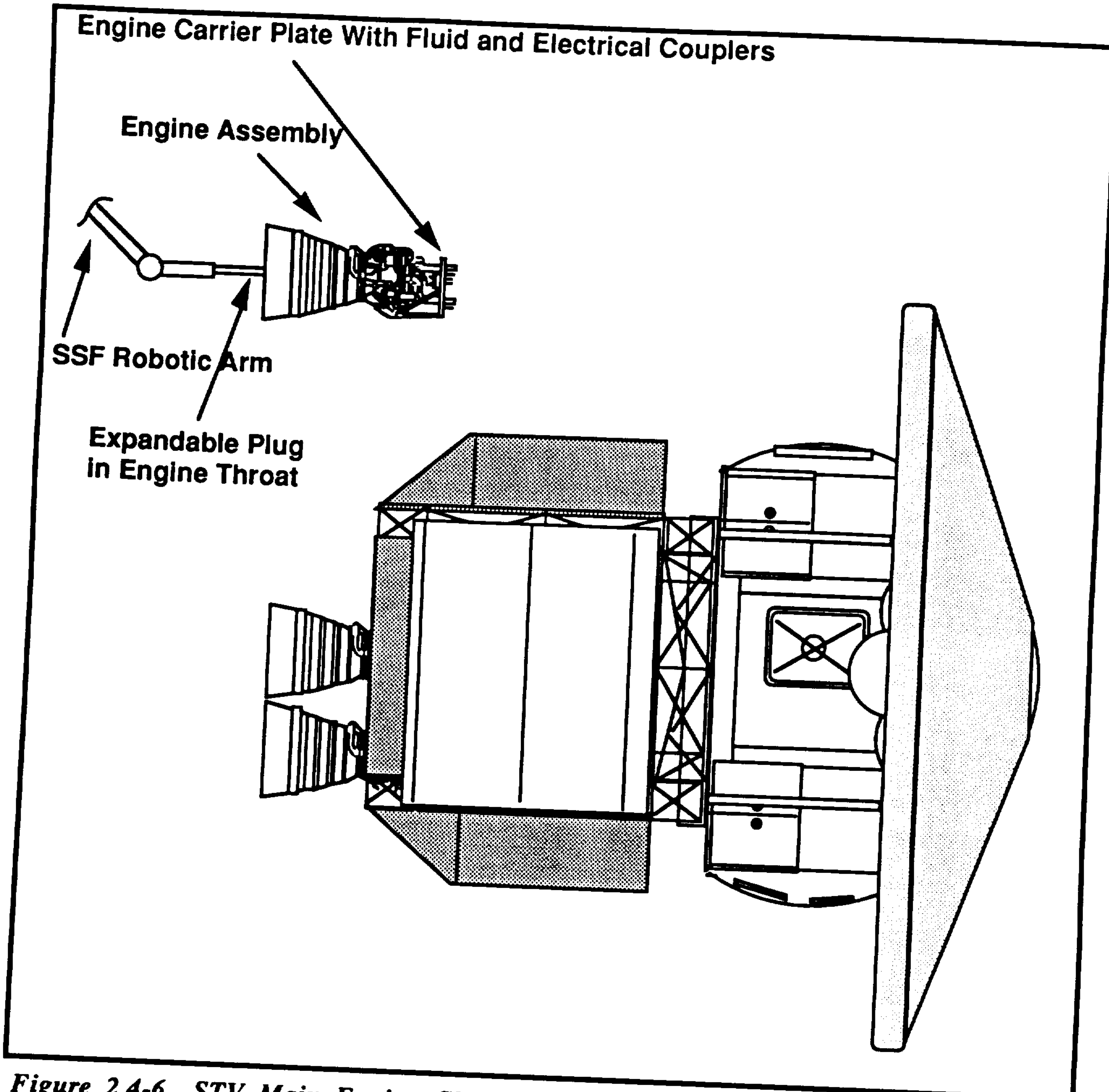


Figure 2.4-6 STV Main Engine Changeout Scenario

require little or no support from SSF. Next it considered that the LTS/STV is operational, but is basically a core vehicle, useful for near earth missions, or for missions that do not require drop tanks. Since the ultimate cost of the LTS/STV system will be strongly influenced by the cost to provide operational support, it is imperative to limit the operational fluids to the absolute minimum number. Table 2.4-4 lists the operational fluids and their servicing locations. The preliminary fluid schematic for the core vehicle, shown in Figure 2.4-7, incorporates the technologies necessary to prevent servicing of a number of fluids. The core tank pressurization system is shown to be autogenous. The RCS system is shown as it might appear using a H/O system, with gas generators providing heat and power to gasify the propellant and allow storage as a high pressure gas. Fuel cell hydrogen and oxygen, and oxygen for the crew is supplied through molecular sieves. Engine functions do not require helium. The implementation of this schematic into the operational system is shown in Figure 2.4-8, including the routing of the propellant lines from the drop tanks to the core vehicle and from the core vehicle to the aerobrake.

Table 2.4-4 Fluid Systems Support Required of ETO & SSF

Initial STV Mission	Operational STV Mission	GEO or Heavy STV Mission	Lunar Mission
ETO			
H/O Supply-Core H/O Vent GN2 Purge-Cargo Bay HP H/O-Integral RCS	H/O Supply-Core H/O Vent Gn2 Purge-Cargo Bay	H/O Supply-DropTnks H/O Vent GN2 Purge-Cargo Bay	H/O Supply-DropTnks H/O Vent GN2 Purge-Cargo Bay LN2 Supply-Breathing H2O Supply-Crew/ Shield
(Technology Driven) H/O Supply-Fuel Cells HP He-Engine HP He-Tank Pressn Hydraul Fluid-Gimb Act N2H4 Supply-RCS N2H4 Haz Vent	H/O Supply-Fuel Cells HP He-Engine HP He-TankPresn HydraulicFluid-Gimb Act N2H4 Supply-RCS N2H4 Haz Vent	H/O Supply-Fuel Cells HP He-Engine HP He-Tank Pressn Hydraulic Fluid-Gimb Act N2H4 Supply-RCS N2H4 Haz Vent	H/O Supply-Fuel Cells HP He Engine HP He-Tank Pressn Hydraulic Fluid-Gimb Act N2H4 Supply-RCS N2H4 Haz Vent
SSF (or other node)			
None	H/O Supply-Core	H/O Supply-Drop Tanks	H/O Supply-DropTnks H2O Supply-Crew/ Shield LN2 Supply-Breathing
	(Technology Driven) H/O Supply-Fuel Cells HP He-Engine HP He-Tank Pressn Hydraulic Fluid-Gimb Act N2H4 Vent-RCS	H/O Supply-Fuel Cells HP He-Engine HP He-Tank Pressn Hydraulic Fluid-Gimb Act N2H4 Vent-RCS	H/O Supply-Fuel Cells HP He-Engine HP He-Tank Pressn Hydraulic Fluid-Gimb Act N2H4 Vent-RCS

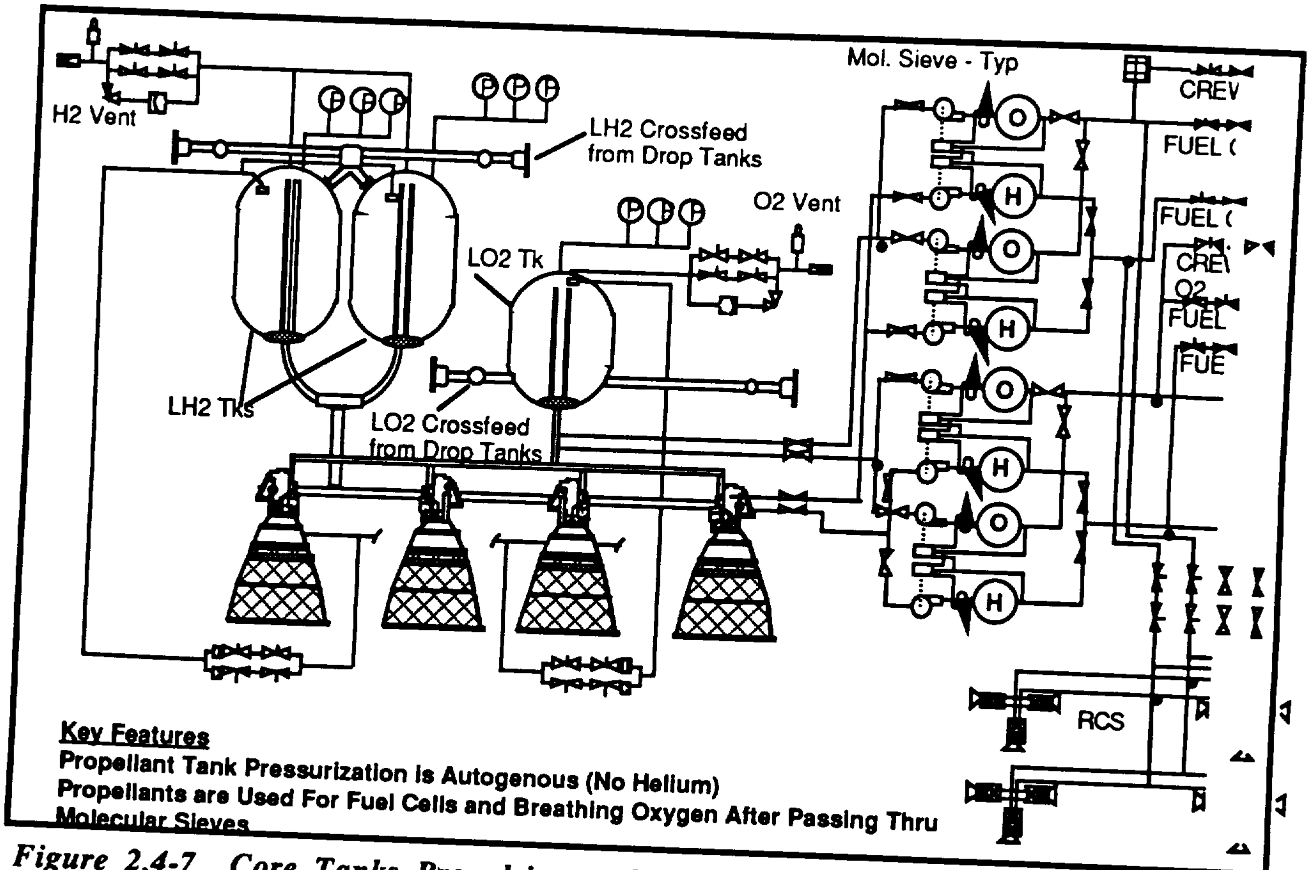


Figure 2.4-7 Core Tanks Propulsion and Fluids Schematic

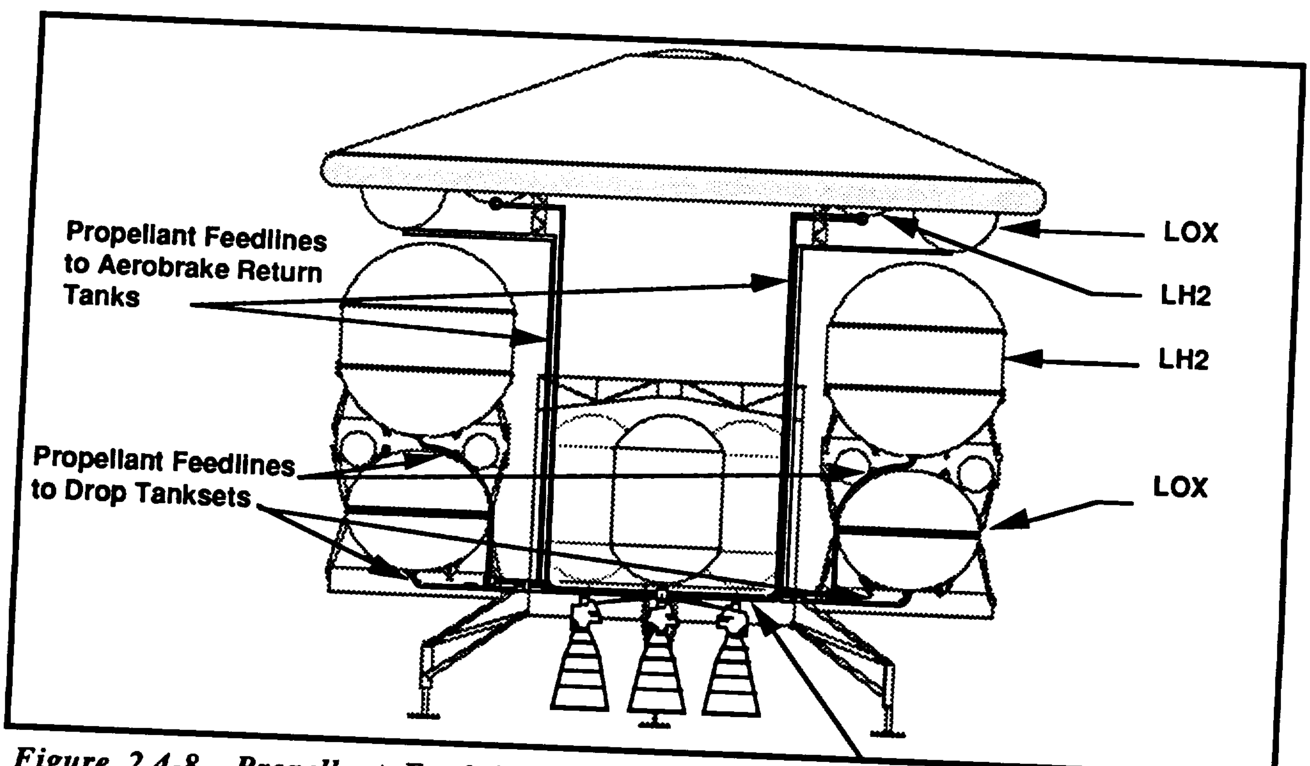


Figure 2.4-8 Propellant Feed System

Other propellant studies conducted throughout this study included settling of the propellant and system pressurization during zero-g phases of the mission. For propellant settling, a number of methods may be used to settle the cryogenic propellants in order to accomplish transfer to another tank or to the engines. These methods have been evaluated by detailed mission phase and are shown in Figure 2.4-9.

	Advantage: Autogenous LOX	Advantage: Helium Pressurization	Comment
Cost	x		Equipment Costs May Be Similar. Operational Costs Are Higher Using Helium, Assumming Equal Maintenance
Equipment Weight	x	x	1235 lb (561.4 kg) for Autogenous against 1388 lb (631 kg) for GHe Pressurization, plus gas weight on previous chart.
Complexity	x	x	Complexity of Both Methods Are Similar Since Similar Types of Components Are Involved.
Risk/Reliability	x		Slightly Greater Risk Is Associated with Helium Pressurization Due to Higher Pressure Requirement (3500 psia vs 300 psia for Autogenous)
Responsiveness	x		Helium Loading Is Eliminated. Non-condensible GHe in Tank Complicates On-orbit Resupply
Vehicle Performance Effect		x	Lower Tank Ullage Mass Is Left for GHe Pressurization after Engine Burn - Autogenous Penalizes Vehicle To Carry an Additional 2119 lb (961 kg) of Propellant
Operations	x		Autogenous System Reduces Number of Dfferent Fluids That the Vehicle Needs To Carry by Completely Eliminating GHe Usage and Eliminates On-orbit GHe Resupply Requirement

Figure 2.4-9 LOX Autogenous vs GHe Pressurization Summary

Several studies were performed to evaluate preliminary insulation concepts. These studies addressed insulation concepts for the LTV TLI and LLO drop tanks and the LEV while on the lunar surface and onorbit (Figure 2.4-10 & 11), propellant transfer under low gravity conditions, MLI/Boiloff Weight Parametrics, Effect of Orbital Storage Time on Boiloff, and Effect of Tank Size on Boiloff.

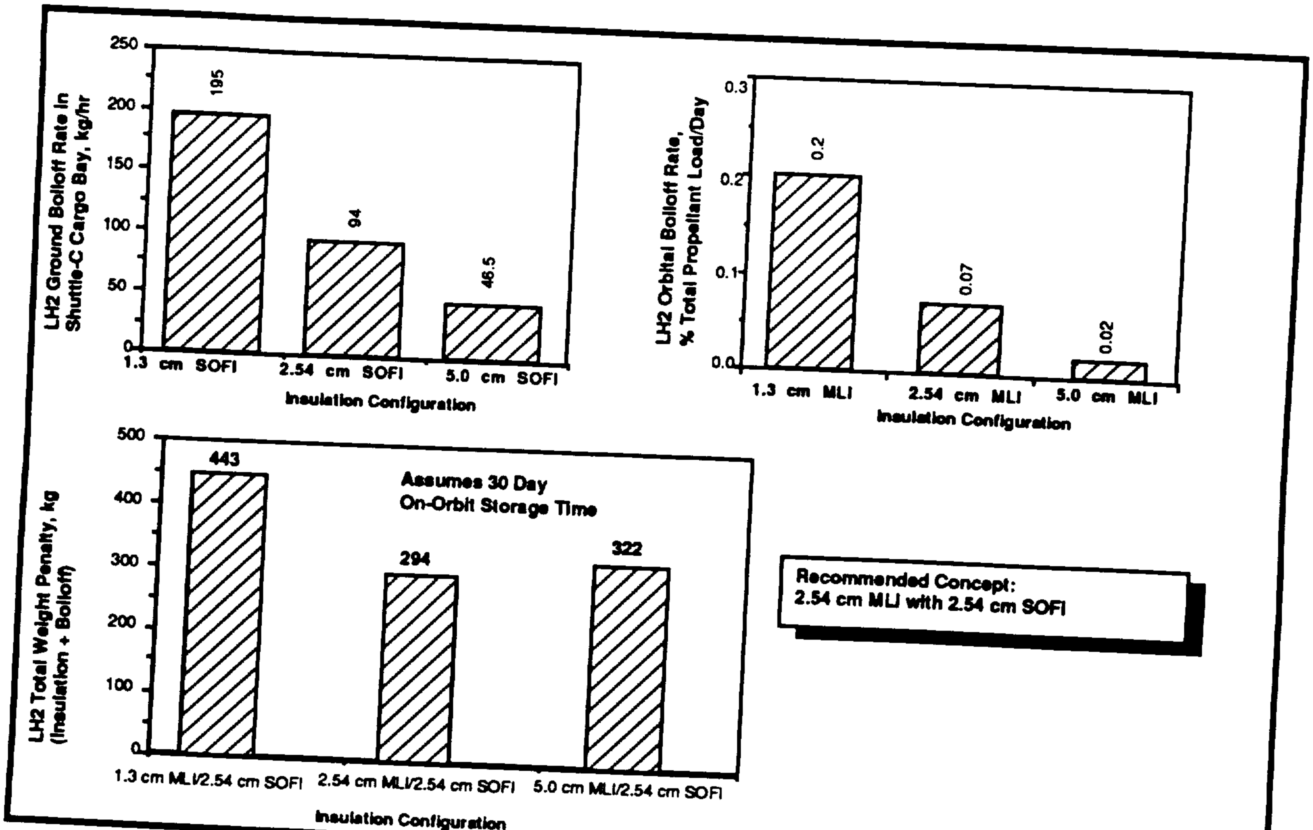
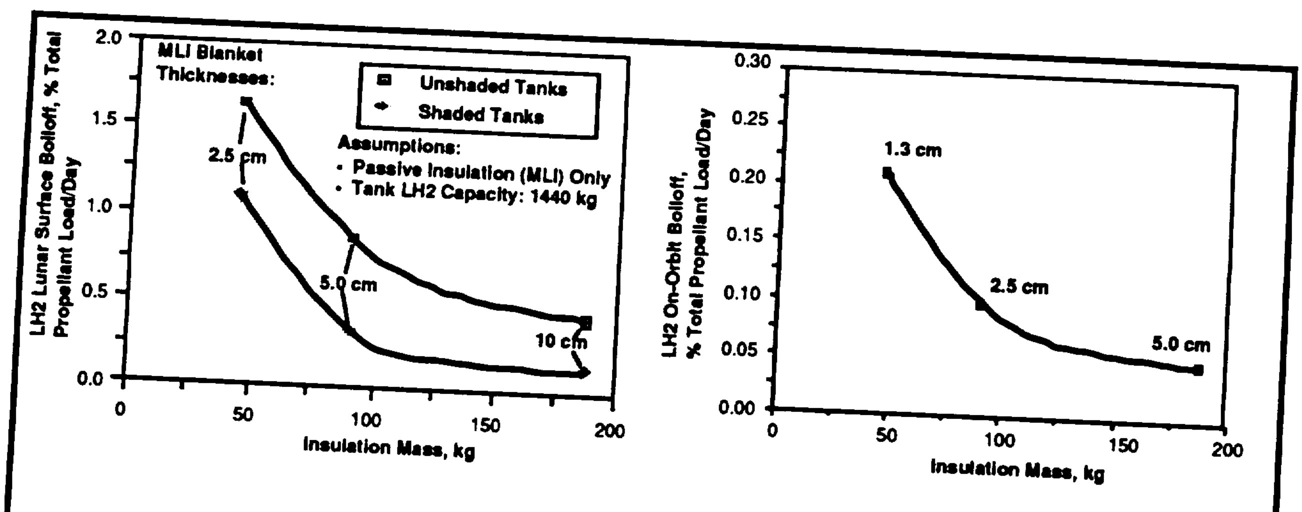


Figure 2.4-10 LTV Insulation Study Results



Preliminary Insulation Configuration:

- 5.0 cm MLI Selected Insulation Baseline; Provides Reasonable Boiloff Losses and Insulation Weight

Figure 2.4-11 LEV Insulation Study Results

Definition of the RCS and the corresponding thrust levels required for the LTS/STV included system sizing, RCS Thruster Layout (Figure 2.4-12), and RCS System Options (Table 2.4-5 and Figure 2.4-13).

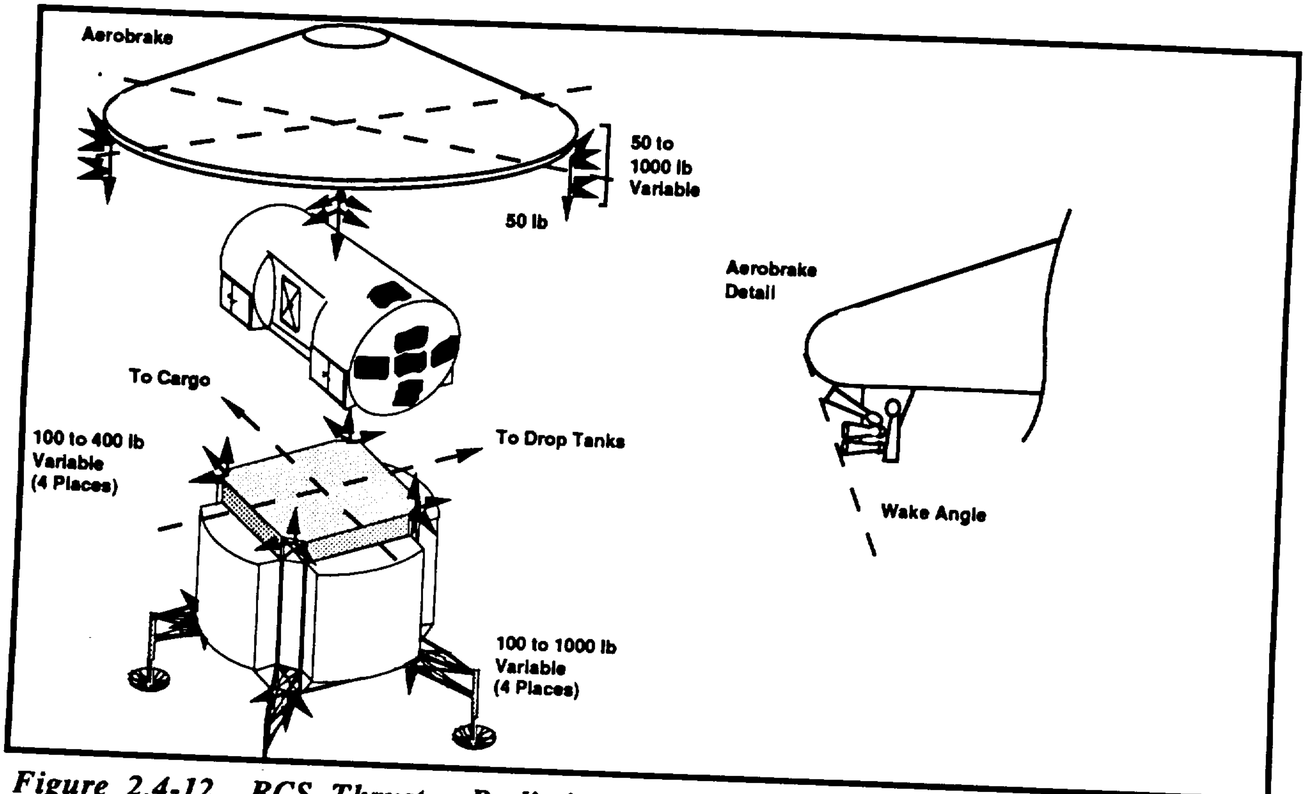


Figure 2.4-12 RCS Thruster Preliminary Arrangement

Table 2.4-5 RCS System Options

Options If Ground Based	Options If Space Based	Advantages	Disadvantages
BiPropellant	Eliminated for Space Based Option Due to Storable Fluid Servicing Complexities (Non-Integrated)		
MonoPropellant			
Cold Gas	Cold Gas	Potentially Simpler System	Low Performance High Pressure Storage Low Density Storage
Bi Gas (H/O)	Bi Gas (H/O)	Emerg. Return to EO Useage Flexibility High Flowrate Good Long term Stg	High Pressure Storage Low Density Storage Complex System
Cryo (H/O)	Cryo (H/O)	Low Pressure Storage High Density Storage High Isp	Large Thermal Losses Poor Long Term Storage Complex System
Supercritical (H/O)	Supercritical (H/O)	Low Pressure Storage High Density Storage	Low Flowrate (Prepres or High Demand RCS) Poor Long Term Storage Complex System

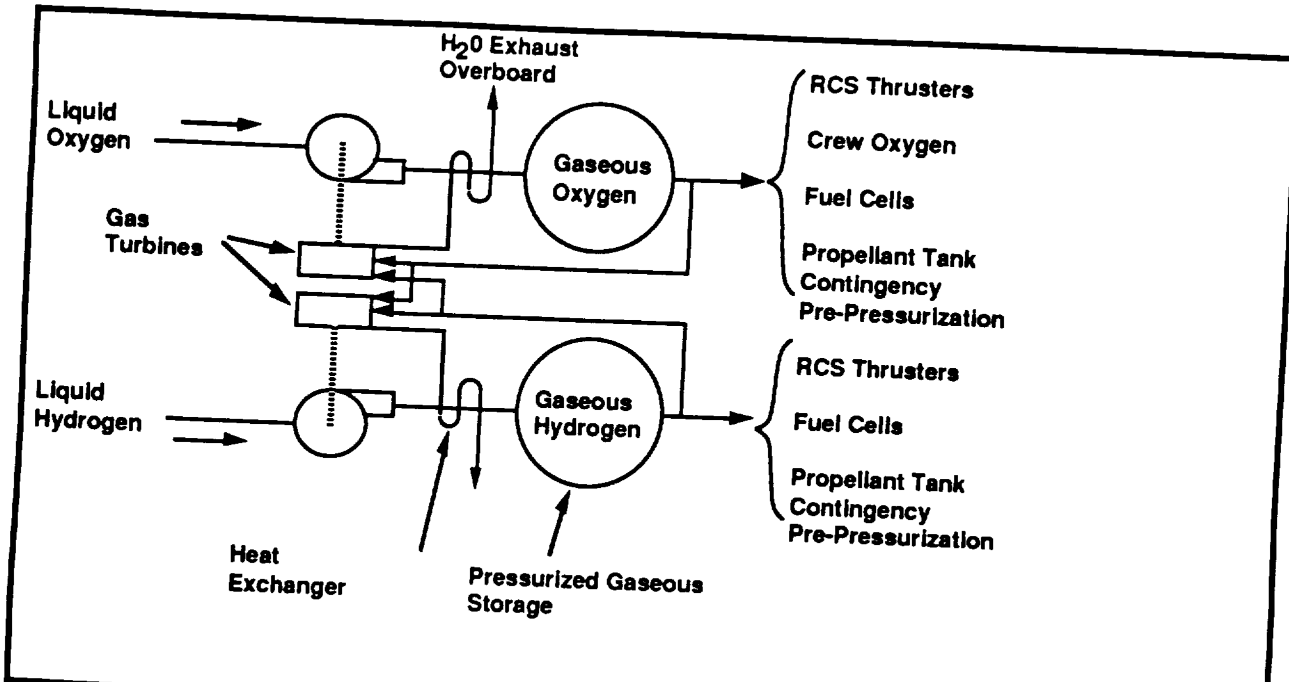


Figure 2.4-13 Integrated RCS System - Gaseous H/O

Aerobrake Analysis - The aerobrake activity conducted as part of the STV study program provided an assessment of the benefits and issues associated with two brake configurations, rigid and flexible, and reexamined some initial aspects of guidance and control of the system during the aeroassist maneuver. The evaluation of the rigid aerobrake was conducted with four configurations, an eight panel segmented design with ribs either integral to the panels or with folding ribs hinged to the center section, and a three piece design with the pieces either separate or hinged together. The eight panel rigid aerobrake with folding ribs is shown in Figure 2.4-14. It is similar to the integral rib rigid aerobrake configuration; the major difference being that the ribs (and struts) are deployable rather than being built into the panel assemblies. In the folded rib concept, the eight panel segments are attached to the ribs after the ribs are deployed by the robotics arm. Rotating the struts and pinning them to the ribs completes the LEO assembly. The flexible aerobrake concept shown in Figure 2.4-15, contains 16 ribs covered by Tailorable Advanced Blanket Insulation (TABI) material outboard of the 24 foot diameter tile protected rigid center section. A single hinged strut braces each rib. The TABI is permanently attached to the center section where it adjoins the rigid TPS material. An aerobrake diameter of 45 foot is compatible with packaging in a 25 foot diameter cargo bay launch vehicle although aeroheating levels and TABI temperature limits may require a larger aerobrake diameter (lower ballistic coefficient) to keep the TABI within its temperature limits.

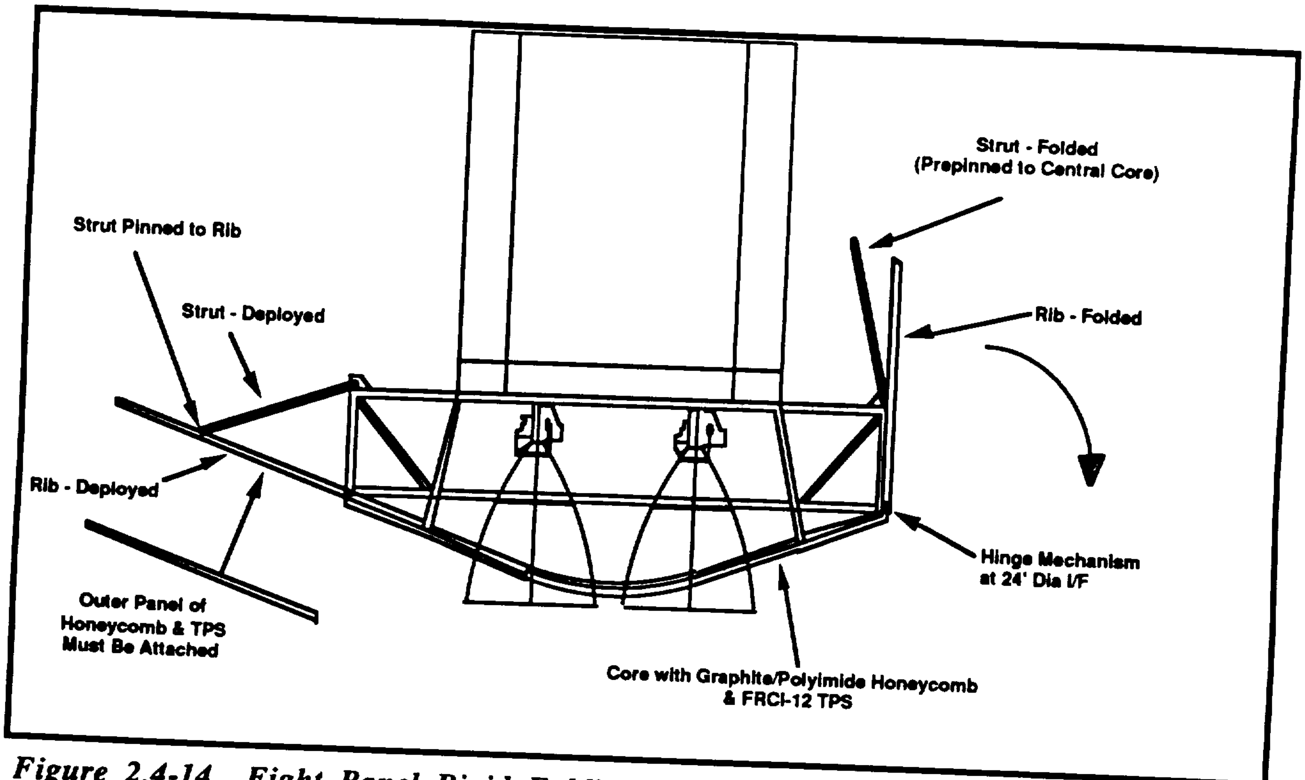


Figure 2.4-14 Eight Panel Rigid Folding Rib Aerobrake

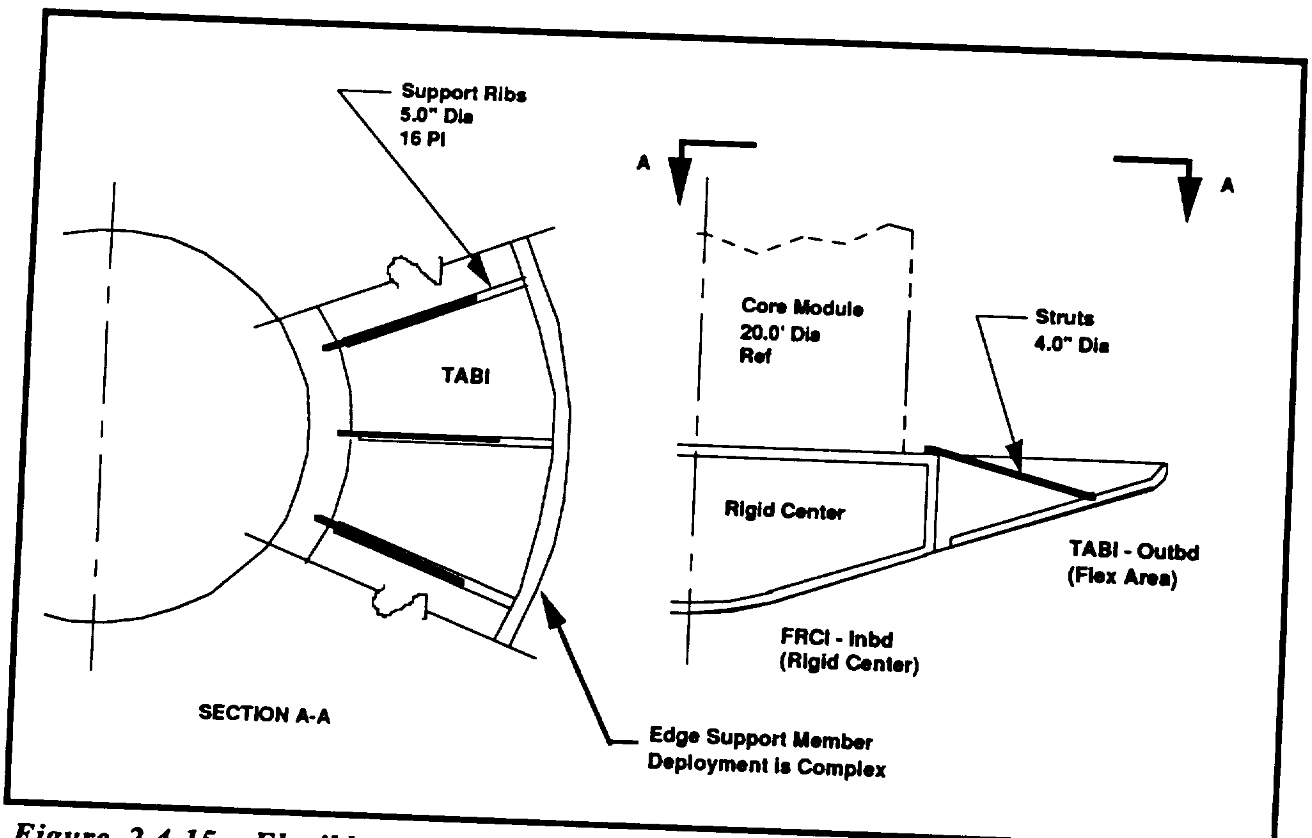


Figure 2.4-15 Flexible Aerobrake - Deployed

The rigid aerobrake appears to afford a somewhat lower risk approach based on these preliminary configuration definitions. At this time, however, it appears the potential for simplifying on-orbit assembly, along with the other identified potential advantages, are sufficient to warrant the continued to pursuit of flexible and rigid designs. Also, it appears that further optimization of the hinged rigid three-piece design could result in achieving some of the deployment benefits associated with the flexible concept.

Structures Analysis—The structures analysis and study activity conducted in the STV Study program provided an in-depth assessment of the LTS structural material and design configuration. The primary area of focus surrounds the design and material selection for the propellant tanks. These areas represent a significant impact on the overall transportation system weight, manufacturing, and LEO assembly requirements. The methodology used to analyze conventional and nested-dome tank configurations consisted of two phases. The initial phase produced a recommended design for both the tank domes and the interconnecting structure for the intertank and the nested dome configurations. The intertank design is shown in Figure 2.4-16, the design

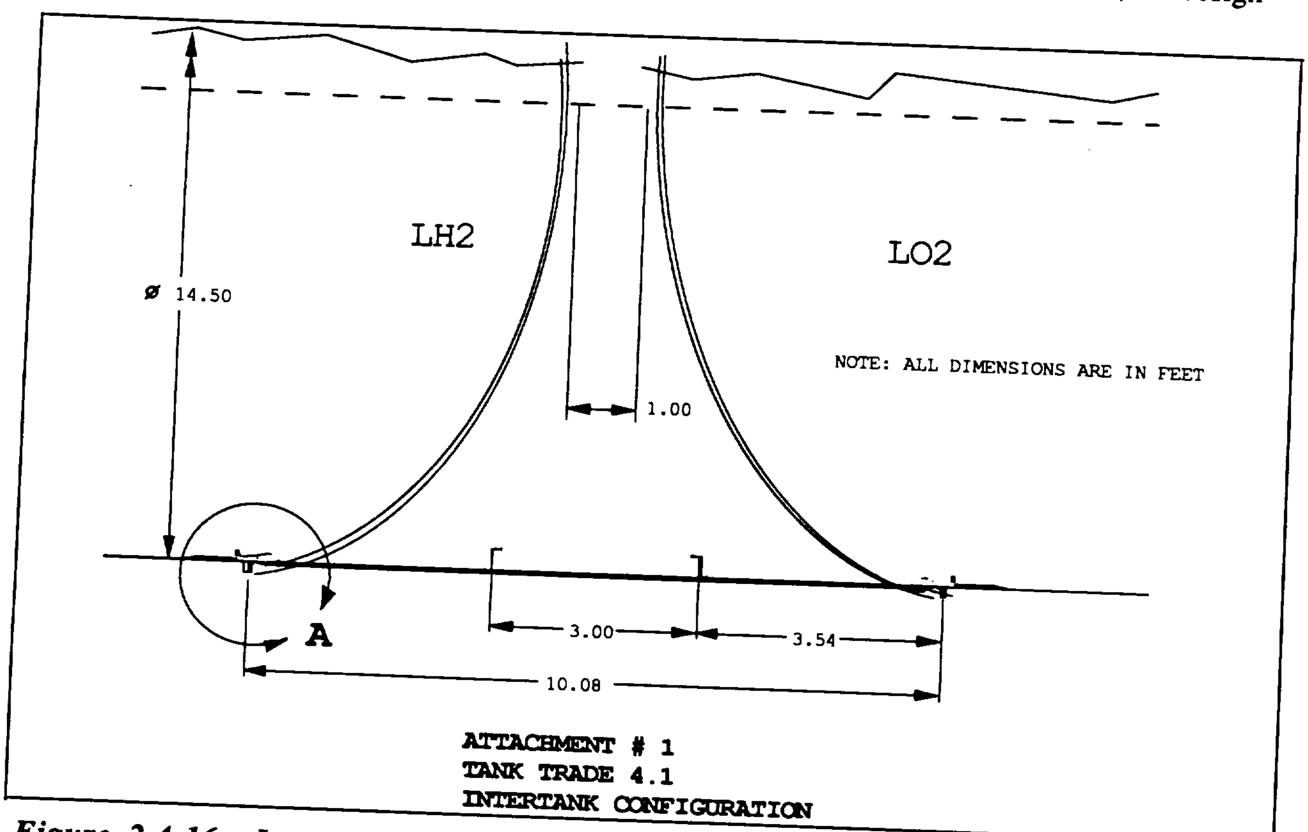


Figure 2.4-16 Intertank Configuration

for the nested dome configuration is shown in Figure 2.4-17. In the second phase, these designs were evaluated for weight, estimated cost impacts, schedule impacts and constraints, and tooling impacts.

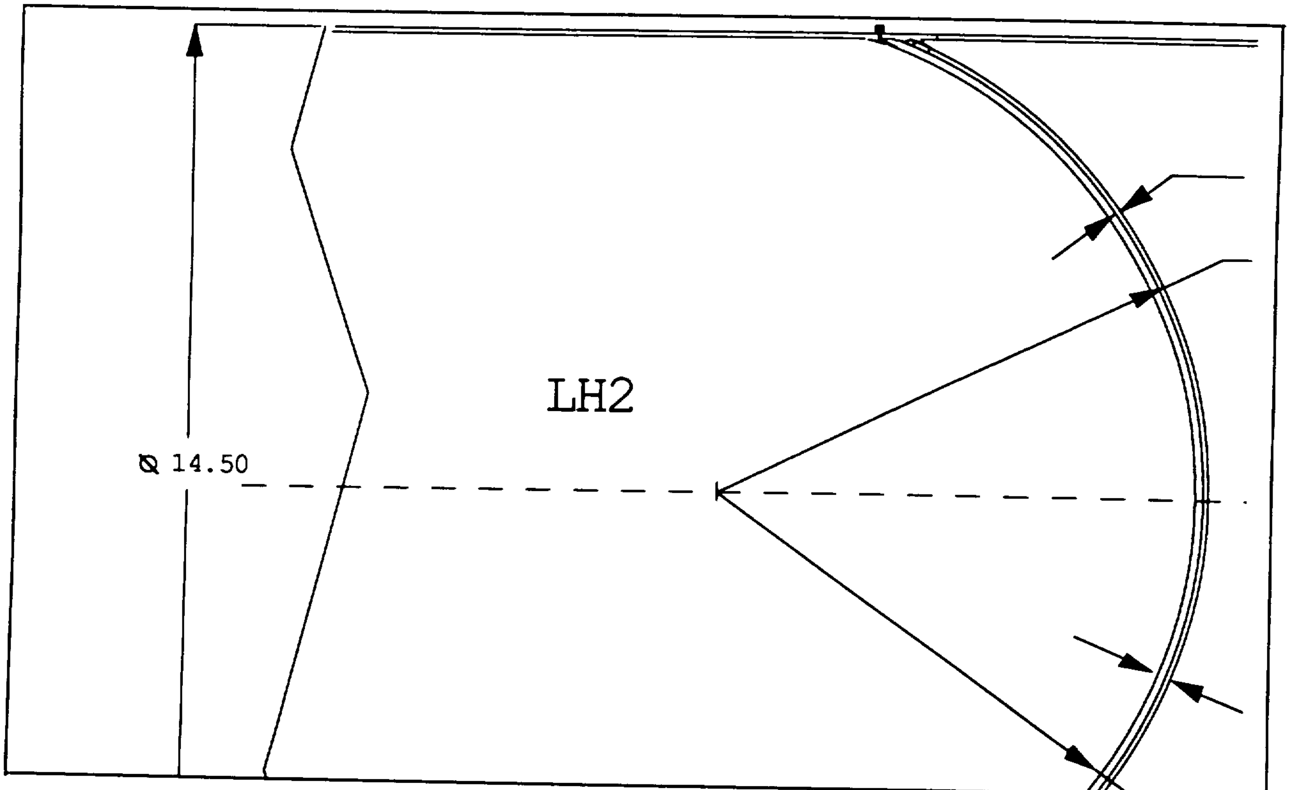


Figure 2.4-17 Nested Dome Configuration

The study's recommendation is that the intertank configuration remain the baseline design since the small weight reduction provided by the nested dome configuration does not offset the additional schedule risk and manufacturing difficulties anticipated with the nested dome configuration.

Another study was a comparison of 2219 Al Alloy with the baselined Weldalite™ to determine the most cost effective structure. Key issues addressed were weight, cost, and producibility. The basic system impact is manufacturing the various vehicle components, one of which is the propellant tanks. Due to the near term cost of Weldalite™, a trade on the weight benefits of Weldalite™ against a more cost effective method of manufacturing propellant tanks was suggested. The analysis was conducted in two phases. The initial phase produced a recommended tank set design using both Weldalite™ and 2219 aluminum alloy material. In the second phase the designs were evaluated for weight, estimated cost impacts, schedule impacts and constraints, and tooling impacts. The recommendation emerging from this study is that further analysis will be required as the configuration definition matures. If weight/performance is most critical, Weldalite™ should be incorporated into the design since it represents a weight saving potential over 2219 aluminum alloy as well as processing increased mechanical properties. If material cost is key, 2219 aluminum alloy should be incorporated into the design because of its manufacturing cost advantages, which have been established through proven manufacturing techniques and tooling requirements. An alternate approach would be to use Weldalite™ for the more highly stressed components and 2219 aluminum alloy where section properties are believed to be more important than mechanical properties.

Crew Module Analysis—The analysis and study activity performed against the crew module, provided the operational and design data incorporated into the final LTS configuration recommendations. The primary areas of focus involved the basic configuration of the crew module itself as well as specific operational concerns addressing crew visibility. Results of these studies include LTS crew module configurations as well as key life support and safety issues relative to operation and rescue. The objective was to select an overall configuration for the crew module(s) best suited for the LTS mission. The key issues addressed focused on whether the crew module(s) require a new design, a modification of the Apollo design; one or two modules; or a hybrid version being developed as part of the LTS; and whether the LTS crew module(s) should incorporate an EVA air lock or if depressurizing the entire cab would be necessary. In addressing these issues, an assessment of the operational scenarios determining crew module quantities based on nodal operations - such as rendezvous and docking functions in Low Lunar Orbit (LLO) - and determining the sensitivities of differing crew module configuration to mission scenarios, the operational concepts, and demonstrated growth capabilities were considered. The analysis methodology approach to analyze the crew module configurations was comprised of three primary phases. Phase I addressed the feasibility of developing a new module versus using the Apollo design, Phase II optimized the quantity of modules, one, two, or a hybrid configuration; and Phase III defined the module sensitivities of mass and volume based on depressurization or addition of an EVA airlock.

Comparison of the LTS crew module to the Apollo Command Module (CM) and the LEM was difficult as the mission requirements are drastically different. New modules is the preferred recommendation over modification of modules designed for different requirements. A derivation of the CM could be used as a crew rescue vehicle; although currently this is not an STV or SEI requirement. Based on this study the hybrid crew module concepts provides no advantages over either the single or separate module concept. The selection of a single module approach versus the two separated modules is dependent on the final LTS configuration. Separate modules are the recommended approach at this time if the LTS is made up of separate transfer and landing (excursion) vehicles; a single crew module is recommended for an LTS that employs a common transfer and landing vehicle. The weight and volume impact for implementing an airlock system in the crew module are extreme; however the entire module can be repressurized enough of times to meet all EVA requirements for a minor weight penalty of 3.5% of the module mass. Therefore, our recommendation is that the cabin be depressurized then repressurized to support EVA activities. The design of the crew module will also incorporate the appropriate number of windows for viewing all critical operations. Every effort will be expended to assure adequate window viewing to provide as large a FOV as possible. Figure 2.4-18 shows the current crew module configuration and the available FOV in both the vertical and horizontal planes, windows have also been provided allowing the crew to observe the rendezvous and docking operation in LLO.

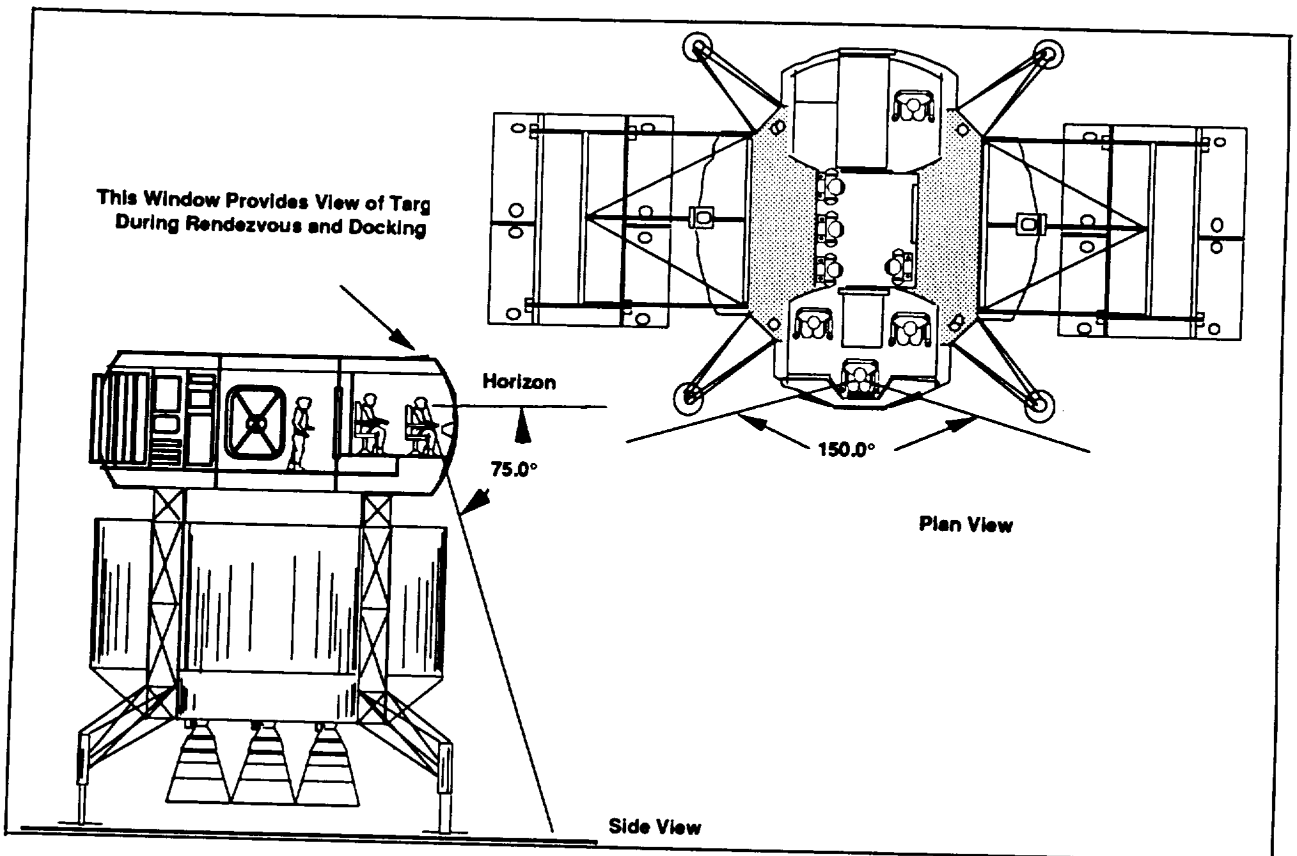


Figure 2.4-18 Crew Module FOV Considerations

3.0 STV CONCEPT DEFINITION

The STV Concept Selection Trade Study analysis shows that the lunar missions impose the most stringent STV requirements. The approach has been to develop a vehicle that meets the design requirements and then evaluates the design to identify the elements that best satisfy the mission requirements for a ground-based STV, a space-based STV, and finally a Mars mission profile.

The STV concept definition for a lunar mission vehicle is based on the requirements in the STV Statement of Work with additional derived requirements from the Option 5 Planetary Surface System documents, and the system trade studies and analyses. These studies and analyses recommend that the orbital mechanics designated as Lunar Architecture #1 (LA#1) best meets these requirements. LA#1 uses a LEO node as the start and finish of the lunar mission for both crew and cargo flights. The LEO node is used for assembly, checkout, and refurbishment. Additional elements of the orbital mechanics require the vehicle to orbit in Low Lunar Orbit (LLO) before descent, to have a lunar trajectory with a free earth return abort scenario, and to return to the LEO node via aerobraking.

Once the lunar mission profile, shown in Figure 3.0-1, was selected, the following key design drivers were integrated into the development and definition of vehicle configuration candidates.

a) The system shall deliver 14.6 tonnes of cargo and 4 crew to the lunar surface and return
b) The system shall deliver 33.0 t of cargo on an unmanned flight to the lunar surface
c) The LEO transportation node shall be Space Station Freedom (SSF)
d) The propulsion system shall use cryogenic propellant
e) The system shall be reusable for a minimum of five missions

These design drivers were also filtered through the subsystems trade study analysis and finally incorporated into the vehicle design.

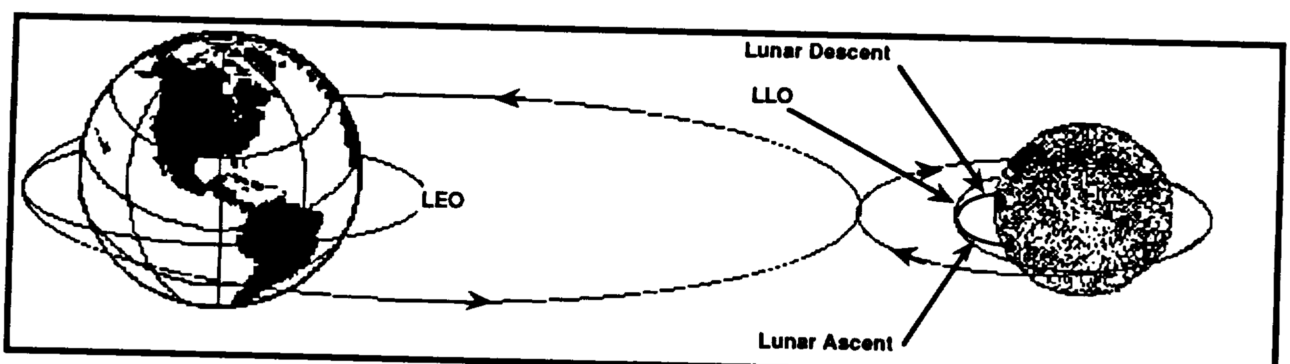


Figure 3.0-1: Lunar Mission Profile

3.1 LUNAR STV CONCEPT DEFINITION

The STV consists of a family of vehicles which share common elements performing both cargo and piloted/cargo missions such as GEO delivery, lunar, and planetary (Mars mission). That portion of the STV family that deals with the lunar missions is called the lunar STV or the Lunar Transportation System (LTS). The LTS is comprised of two mission profiles: (1) the cargo mission capable of delivering 33 tonnes to the lunar surface and (2) the piloted/cargo mission capable of delivering a crew of 4 plus 14.6 tonnes to the lunar surface.

According to a derived study requirement, the final cargo and piloted vehicles would share common elements, producing a family of vehicles that have common structural core, propulsion and avionics equipment, drop tanks, and can be configured for either type of mission with no major modification to these elements. The definition of each vehicle configuration, performance, and mass properties are discussed in the following section.

Piloted Concept Overview—The LTS piloted configuration for the single propulsion system concept is shown in Figure 3.1-1. A crew module, six drop tanksets, and an aerobrake

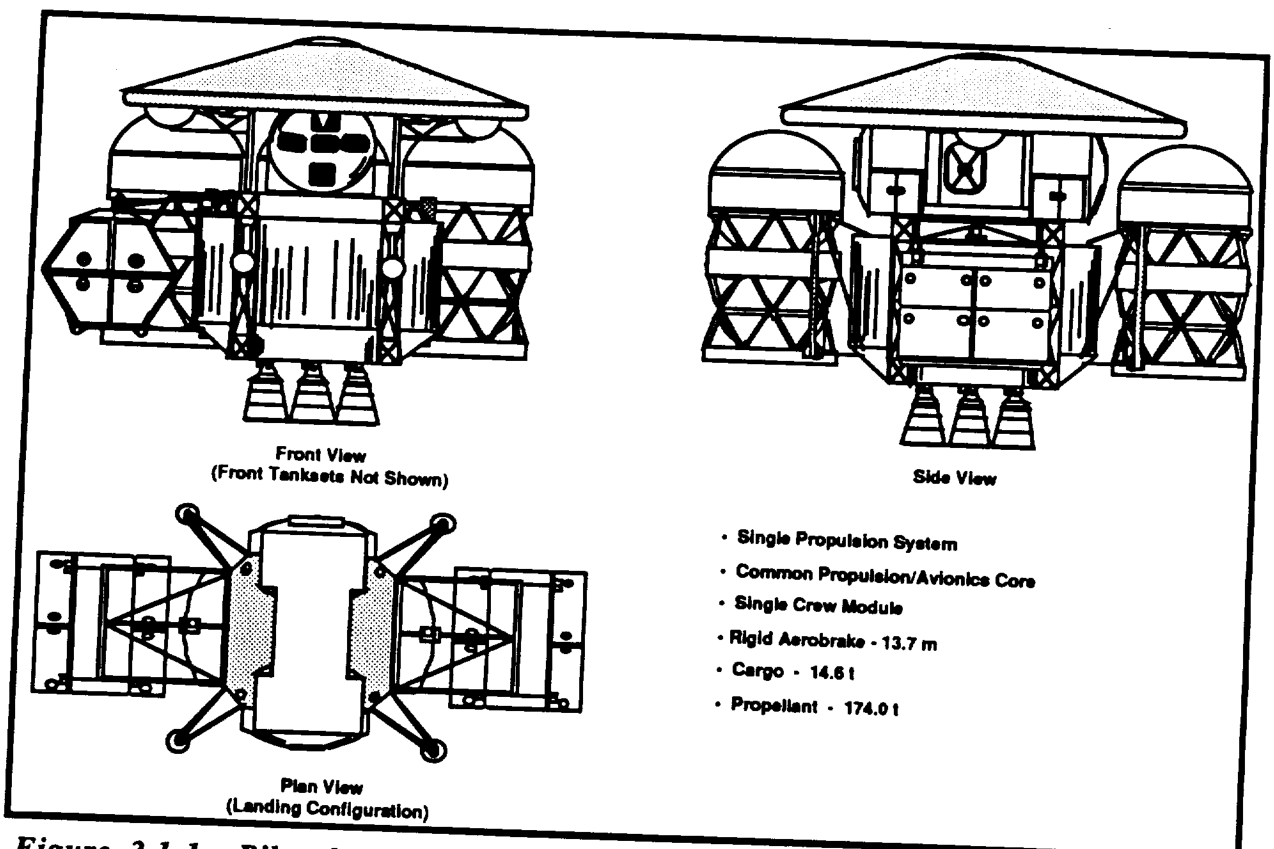


Figure 3.1-1 Piloted LTS Configuration

mass is 27.58 tonnes. The configuration can deliver 15.26 tonnes of cargo (14.6 tonnes cargo plus cargo supports) in addition to the crew of 4 to the lunar surface and return the vehicle and crew to LEO using approximately 174 tonnes of LO_2/LH_2 propellant. TEI and LOI propellant is housed in the drop tank sets, ascent and descent propellant is found in the core, and the return propellant is housed in two sets of tanks within the aerobrake. The 13.72 m rigid aerobrake has been designed to protect the crew during the aeroassisted maneuver before returning to Space Station Freedom.

Cargo Concept Overview—The LTV cargo expendable configuration for the single propulsion system concept is shown in Figure 3.1-2. To form the cargo expendable configuration, a cargo platform (10.5 m x 14.8 m) and six drop tanksets have been added to the propulsion/avionics core. The cargo vehicle dry mass is 18.75 tonnes and can deliver 33 tonnes of cargo to the lunar surface using 146.5 tonnes of LO_2/LH_2 propellant loaded into the drop tanks and core tanks. The flight 1 cargo manifest shown in the plan view is a typical arrangement for the four cargo missions.

Performance Overview—Missions designed for the LTS include piloted, cargo expendable

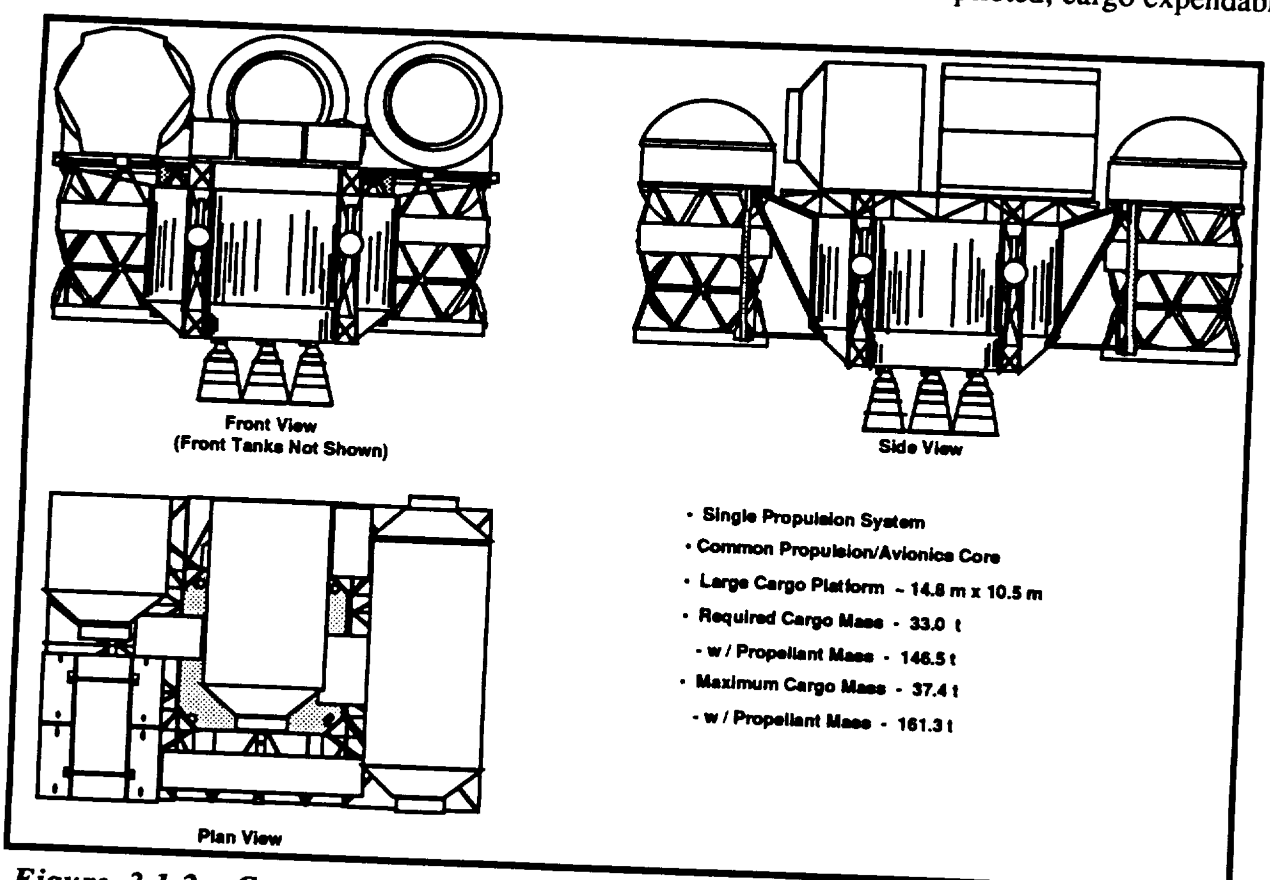
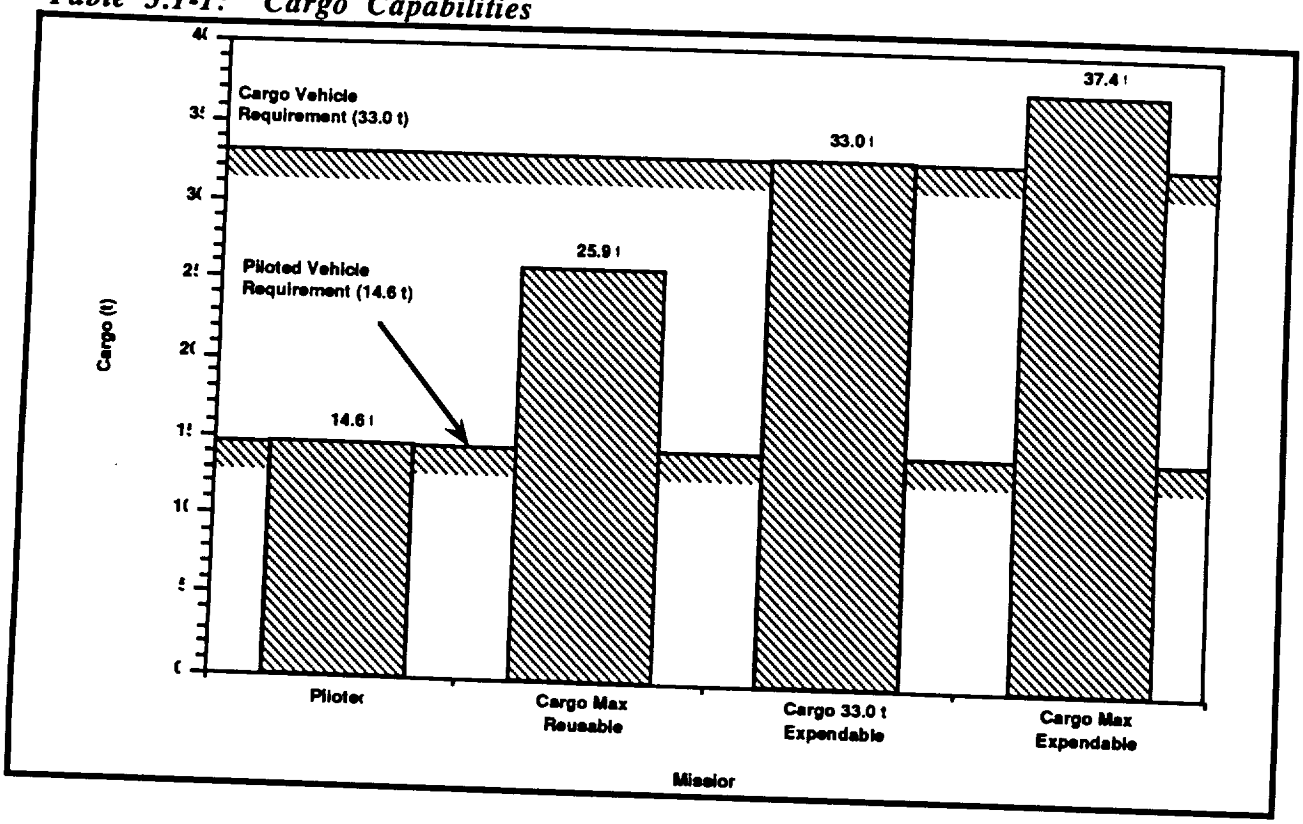


Figure 3.1-2 Cargo LTS Configuration

and an optional cargo reusable. Vehicles sizes, capabilities, propellant loads, and IMLEOs were determined based on the cargo requirements and the groundrules established for the STV study. The piloted mission (crew plus 14.6 tonnes of cargo) was found to be the vehicle sizing driver. Once the baseline vehicle was determined, the cargo capabilities shown in Table 3.1-3 defined a maximum capability for an expendable cargo mission of 37.4 tonnes, or 4.4 tonnes over the required capability. The required delivery of 33 tonnes of cargo is met by offloading 27.5 tonnes of propellant. The optional cargo reusable mission delivers 25.9 tonnes of cargo with a full propellant load and returns to SSF.

Table 3.1-1: Cargo Capabilities



3.2 SUBSYSTEM COMMON ELEMENTS

The common propulsion/avionics core shown in Figure 3.2-1, represents the heart of the single propulsion system family vehicle. Crew module, aerobrake, cargo pallets or platforms, and drop tanksets can be added to form various configurations allowing the STV vehicle family the versatility to capture other missions. The core consists of five internal propellant tanks (4 LH_2 and 1 LO_2 tanks), primary structure and the four landing legs mounted to the lower cross beam, and critical subsystems. These are the propulsion system, that is made up of five Advanced Space

Engines (ASE), RCS, GN&C, communication & data handling, power, and thermal control. Table 3.2-1 provides the core vehicle mass properties breakdown, including these systems.

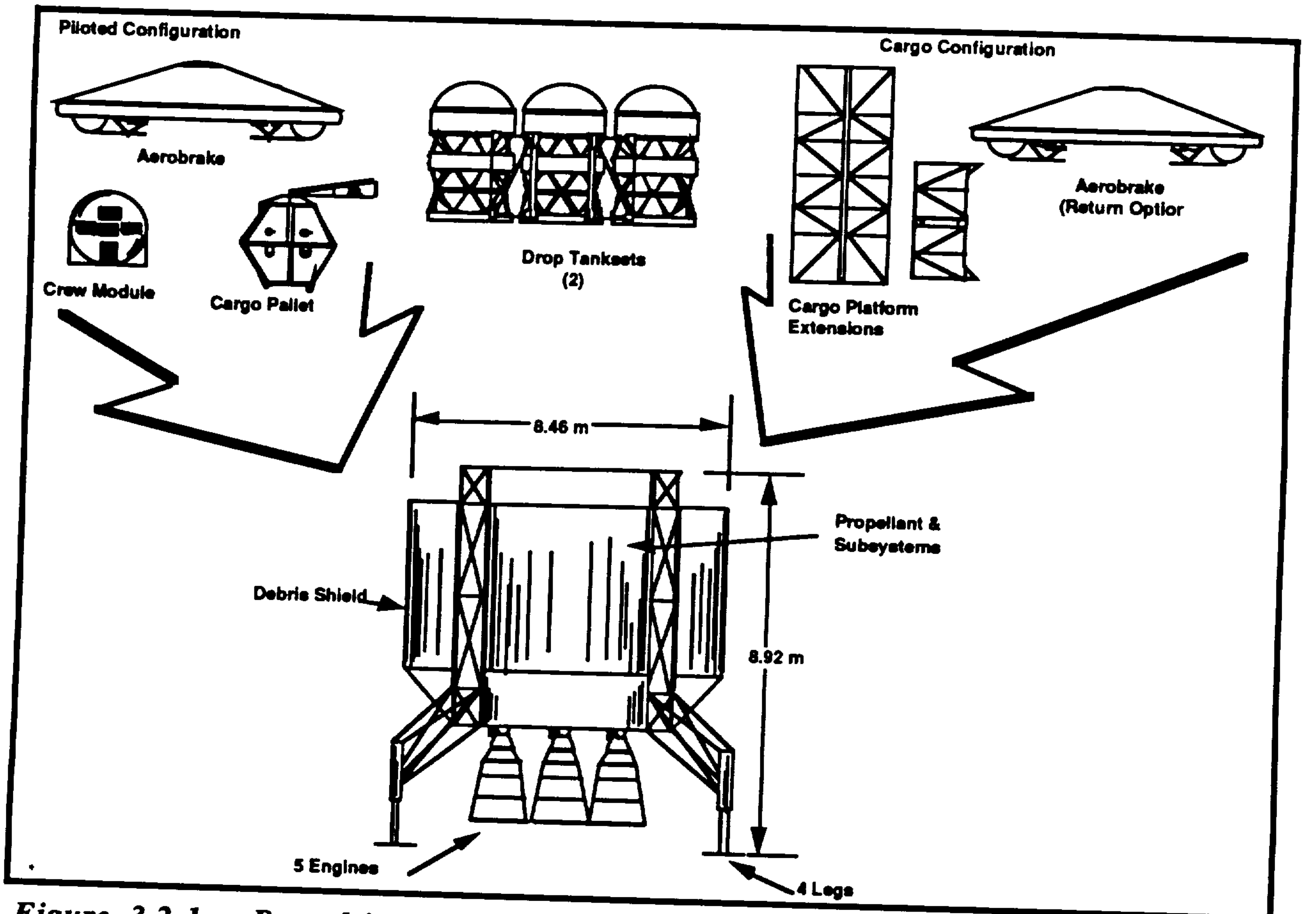


Figure 3.2-1: Propulsion/Avionics Core Module

Table 3.2-1 Mass Properties Breakdown - Core Vehicle

	DESCRIPTION	MASS	MASS
	CORE VEHICLE SUMMARY	KG	M.TONS
02	STRUCTURE	2363.15	2.36
03	PROPELLANT TANKS	802.86	0.80
04	PROPULSION SYSTEM	380.34	0.38
05	MAIN ENGINES	1150.11	1.15
06	RCS SYSTEM	122.45	0.12
07	G. N. & C.	195.46	0.20
08	COMMUNICATION & DATA HNDLG	242.70	0.24
09	ELECTRICAL POWER	444.22	0.44
10	THERMAL CONTROL SYSTEM	553.47	0.55
11	AEROBRAKE	0.00	0.00
19	GROWTH	938.21	0.94
	DRY WEIGHT	7192.97	7.19

Structure—The following section deals with the structural elements of the propulsion/avionics core. The elements include the airframe, the core and drop tank sizes, material and mass, the meteoroid and debris shielding, and the general arrangement of the equipment located in the core. The meteoroid and debris shielding sizing requirements are discussed in another section of the report. The propulsion/avionics core primary structure is composed of graphite epoxy square tubing with aluminum end fittings forming two trusses consisting of a lower and upper box beam and the connecting longitudinal members. The lower cross beam is the thrust frame, equipment mount and support structure for the landing legs. The upper cross beam supports the cargo platform, crew module and payloads. The secondary structural members are graphite epoxy round tubing with aluminum end fittings. They tie the two trusses together and form the mounting braces for the four LH_2 tanks. Figure 3.2-2 gives an overview of the major core structure.

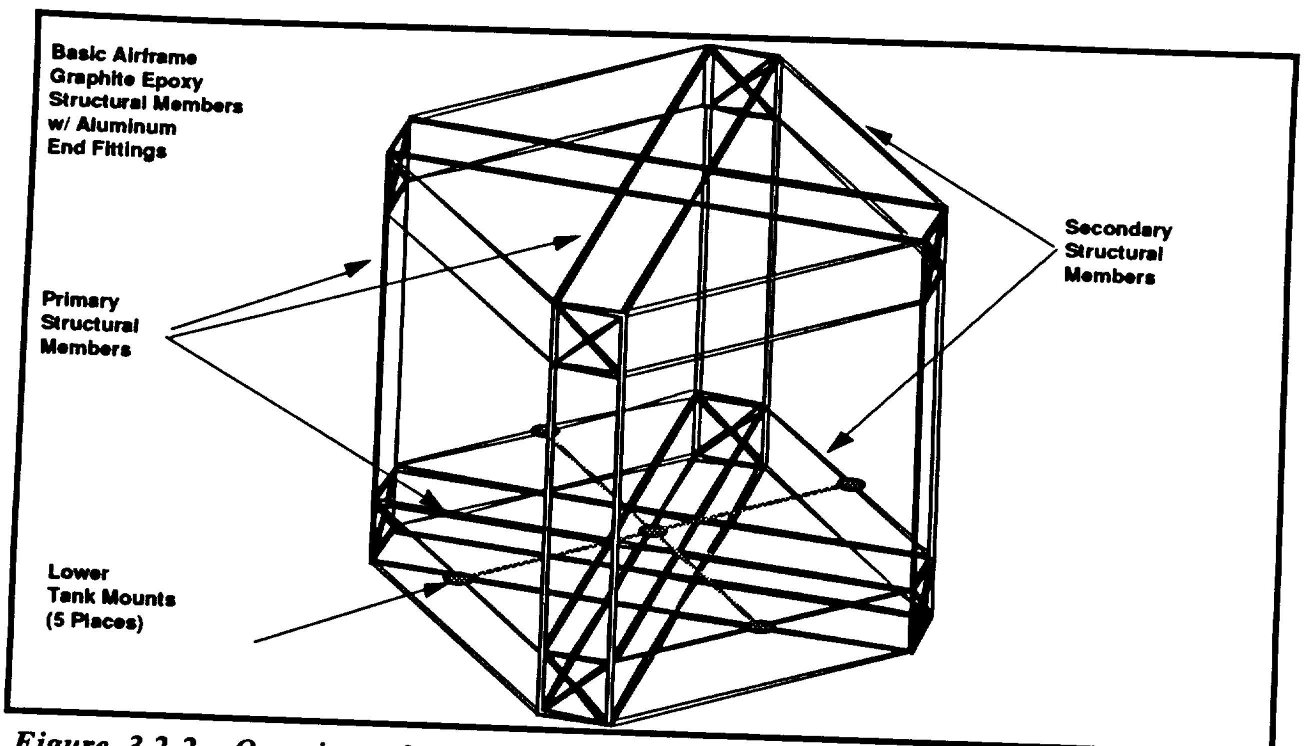


Figure 3.2-2 Overview of the Major Core Structure.

Core Tanks—The isometric view of the propulsion/avionics core shown in Figure 3.2-3, locates the five core tanks - 4 LH_2 tanks and 1 LO_2 tank. The spacing between the tanks and the structure is used for packaging the subsystem components. Graphite polyimide debris shields are attached to the four sides of the core structure to provide micrometeoroid and debris protection for the tanks. The details of the propulsion/avionics core tanks are shown in Figure 3.2-3. The four LH_2 tanks, composed of aluminum-lithium spun domes and isogrid barrel panels to conserve weight, are spaced symmetrically around a center LO_2 tank and mounted to the upper and lower cross beams of the core structure. The LO_2 tank is 4.4 m in length and 2.9 m in diameter, and the LH_2 tanks are 4.2 m long and 2.6 m diameter. Combined, these tanks represent a total propellant capacity of 32.5 tonnes.

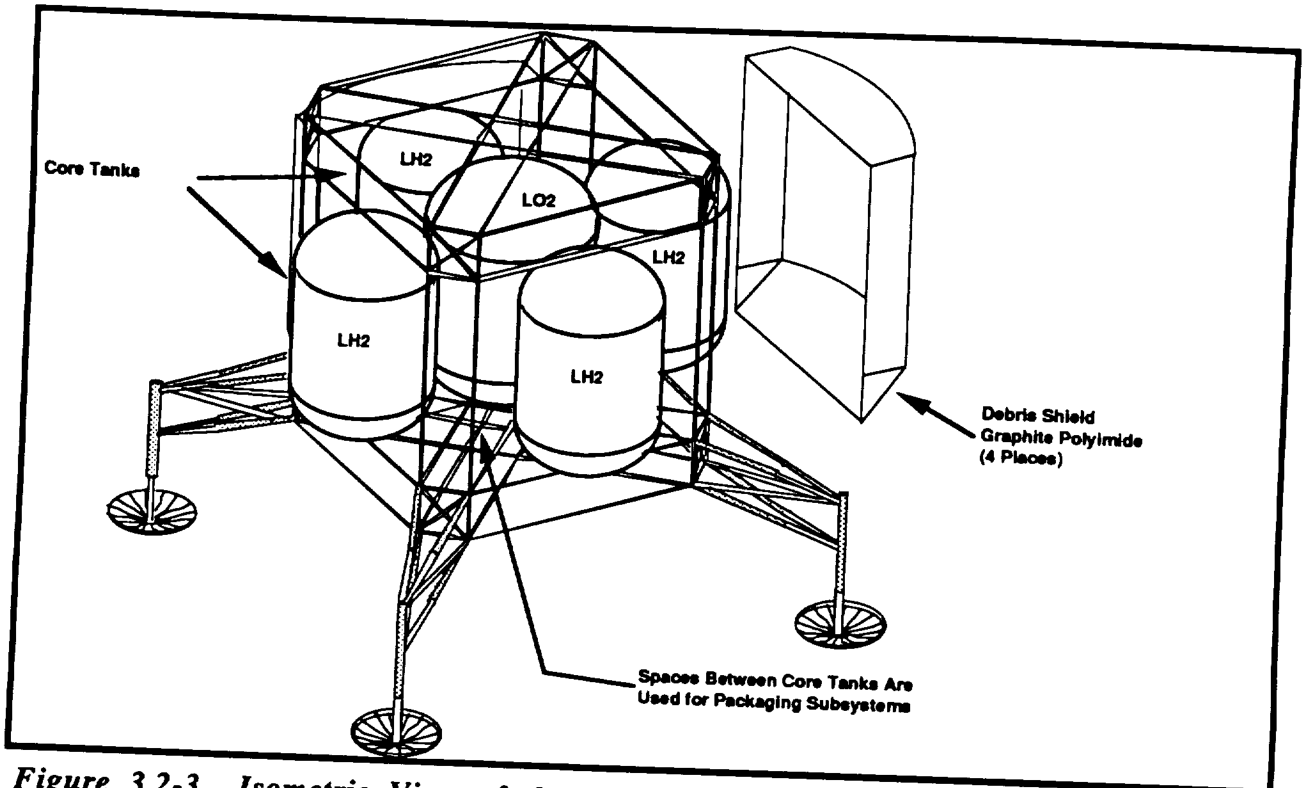


Figure 3.2-3 Isometric View of the Propulsion/Avionics Core

Equipment Layout—Figure 3.2-4 shows the packaging arrangement of the propulsion/avionics core equipment. The placement and size of the propellant tanks allow the subsystem equipment to be packaged in spaces created between the trusses and the tanks. The various tanks for potable water, helium, GO2, and GH2 are packaged in two of the four bays with the fuel cells occupying the other two. The avionics equipment bays are located in the space formed by the upper cross beams. This equipment is packaged around the top and sides of the vehicle to provide access. Leg deployment mechanisms are placed in the lower portion of the core structure and docking ports are provided in the top of the core.

Drop Tanks—The LTS carries two tank arrangements, one on each side of the vehicle, each consisting of three drop tanksets (2 TLI and 1 LOI). Figure 3.2-5 shows the details of a typical tank arrangement. The two TLI tanksets attach to the center LOI tankset using struts with end fittings using clip-in locking pins. The LOI tankset is directly mounted to the core structure using a

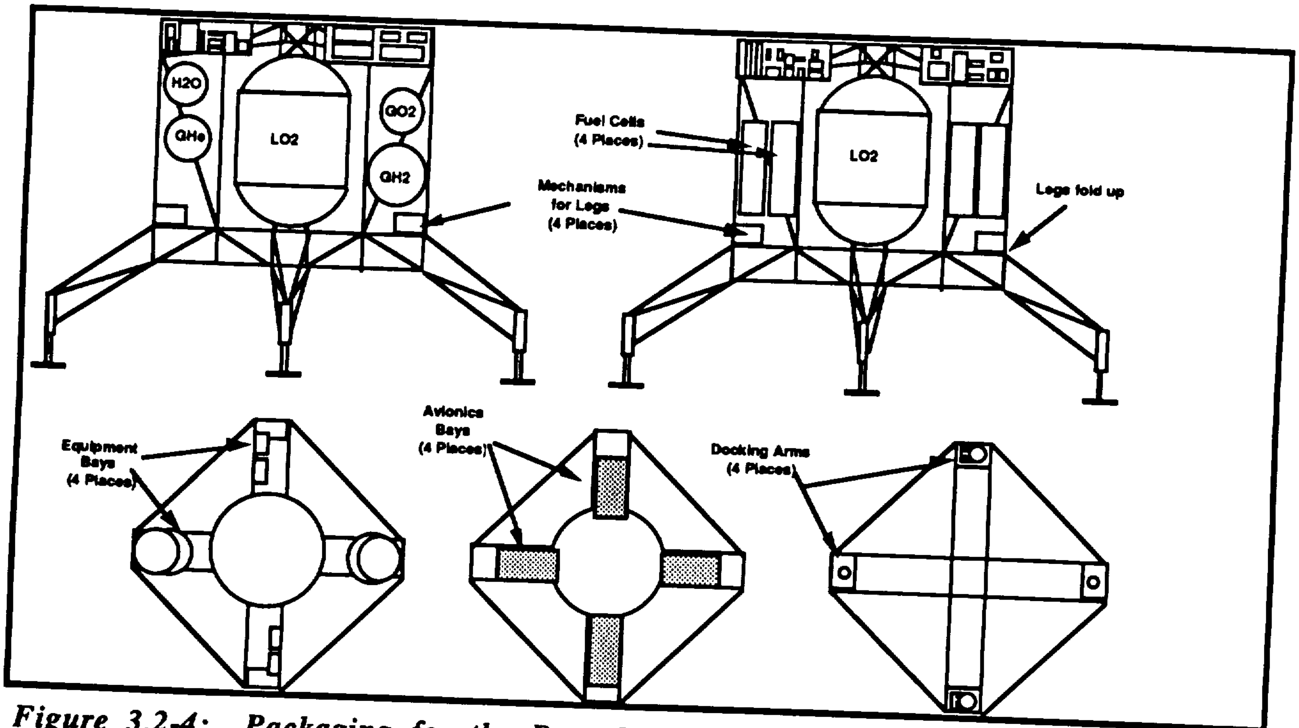


Figure 3.2-4: Packaging for the Propulsion/Avionics Core Equipment

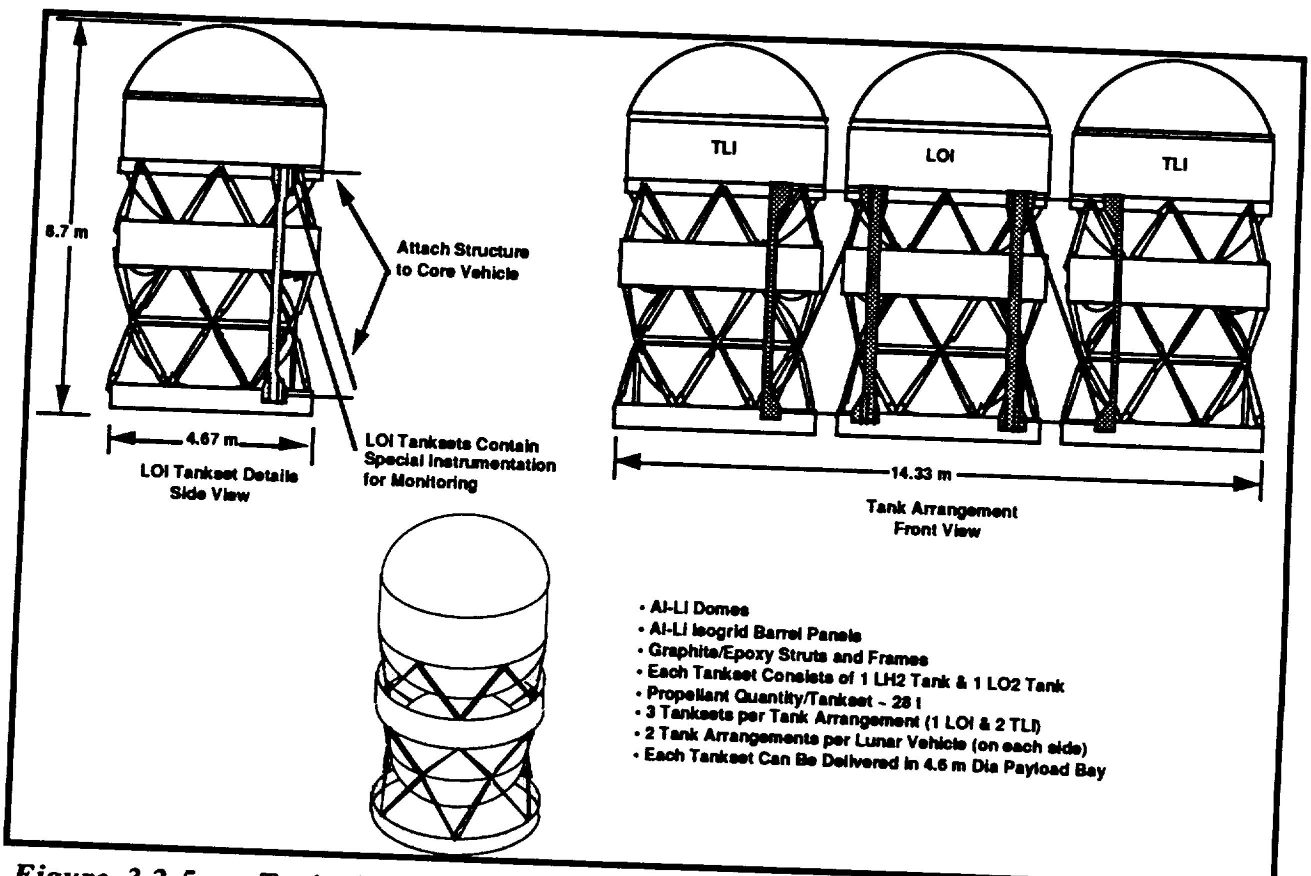


Figure 3.2-5: Typical Tank Arrangement Details

similar strut and end fitting arrangement. The propellant capacity of an individual drop tankset consisting of 1 LH2 tank and 1 LO2 tank is approximately 28 tonnes, or 84 tonnes when combined into a set of three tanks. The positioning of the TLI tanksets allows them to be separated after the TLI burn leaving the LOI tankset with the vehicle until LLO insertion, where they are then released. Tanks are constructed of aluminum-lithium domes and isogrid barrel panels. The tanks are connected by graphite-epoxy struts and frames, and fit within a 4.6 m (15 ft) payload shroud. For ground heat leaks and on-orbit thermal protection, tanksets have spray-on-foam-insulation (SOFI) and multi-layer insulation (MLI). A helium pressurization system and instrumentation are integrated into each tankset.

Propulsion System

This section describes the propulsion/avionics core propulsion subsystem which consists of the main engine system, RCS system, a propellant management system, propellant tanks and their associated feed lines.

Engines - The layout of the main propulsion engines is shown in Figure 3.2-6. Five advanced space engines are mounted to the lower cross beams of the core, spaced 2 meters from center to center of engines, with a nozzle exit diameter of 1.34 meters. This spacing pattern accommodates a gimbal range of ± 8° except for the center engine which is not required to gimbal. Electrical mechanical actuators are used to drive the gimbaling action.

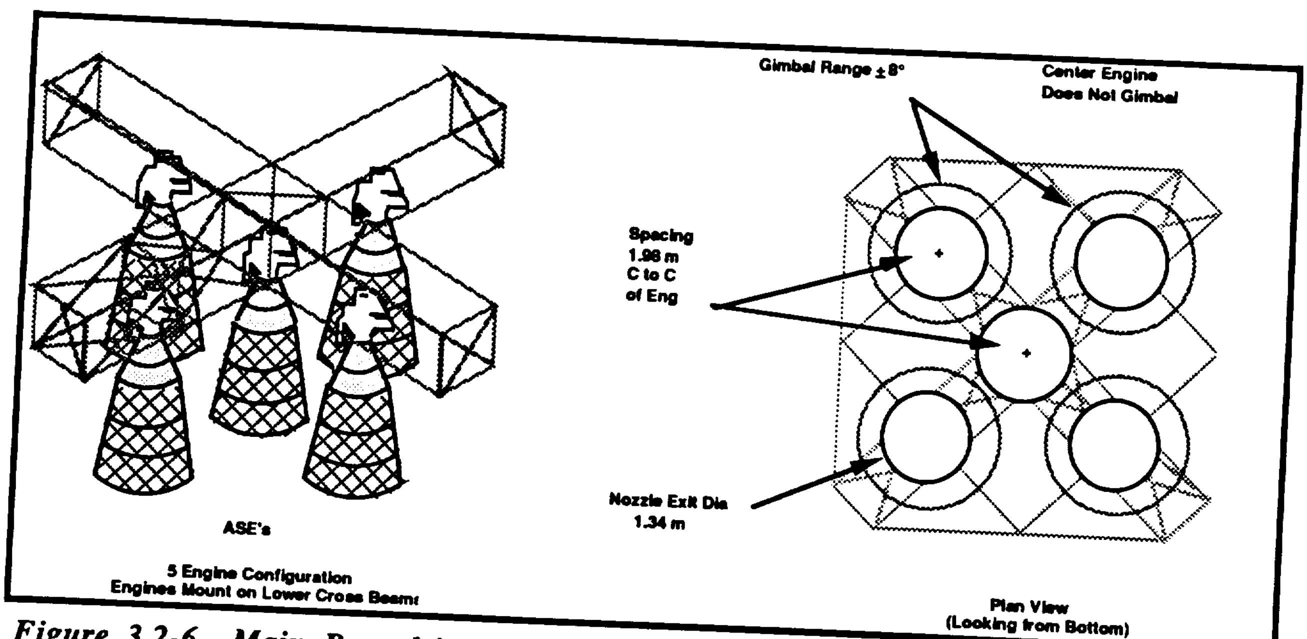


Figure 3.2-6 Main Propulsion Engine Layout

Attachment of the engines to the core occurs through vehicle/engine carrier plate quick disconnects, allowing easy change out during surface or in-space maintenance. The vehicle carrier plates are incorporated into the lower portion of the box beam engine support. The engine is assembled onto an engine carrier plate including all of the engine interfaces, which is then mated with the vehicle carrier plate disconnects, as shown in Figure 3.2-7. Additional details of the engine carrier plate are shown in Figure 3.2-8. The disconnects penetrate the vehicle carrier plate and lock into place

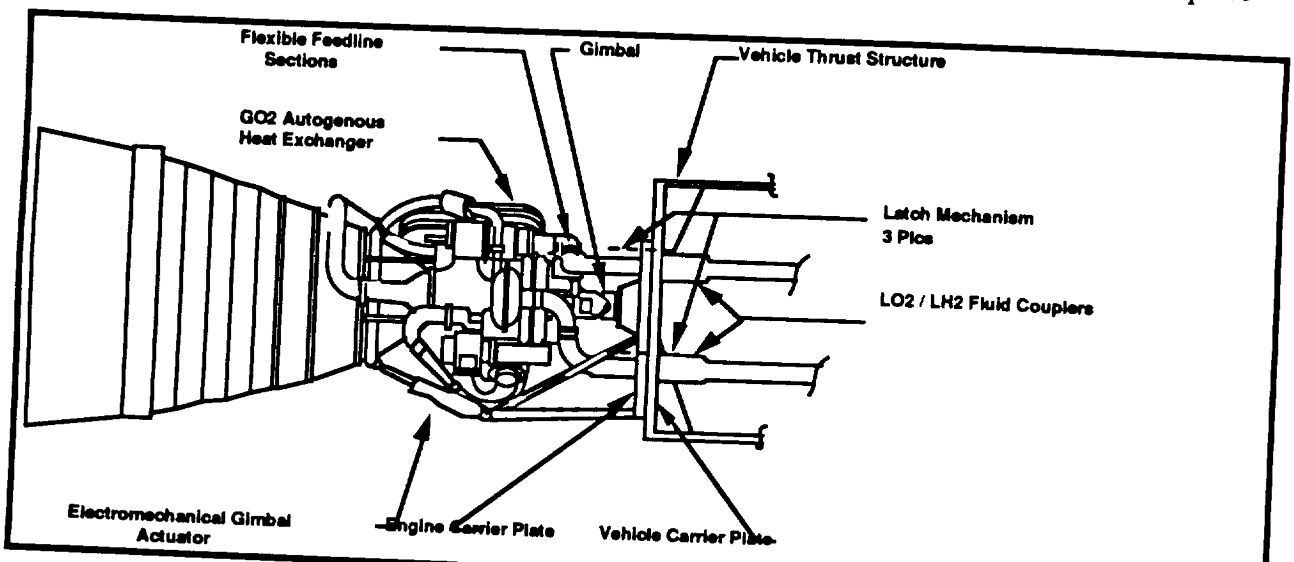


Figure 3.2-7 Engine Replacement

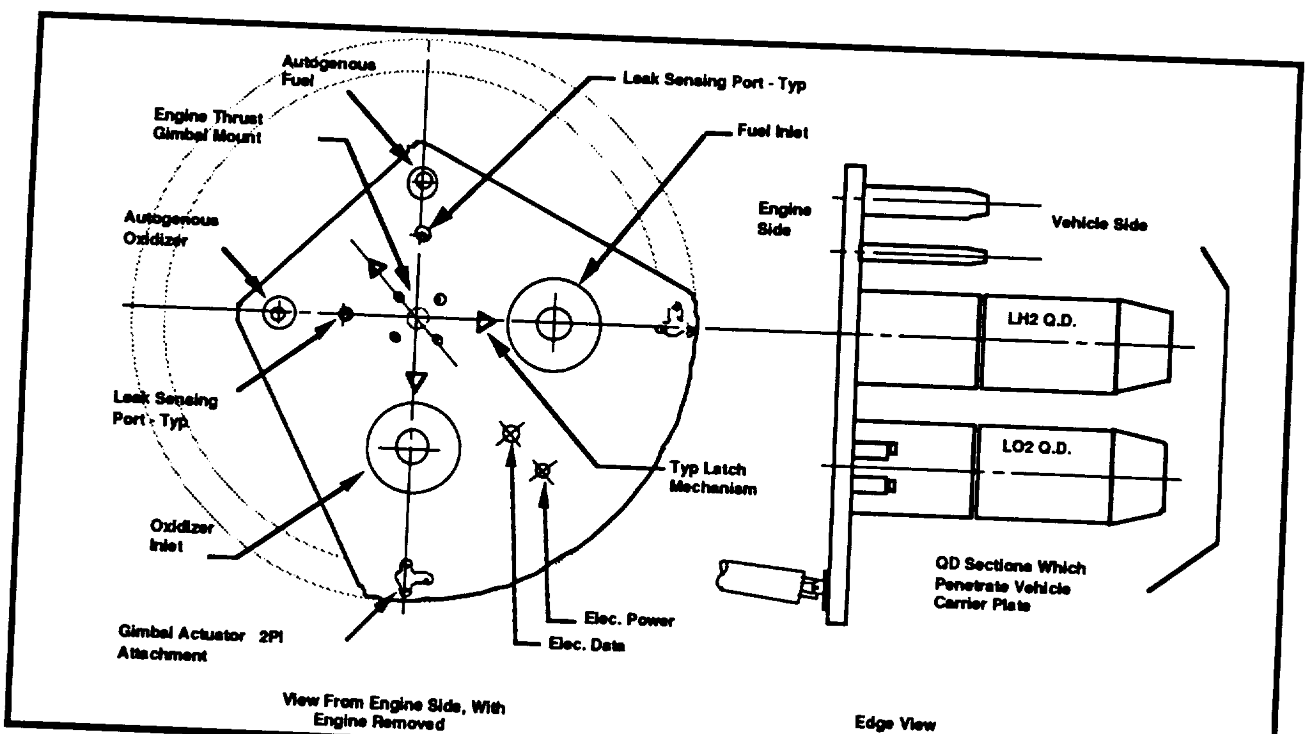


Figure 3.2-8 Engine Carrier Plate

to complete installation of the engine. A common engine interface approach was used to allow different engine versions to be installed as upgrades are made or for the tailoring of the engine configuration to specific missions.

Reaction Control System (RCS) - The LTS RCS thrusters consist of two separate systems as shown in Figure 3.2-9, one located on the propulsion/avionics core and the other on the aerobrake. Six degrees of freedom, with redundancy, are provided for each vehicle by its 24 thrusters. The RCS system is self contained on the core, totally separate from the cargo and crew module. Variable thrust levels are used to accommodate the wide variation in vehicle mass during a mission. The thrusters at the upper end of the core vehicle are inactive when the vehicle is fully assembled.

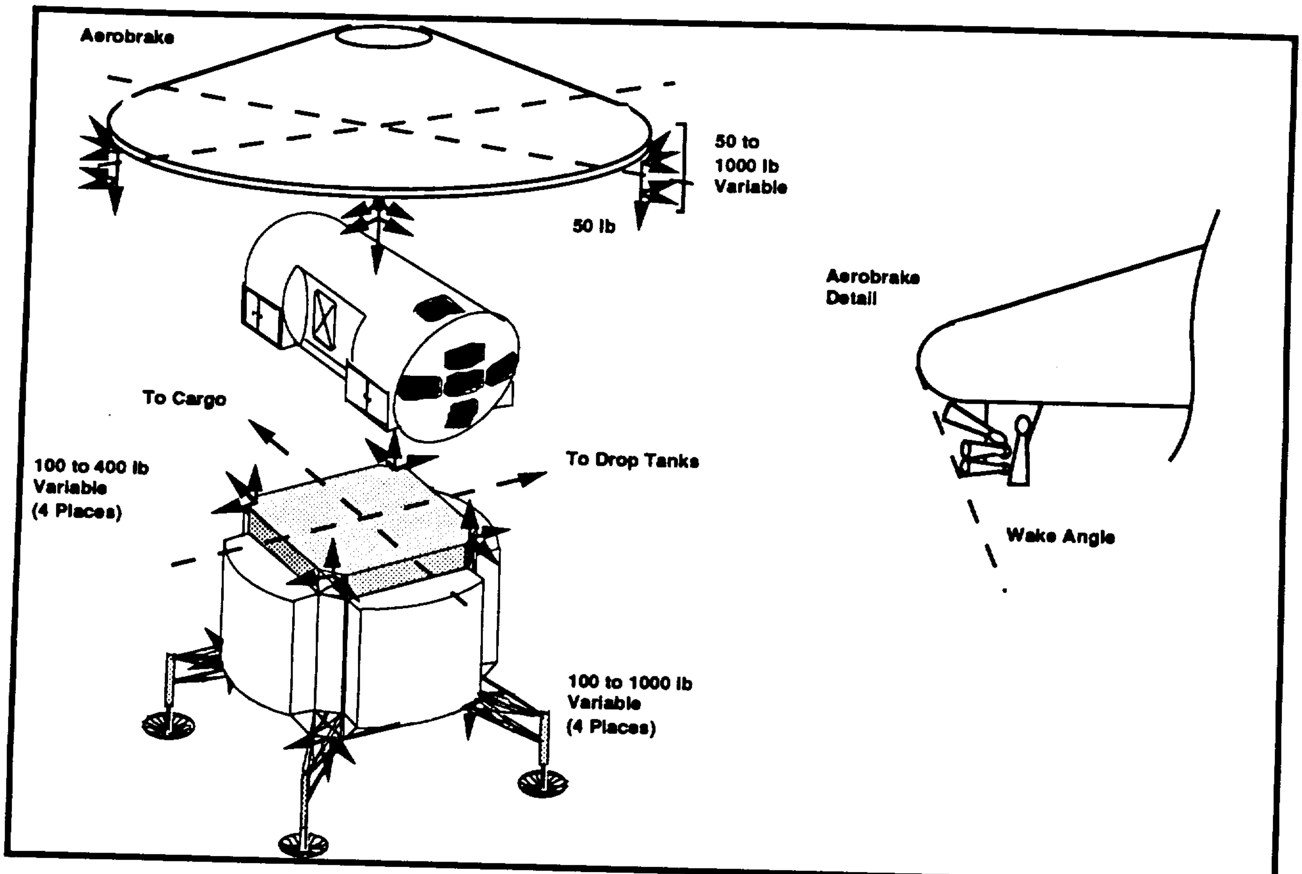


Figure 3.2-9 RCS Thruster Arrangement

Drop Tank Feed Lines and Disconnect - Feed lines connect the two TLI tanksets (both LO2 and LH2) through an umbilical to the LOI tankset that then merges at an umbilical connection to the core tanks. When the TLI tanksets are separated after TLI burn, the propellant disconnect is made at this TLI/LOI umbilical, with the LOI disconnect made at the LOI/core tank umbilical. Figure 3.2-10 depicts a typical fluid schematic for each of the tanksets.

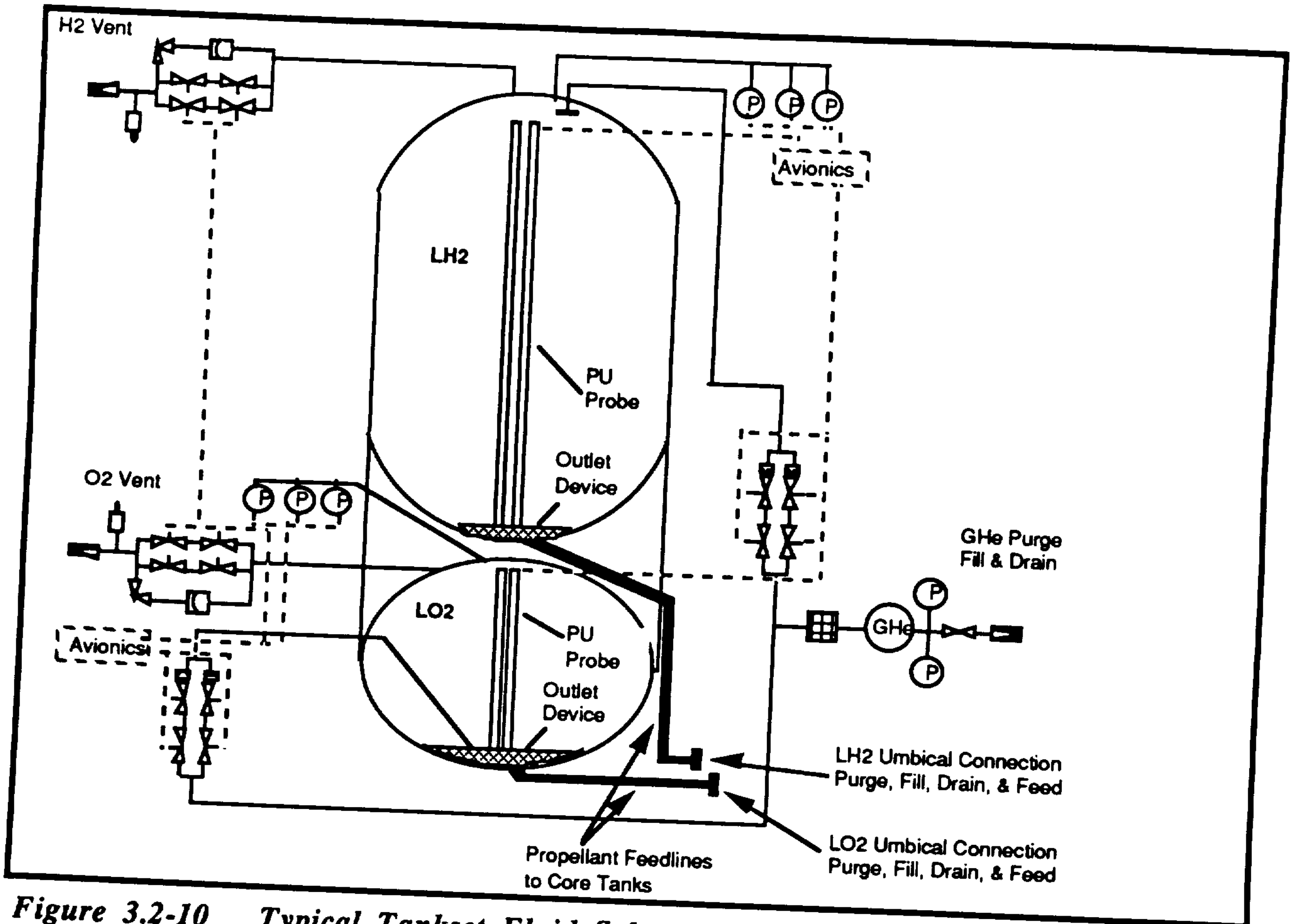


Figure 3.2-10 Typical Tankset Fluid Schematic

Core Tank Feed Lines - Propellant is fed from the drop tanksets to the core tanks through the LOI/core tank umbilical, with the two core LH2 tanks fed by one of the LOI tanks. Each LH2 core tank then feeds a manifold with separate feed lines to each individual engine.

Return Tank Feed Lines - Figure 3.2-11 illustrates the flow of propellant from the return tanks in the aerobrake to the core engines. After the core has performed the rendezvous and dock with the aerobrake, umbilical connections are made at two locations (180° opposite each other) from which separate LO2 and LH2 lines are routed along the core structure

Avionics—The LTS avionics, located in the aerobrake, crew module and the propulsion/avionics core, represents a man rated quad redundant system. The avionics system, located in the propulsion/avionics core, handles all cargo operation functions and interfaces with those elements in the crew module during the piloted operations. This system is composed of two major groups, Guidance, Navigation and Control (GN&C) and Communication and Data Management (C&D Mngmt). Tables 3.2-2 & 3.2-3 summarize the components, their quantities, and total mass.

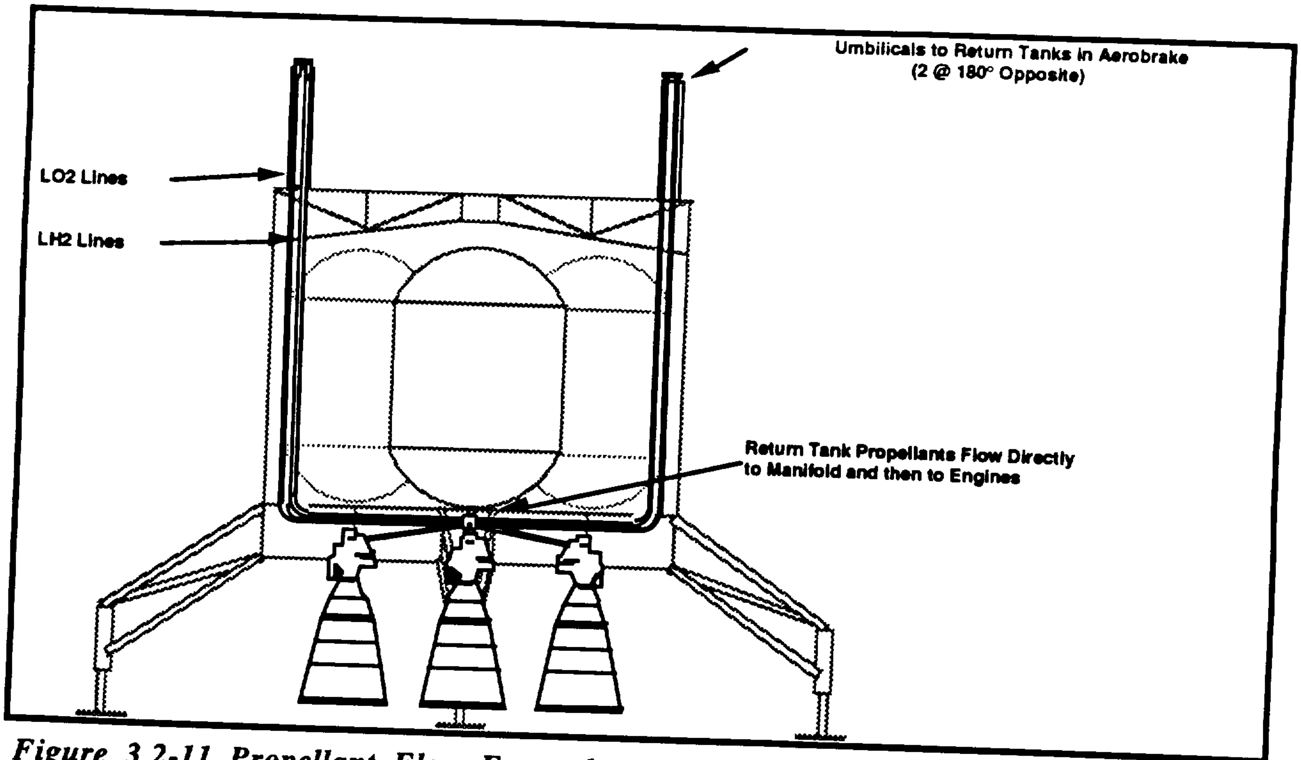


Figure 3.2-11 Propellant Flow From the Aerobrake Return Tanks

Table 3.2-2 Guidance, Navigation, & Control

Components	Units	WT	Total
IMU(3 RLG & 3 PMA)	2.00	24.00	48.00
GPS Receiver	2.00	20.00	40.00
GPS Antenna - High	2.00	5.00	10.00
GPS Antenna - Low	1.00	5.00	5.00
EMA Controller	2.00	10.00	20.00
RCS VDA	32.00	0.50	16.00
Guidance & Control Total			139.00
Star Scanner	4.00	6.00	24.00
Navigation Total			24.00
Landing Radar Altimeter	2.00	25.00	50.00
Rendezvous Radar	2.00	25.00	50.00
Landing Radar Electronics	2.00	49.00	98.00
Lander Antenna	2.00	5.00	10.00
Landing & Rendezvous System			208.00
Pan Tilt Cameras	2.00	15.00	30.00
Video Recorders	2.00	15.00	30.00
TV System			60.00
GN&C Core Total			431.00

Table 3.2-3 Communication and Data Management

Components	Units	WT	Total
GPS Antenna System	2.00	15.00	30.00
STDN/TDRS Transponder	2.00	15.50	31.00
20W R.F. Power Amp	2.00	6.00	12.00
S-Band R.F. System	2.00	50.00	100.00
UHF Antenna	2.00	10.00	20.00
UHF System	2.00	10.00	20.00
TLM Power Supply	2.00	7.00	14.00
Enclosure Box	1.00	26.55	26.55
Communication			253.55
GN&C Computer	4.00	20.00	80.00
Master Timing Units	2.00	5.00	10.00
Health & Status Computer	4.00	20.00	80.00
TM System	2.00	22.00	44.00
GN &C IU	4.00	10.50	42.00
Enclosure Box	1.00	25.50	25.50
Data Management			281.50
C&DM Core Total			535.05

Power—Power for the propulsion/avionics core is provided by four fuel cells similar to those aboard STS, but supplied with propellant grade cryogenic reactants through molecular sieves. Each fuel cell delivers 12 kw at peak (27.5 V and 436 A) and an average output of 7 kw. 2 kw provides 32.5 V and 61.5 A. The water supplied as a by-product of the fuel cells provides potable water during the mission. Emergency power is provided by Ag-Zn batteries. Table 3.2-4 summarizes the power supply components, their quantities, and total mass

Table 3.2-4 Power System - P/A Core

Power System - P/A Core	Qty	Unit Wt lbs	Total
Fuel Cell System	4	86.25	345.00
Radiator System	4	28.75	115.00
Residual H2O System	2	17.25	34.50
Batteries	2	100.00	200.00
Power BUS	4	10.50	42.00
Power Distribution Equipment	4	27.00	108.00
Wiring,Harness, & Connectors	1	100.00	100.00
Enclosure Box	1	15.00	15.00
Total			959.50

Meteoroid & Debris Protection—A meteoroid and debris protection analysis was conducted to determine the best type of protection needed at LEO, at the lunar surface, and at the hanger at SSF, for the environments to which the STV elements were exposed. Figure 3.2-12 shows the flux and particle size differences encountered at each stage of a mission. Since the penetration resistance varies with velocity, density and obliquity, the reliability given by Probability of No Penetration (PNI), has been defined as a reference point to estimate shielding requirements.

$$\text{Probability of No Impact (PNI)} = \exp(-\text{Flux x Area x Time}) = e^{-(N \cdot A \cdot T)}$$
$$\text{If "}N \cdot A \cdot T\text{" is small (reliability is high), then PNP} = 1 - N \cdot A \cdot T.$$

The figure defines the particle environment and the critical flux for 0.09955 PNI for key mission phases. The PNP (which covers the entire velocity and obliquity spectrum) for STV elements as well as the threat must be higher than 0.9955 if the overall reliability from impact is to be 0.9955. The shielding recommended for all STV elements accounts for these preliminary estimates.

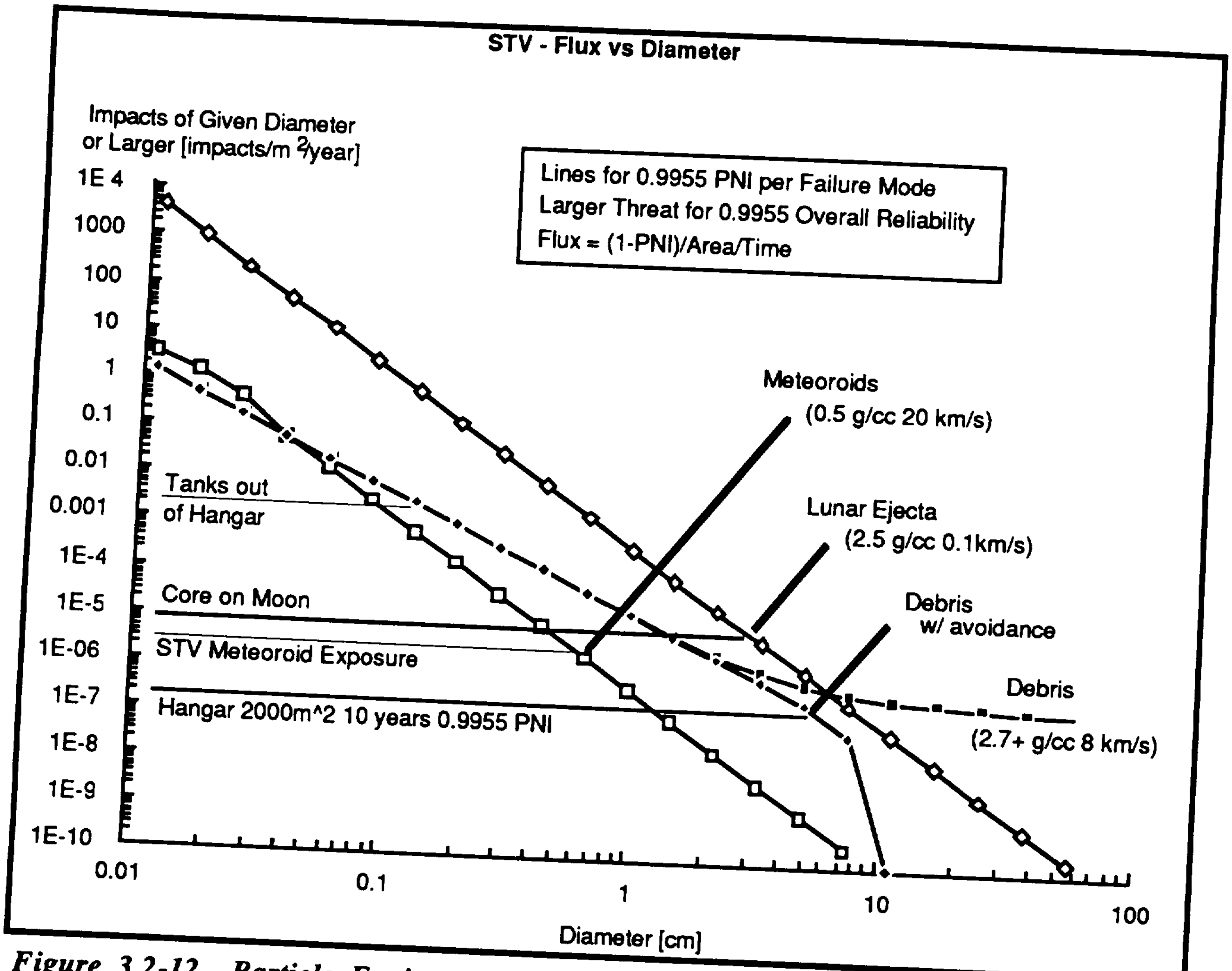


Figure 3.2-12 Particle Environment vs Critical Flux

Shielding recommendations were generated for protection from space debris, meteoroids, and lunar ejecta. The entire threat spectrum was addressed, including particle size, impact velocities and obliquity versus the performance of optimized multilayer shield designs. Table 3.2-5 provides a method of estimating shield thickness and spacing as a function of estimated particle size. Multi-wall shields are not as effective at 3 km/s or for 45° obliquity impacts as they are for normal impacts at 7 km/s since the debris particle does not fragment as well, therefore the total weight of the shield increases to account for the non-optimum performance. The design of the hangar shield uses multi-wall designs developed under Martin Marietta IR&D, and under contracts from NASA and the U.S. Air Force Defensive Shields Program. The lunar ejecta shield thickness estimate is preliminary at this time with additional data to be provided as they become available. Composites or ballistic cloth may be much more effective in stopping that velocity of a particle than the estimated weight of monolithic aluminum.

Table 3.2-5 Shielding Requirements As a Function of Particle Size

STV - Shield Requirements

• Areal Density of Shield is Proportional to Diameter of Impacting Particle

	Equivalent Total Thickness of Aluminum	Areal Density kg/m^2 (D in cm)	Minimum Bumper Standoff
Space Debris	0.75 D	20 D	20 D
Meteoroids	0.15 D	4 D	10 D
Lunar Ejecta	0.15 D	4 D	Not Sensitive

• Total Shield Thickness and Density includes TPS and Rear Wall

• Optimum Designs may Require Multiple Layers or Geometric Disruptors (developed on IRAD, NASA, and Air Force/Defensive Shields Programs)

• Debris Shield Thickness Accounts for Reduced Resistance to Oblique (45°) and High Velocities (16 km/s) or Low Velocities (3 km/s)

Aerobrake—The aerobrake provides the thermal protection for the LTS during the aeropass maneuver before returning to SSF. Studies have determined that the aerobrake design provides a sizable savings in propellant, directly translating into a cost savings. The study analyzed different types of aerobrake construction and recommended a rigid, hard shell design. Analysis of on-orbit assembly determined that a minimum number of pieces requiring assembly was desirable, which resulted in the three piece folding concept. The manifesting of the folding aerobrake in the ETO launch vehicle was considered and found to be compatible with a 7.6 m payload envelope. An isometric view of this rigid aerobrake structure is shown in Figure 3.2-13.

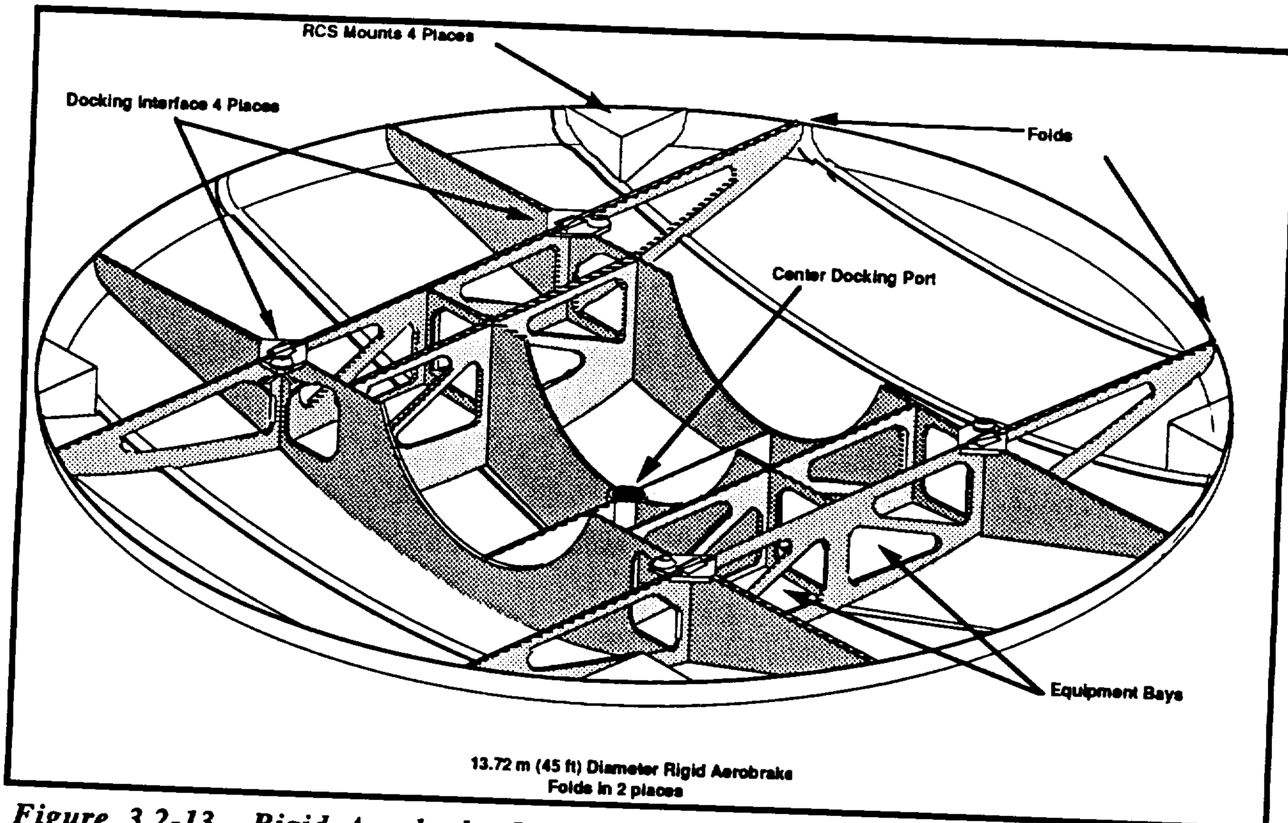


Figure 3.2-13 Rigid Aerobrake Isometric

Operation of the lunar mission requires the aerobrake and the lander to separate in LLO before the lander makes the lunar descent, leaving the aerobrake in a 60 x 100 nm orbit. This requires that the aerobrake have station keeping, rendezvous, and docking capabilities. This is accomplished by the aerobrake converting from a passive element to an active vehicle using its own avionics, power, and RCS subsystems to control. The following sections detail the structural elements and the subsystems associated with the aerobrake

Structure - The aerobrake is a graphite-polyimide structure with overall dimensions of 13.72 m in diameter and 2.59 m in depth, covered with shuttle type ceramic tiles (FRICS-20). Two major longitudinal and three major transverse bulkheads provide the primary structural elements, with additional frames and intermediate bulkheads for support. The bulkheads are fabricated from graphite-polyimide face sheets and a foam core and the frames are extruded graphite epoxy "T"-sections. The surface panels are formed from graphite-polyimide face sheets with an aluminum honeycomb core. The center section panels are 0.51 cm thick and the outer panels are 0.38 cm thick and are mounted to the surface panels extruded graphite epoxy angles.

LEO assembly of the aerobrake is performed by rotating the two outer sections into place about hinges located at the intersection of the longitudinal and outer transverse bulkheads. Proper alignment to the center section is assured by a male/female aluminum joint along the intersecting surface panels. The outer section is then secured into place through the use of locking pins located on the outboard side of the longitudinal bulkheads. A section of the outer ceramic tile around the interface area is initially not installed to allow the hinged motion required for deployment. Once the side sections are deployed, the ceramic tile must be installed on orbit over the interface area.

Subsystems - The aerobrake is left in a 60 to 100 nm orbit when the lander separates for descent to the lunar surface. In order for the aerobrake to maintain its position and be able to rendezvous and dock with the lander for the return trip, it was to be outfitted with the components shown in Figure 3.3-14. Avionics bays and equipment bays are located along either side of the longitudinal bulkhead. The docking equipment is located on the central bulkhead and at the intersection of outer transverse bulkheads and the intermediate longitudinal bulkheads. The aerobrake also houses the return propellant for the lander. This is located in two tank pallets consisting of 3 LH2 tanks and 2 LO2 tank in each pallet. The pallets are positioned in the outer sections of the aerobrake leaving the center section free for mating the lander and crew module to the aerobrake.

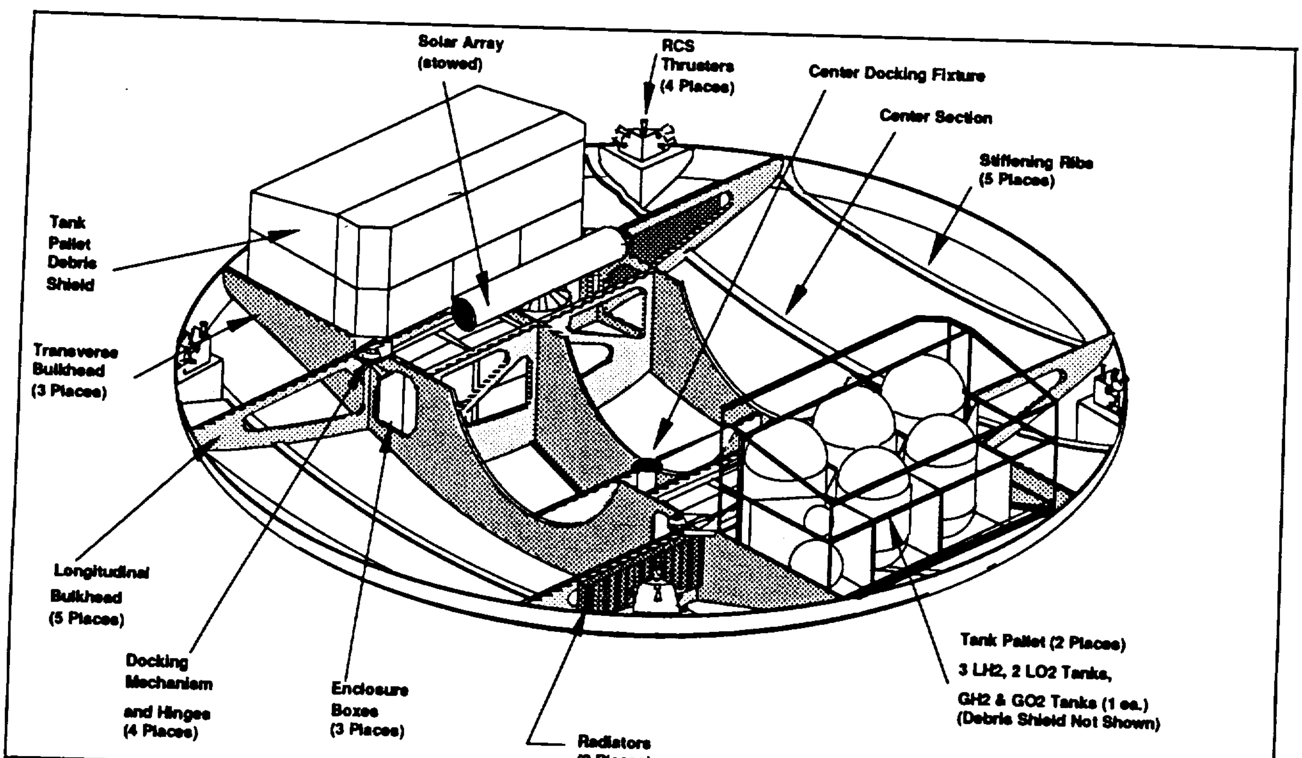
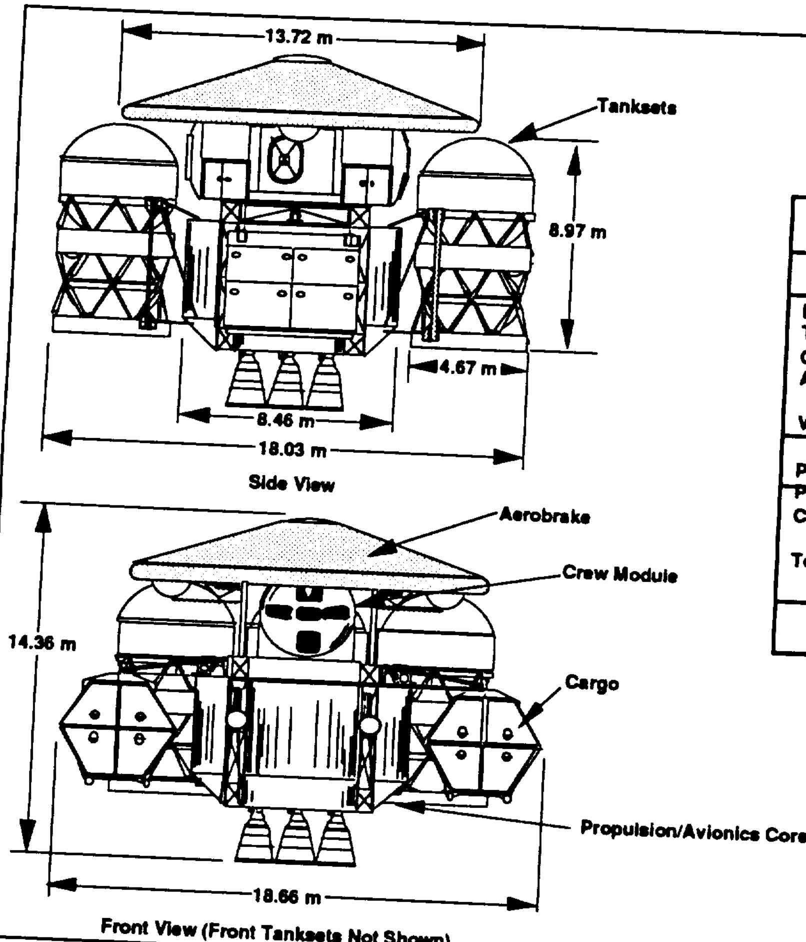


Figure 3.2-14 Avionics/Aerobrake Equipment Relationship

3.3 Piloted Configuration

This section deals with those components unique to the piloted configuration and some of the mission operations. The STV piloted configuration is designed to carry a crew of four and 14.6 mt of cargo using 174 mt of propellant between the various tanks. The vehicle's overall dimensions are 14.36 m by 18.66 m by 18.03 m (Figure 3.3-1) when fully assembled and ready to leave from LEO. The piloted vehicle consists of a crew module, cargo modules and support structure, the two drop tanksets (three tanks per side), and an aerobrake with its associated equipment mounted to the propulsion/avionics core module.



Mass Properties	
Components	Mass (t)
Prop/Avionics Core	7.1
Tanksets (4 TLI & 2 LOI)	9.1
Crew Module	7.7
Aerobrake & Equip	3.5
Vehicle Dry Mass	27.5
Propellant	174.
Personnel/Misc	.6
Cargo w/Sppt	15.2
Total Mass	217.5

Figure 3.3-1 STV Piloted Configuration Dimensional Detail

Crew Module—The crew module is required to support a crew of four during the five to six day trans-lunar and trans-Earth flight and support the crew for the first 48 hours on the lunar surface. Some of the general structural and accommodations requirements for the crew module are:

a) Designed for 5 g loading

b) Two hatches to be provided
c) Capable of berthing to SSF
d) Must fit within the aerobrake wake
e) Meteoroid shield to be used
f) Checkout, repair, and resupply is done at SSF
g) ALSPE shelter to be provided
h) Allow for 2 repressurizations
i) At least 6 cubic meters per person of habitable volume
j) Stored oxygen with regenerable molecular-sieve bed CO2 removal
k) 14.7 psi for normal operations
l) 1.8 kg of food and 2.0 kg of water per man per day
m) Avionics and power interfaces with core module

The general description of the crew module (Figure 3.3-2) is approximately 72 cubic meters in volume and 8.54 m long by 3.67 m in diameter. The crew module is mounted to the propulsion/avionics core with trunnion and keel fittings similar to those used on the STS system. The module is divided into three major sections, the forward section which houses the flight deck, the mid section which serves as EMU storage, storm shelter, and lunar egress, and the aft section which houses the waste management system, the food preparation system, and station berthing. The crew module can also be utilized at SSF as an additional work station and can be utilized on the lunar surface as a remote habitat and/or safe haven. Unpressurized stowage is located along the exterior sides of the module. A side hatch provides lunar egress and a standard berthing ring/hatch is located on the end for attachment to station. Four windows on the forward end provide viewing during lunar landing, and a top window provides viewing for rendezvous and docking.

Four unpressurized areas are provided to accommodate interface connections, stowage and ECLSS equipment. Two of the bays are designated for the avionics, power, and potable water interfaces between the core module and the crew module. These areas also house the batteries for backup power to the crew module. The other two bays are used to mount the cryogenic oxygen and nitrogen tanks needed for the Life Support System. The advantage of these spaces is that it allows for the outfitting and connecting the crew module to core module without having to enter the crew module during the assembly process. While the vehicle is on the lunar surface the crew is able to checkout the interfaces and avoid entering the crew module.

The interior arrangement of the crew module is straightforward. The forward section houses the flight deck and seats three crewmen. The mid section provides stowage for four EMU's as well as

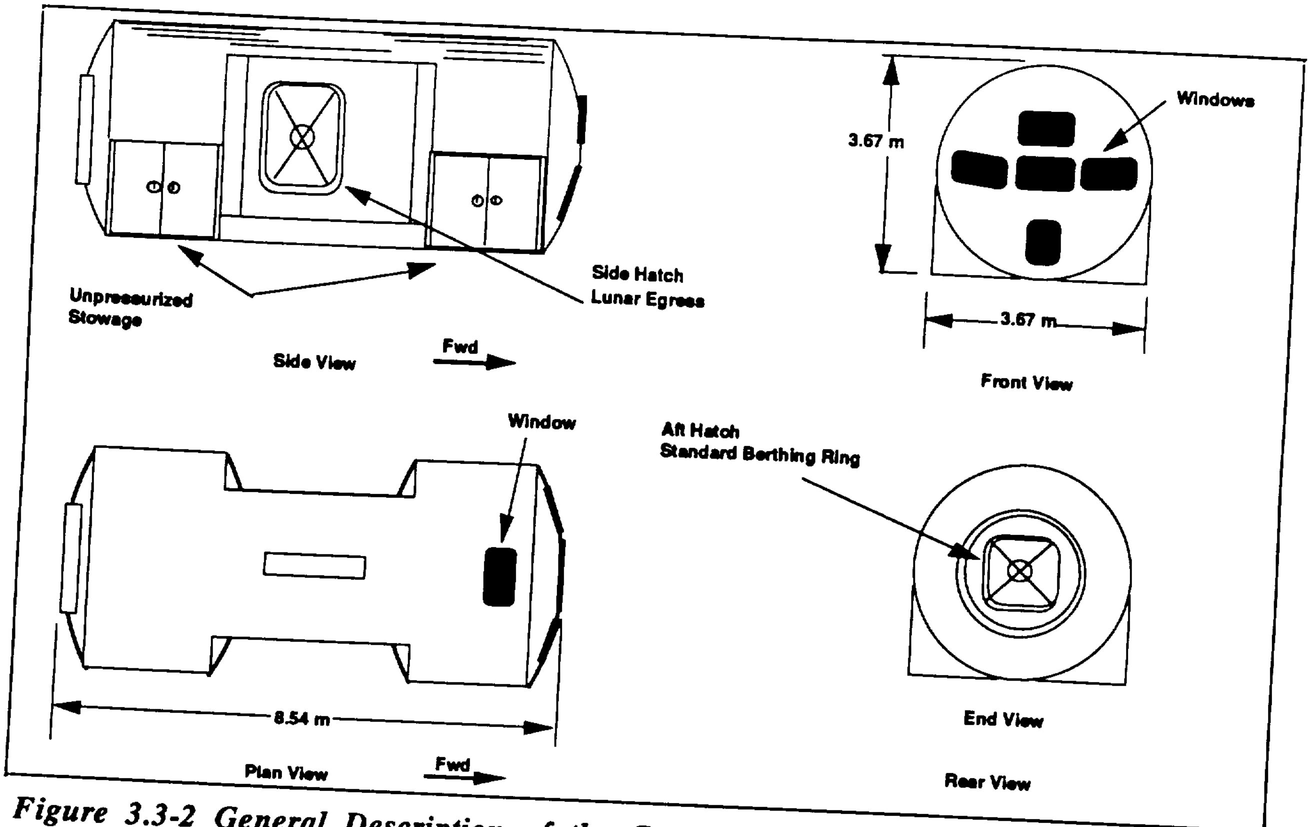


Figure 3.3-2 General Description of the Crew Module

provides lunar egress and storm protection. The aft section houses waste management and the galley and provides seating for one crewman. Equipment bays and internal stowage are located below the floor levels in all three sections. Lightweight, portable, multipositional couches are used for sleep periods, and for body support during ascent, descent and aeropass.

There are five windows which provide viewing for the crew. The four windows located in the forward end of the flight deck provides the pilot and co-pilot with over a 170° field of view angle for landing on the lunar surface. The pilot also has a field of view angle from the horizon to the lunar surface of over 85°. A single window located in the top of the module provides the pilot with a view of the target during rendezvous and docking with the aerobrake in LLO.

When the STV is ready to make the aeropass maneuver, the load forces felt by the crewmen are reversed from the normal acceleration force experienced throughout the mission. The crew would be in the wrong seating position and provisions had to be made to accommodate these load forces on the crew. Reentry couches, similar to those on the Apollo spacecraft, are mounted in the overhead. Prior to beginning the aeropass maneuver, the crewmen would strap themselves into the reentry couches and thus be in the correct position for the aeropass loads. After the aeropass maneuver is completed, the crewmen would return to their normal seating position for circularization and rendezvous with SSF.

In the event that a rescue mission is needed, the crew module can provide space for additional crewmen. Two additional seat/reentry couches would be mounted in the mid section of the crew module. This will provide room for the rescue party, consisting of a pilot and co-pilot , and the four crewmen on the lunar surface to be rescued.

Landing—After LTV has achieved LLO and stabilized its orbit, the crew prepares the vehicle for lunar descent. The aerobrake and the core separate and the core will back away from the aerobrake. The aerobrake will deploy its solar array and assume a solar orientation. The crew then lowers the landing legs and checks to ensure that the legs are locked into place. The RCS thrusters align the vehicle for the decent trajectory angle, and the main engines are fired to brake the vehicle as it descends to the lunar surface. Once the vehicle has landed the crew will checkout all the systems and prepare to disembark and offload the cargo.

Cargo Offloading—Cargo unloading of the piloted vehicle on the lunar surface can be accomplished without the use of the LEVPU. The cargo is supported by cargo supports extending from the sides of the core. Once the vehicle has landed on the lunar surface (Figure 3.3-3), the cargo can be lowered directly to the surface or onto a transporter by using a hoist mounted on the cargo support structure. The cargo on the piloted configuration is supported by cargo supports

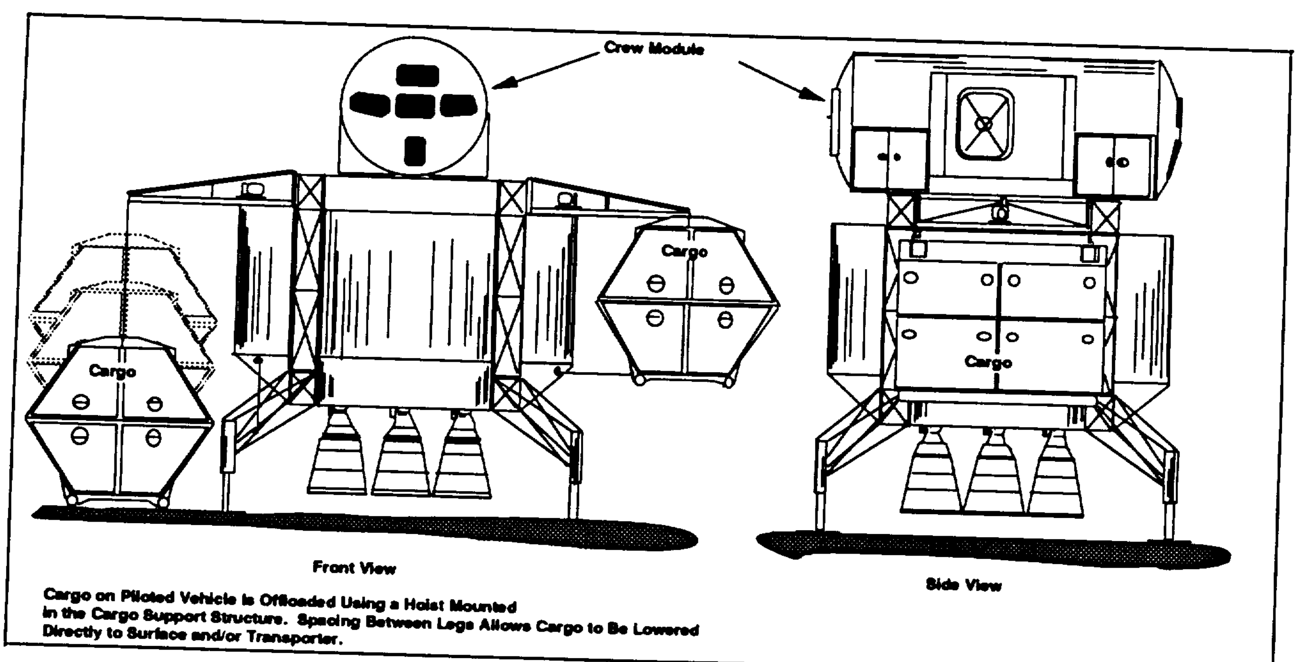


Figure 3.3-3 Piloted Vehicle Unloading Cargo On The Lunar Surface

attached to each side of the core. The hoists located inside the cargo support structure and the spacing between the legs allow the cargo to be lowered directly to the lunar surface. The cargo supports can be retracted or folded to fit within the aeroassist return configuration to allow reuse.

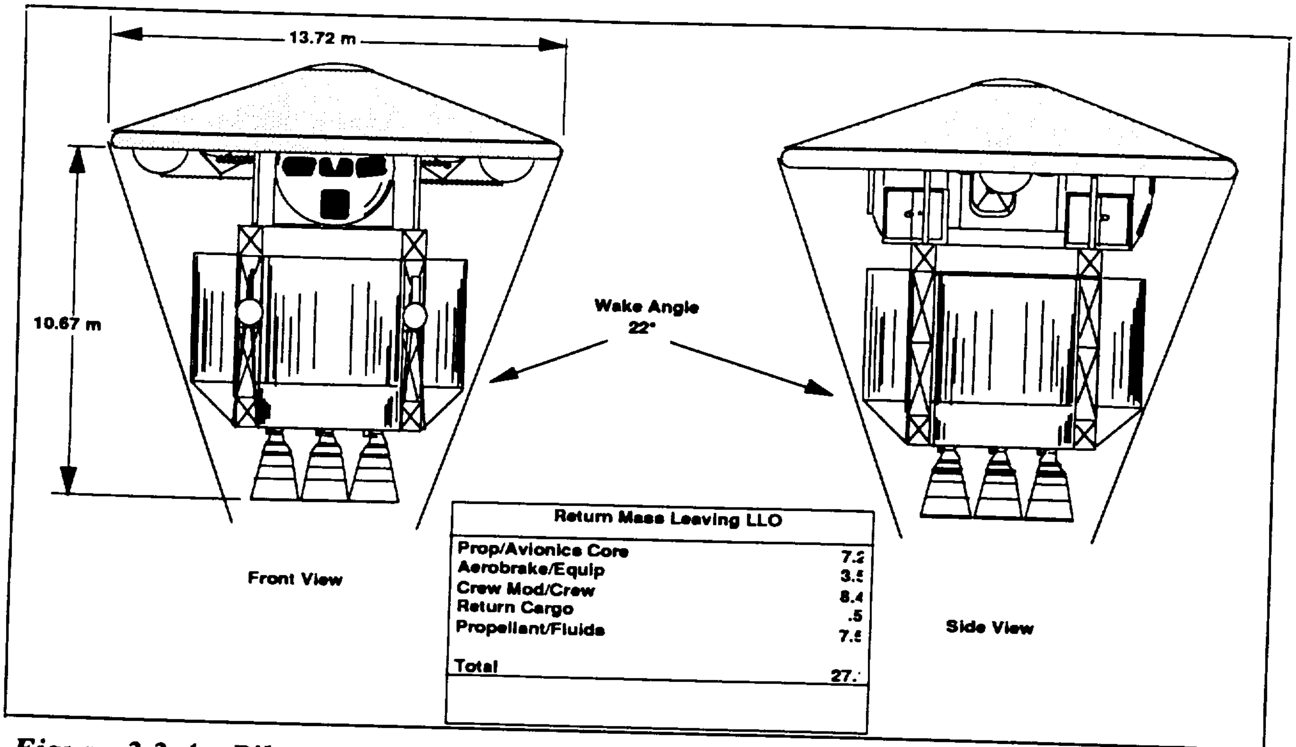
Rendezvous & Docking—After the core and crew module have lifted off from the lunar surface, they must rendezvous and dock in LLO with the aerobrake and its associated equipment for the return flight to SSF. The rendezvous procedure consists of aligning the two vehicles using a target located on the aerobrake. The docking probe on the crew module is extended and then engaged with a grapple fixture located on the aerobrake. Guide rails located inside the aerobrake docking port will help align the vehicles. The docking probe will then be retracted, pulling the crew module/core into the aeroassist position.

After the initial soft dock, the final docking procedure consists of extending the four berthing mechanisms located on the upper platform of the core at each of the corners. These locking probes mate with receptacles located on the aerobrake. Once the final docking has been accomplished, two umbilical connections are made to transfer propellant from the return tanks located in the aerobrake to the engines in the core.

Return Configuration—After the crew module and propulsion/avionics core has ascended from the lunar surface, performed the rendezvous & dock operation with the aerobrake/equipment in LLO, the crew module, core, and aerobrake are returned to SSF using the propellants in the return tanks located in the aerobrake. The piloted return configuration at the beginning of the aeropass is shown in Figure 3.3-4. Once the landing legs of the core are retracted, the crew module and core fit within the 22° wake angle of the aerobrake for the aeroassisted return. The total return mass leaving LLO is approximately 27 mt.

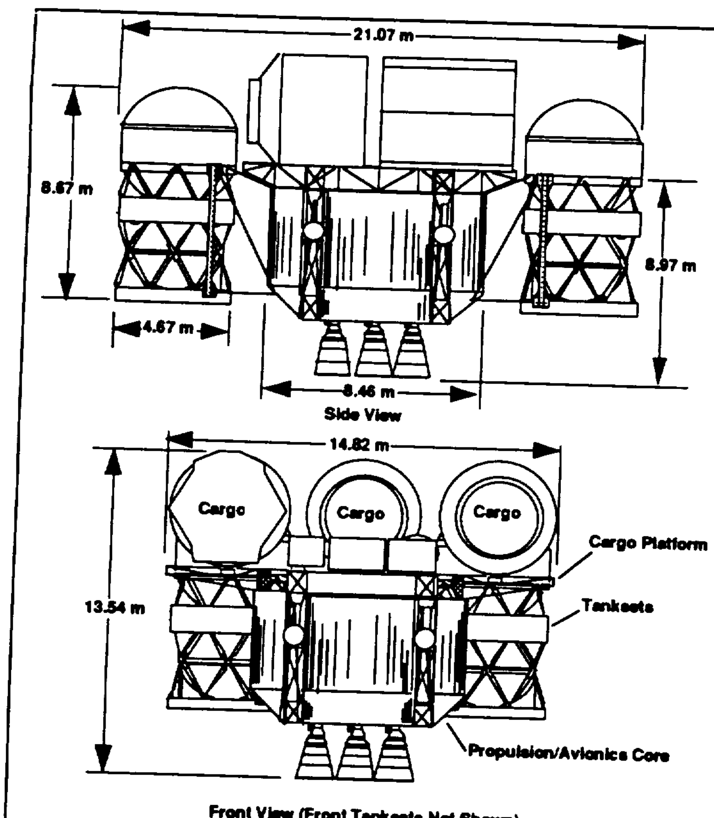
3.4 Cargo Configuration

The cargo configuration is composed of the propulsion/avionics core, a large structural platform, and the drop tanksets common to the piloted configuration. It is designed to deliver 33 mt to the lunar surface in an expendable mode. Figure 3.4-1 shows the overall dimensions of the vehicle as it prepares to leave from LEO. The vehicle is 13.54 m (including the height of the payload) by 14.82 m by 21.07 m. The drop tanks are extended by two meters compared to the piloted vehicle, to accommodate the width of the cargo platform. The core will provide minimum interfaces to the cargo; power but no thermal control. The propellant requirement for the cargo missions is lower than that required for a piloted mission. To keep commonality between both configurations, the



Return Mass Leaving LLO	
Prop/Avionics Core	7.2
Aerobrake/Equip	3.5
Crew Mod/Crew	8.4
Return Cargo	.5
Propellant/Fluids	7.5
Total	27.[illegible]

Figure 3.3-4 Piloted Return Configuration at the Beginning of Aeropass



Mass Properties	
Components	Mass (t)
Propulsion/Avionics Core	7.1
Tanksets (4 TLI & 2 LOI)	9.1
Cargo Platform	2.4
Vehicle Dry Mass	18.7
Propellant Mass	146.5
Cargo Mass	33.0
Total Mass	198.2

Figure 3.4-1 Overall Dimension of Vehicle Leaving LEO

drop tanks are the same as those on the piloted vehicle, however propellant is offloaded to meet the mission requirements. The vehicle can deliver up to 37.4 mt of cargo.

Cargo Platform—In order to accommodate the large volume cargo manifested to the lunar surface, special structure must be added to the basic core structure to provide structural support. The overall view of the platform is shown in Figure 3.4-2. The cargo support area is approximately 14.8 m by 10.5 m in size with the cargo extensions added. The larger area is formed by adding two central platform extensions and two outer platform extensions to the basic core structure. These extensions are made of lightweight trusses, and can be folded and returned for additional uses. Cargo is mounted using center keel and trunnion fittings similar to those on the STS.

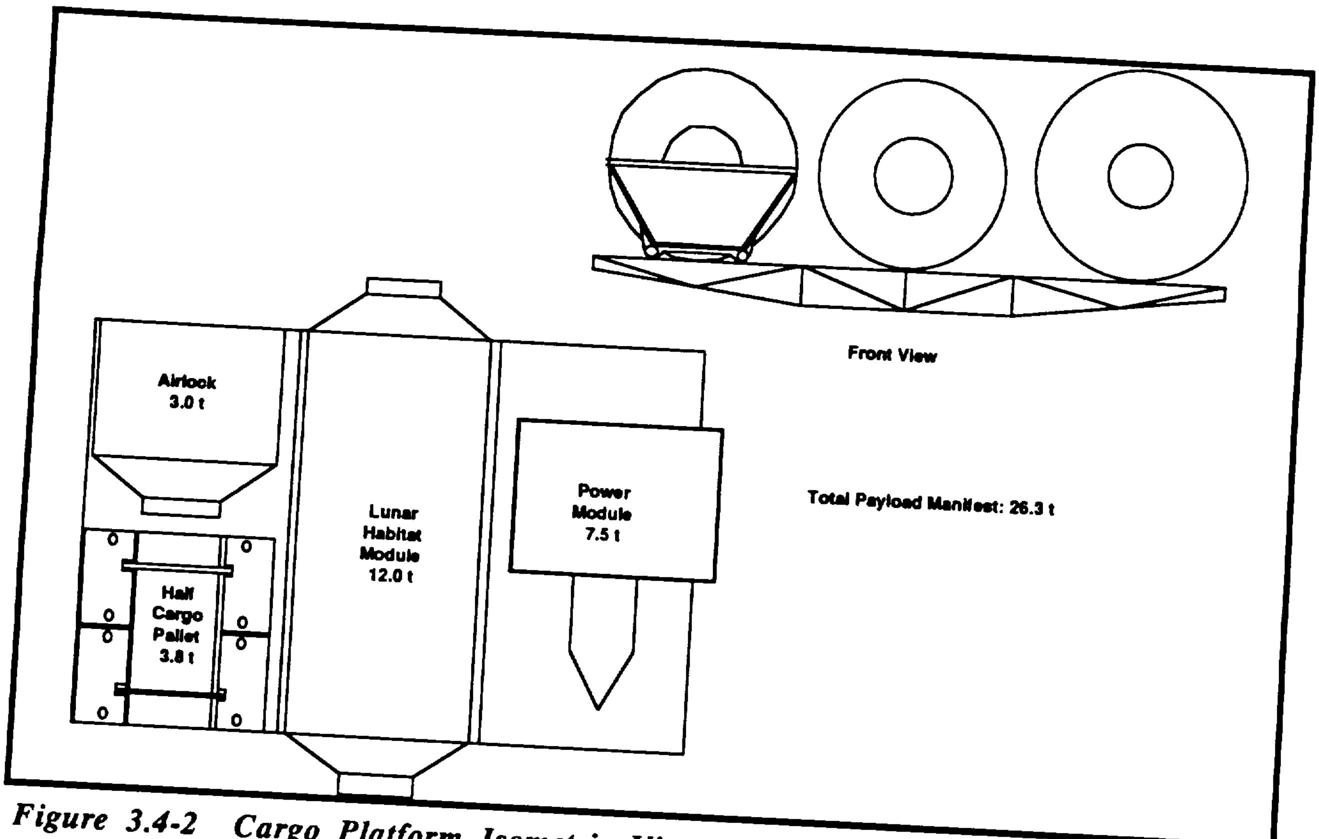


Figure 3.4-2 Cargo Platform Isometric View

Cargo Offloading—Cargo Flight 0 will deliver the LEVPU, a three leg crane, that will unload all the other cargo flights and can assist in unloading the cargo from the piloted vehicle if required. The LEVPU is designed to be self unloading. Figure 3.4-3 shows how the LEVPU will unload the cargo from the cargo expendable configuration once the vehicle has landed on the lunar surface. The platform and the vehicle size allows the payload unloader to straddle the lander vehicle. Once positioned over the vehicle the unloader picks up a piece of cargo, lifts it, and proceeds to roll away from the vehicle. After the cargo has been deposited in its position on the lunar surface or on a transporter, the unloader will proceed back to the vehicle to unload subsequent pieces of cargo.

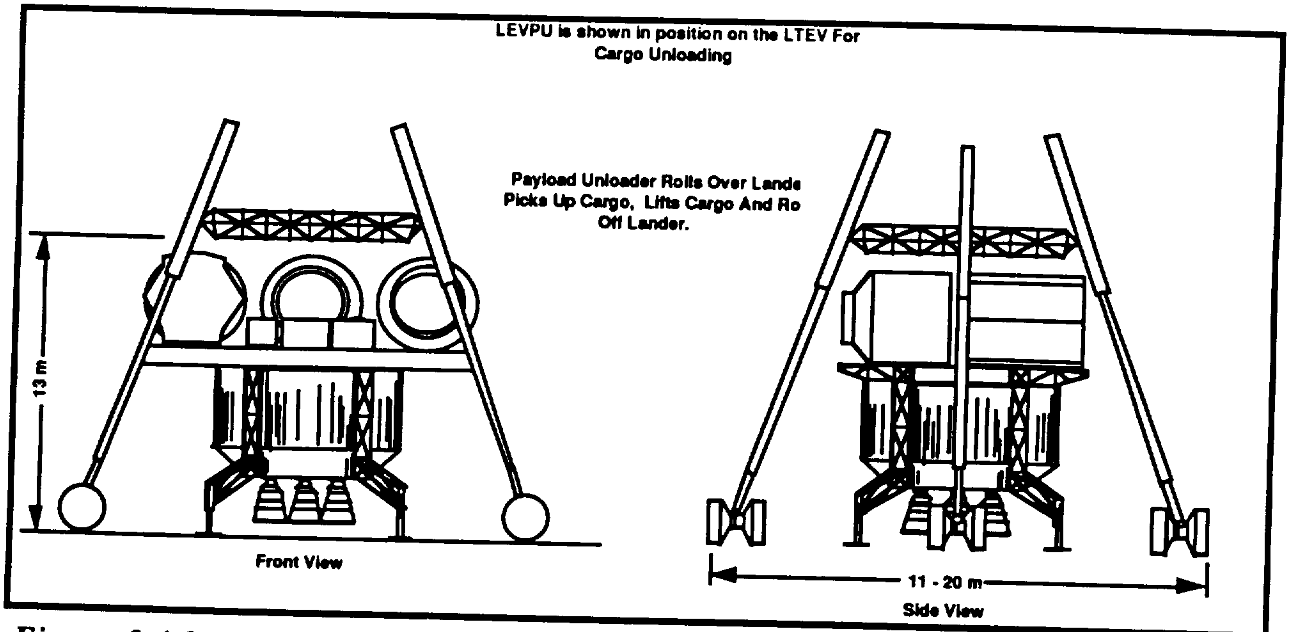


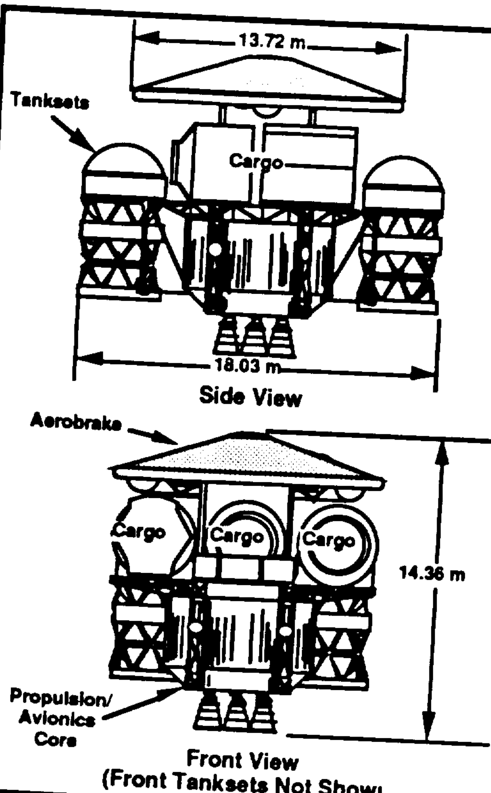
Figure 3.4-3 Shows LEVPU Unload Cargo

3.5 Cargo Reusable Configuration

An optional cargo reusable configuration (Figure 3.5-1) for the single propulsion system concept has been proposed. The six tanksets, an aerobrake and the large cargo platform are attached to the common propulsion/avionics core. The four docking probes provided on the piloted vehicle can be positioned to accommodate the taller payloads. The configuration can deliver approximately 26 mt of cargo to the lunar surface and return the vehicle to SSF using 169.3 mt of LO2/LH2 propellant. The 13.72 m rigid aerobrake protects the vehicle during the aeroassisted return to SSF.

3.6 Initial & Growth STV Concept Definition

A common set of engines, tanksets, cores, aerobrakes, crew modules, subsystems, etc. were found to be applicable in the development of various ground- or space-based, expendable or reusable STV configurations including the lunar transportation system.



Mass Properties	
Components	**Mass (t)**
Prop/Avionics Core	7.19
Tanksets (4 TLI & 2 LOI)	9.11
Cargo Platform	2.45
Aerobrake & Equip	3.50
Vehicle Dry Mass	22.25
Propellant	169.3
Cargo	25.9
Total Mass	217.45

Figure 3.5-1 Optional Cargo Reusable Configuration

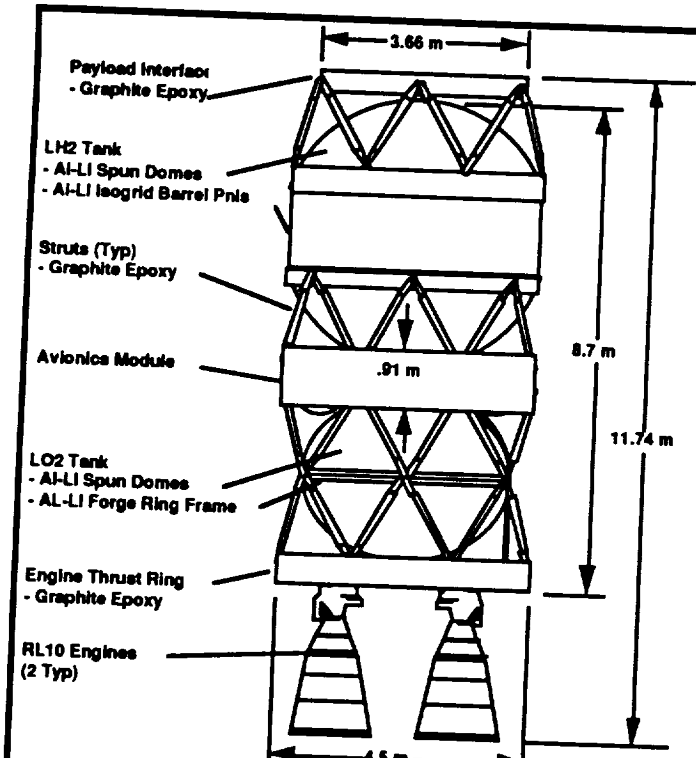
The ability of the baseline vehicle or elements of the baseline vehicle to perform the other DRM cargo requirements was evaluated and is depicted in Table 3.6-1. All DRM cargo requirements can be met by either the initial STV or the baseline's core vehicle with one set of drop tanks. The capability of the stages was determined using the RL10A-4 cryogenic engine at 449.5 seconds of Isp and the various pieces of the LTV as listed in table. The table shows the minimum needs of the core vehicle to meet the DRM cargo requirements in terms of extra propellant and subsystems, e.g. the crew module for the manned mission.

Expendable Initial Concept—The initial STV (Figure 3.6-1), a ground-based expendable version, can be built from the common set of elements and subsystems. A common tankset and two engines with limited subsystems form the basis for this vehicle. It is sized to fit within a 4.6 m (15 ft) diameter payload shroud for delivery to orbit. The dry weight of the vehicle is about 3 t with a length of nearly 12 m. With approximately 28 tonnes of LO2/LH2 propellant in the tankset, the vehicle can deliver 12.9 tonnes of payload to a geosynchronous orbit.

Table 3.6-1 Baseline Vehicle Adaptability

DRM	Description	Cargo Requirement	LTS/STV Configuration
E - 1	Manned GEO Servicing	4.0 t delivery & return	4E-5B Core w/AB, Crew Module, & 43 t Prop in Drop Tanks
E - 2	10 t GEO Platform Delivery *(DELETED IN CNDB '90)*	10.0 t delivery	Interim Vehicle (12.9 t maximum capability)
E - 3	6.4 t GEO Payload Delivery (DoD)	6.4 t delivery	Interim Vehicle (12.9 t maximum capability)
E - 4	Unmanned Polar Platform Servicing	3.5 t delivery & return	4E-5B Core w/AB, & 26.3 t Prop in Drop Tanks
P - 1	Comet Nucleus Sample Return *(DELETED IN CNDB '90)*	16.0 t delivery	4E-5B Core & 5.1 t Prop in Drop Tanks

DRM Propellent Loads Are Based on the Use of RL10A-4 Engines (449.5 sec)



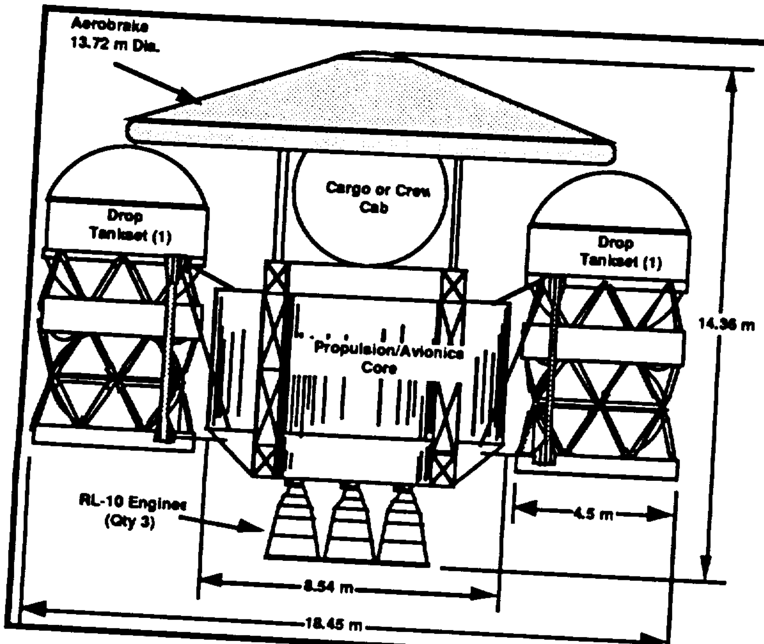
Mass Properties

Components	Mass (t)
Structure	0.68
Propellant Tanks	0.52
Propulsion System	0.31
Main Engines	0.31
RCS ystem	0.09
GN&C	0.07
Communication & Data Handling	0.15
Electrical Power	0.25
Thermal Control System	0.38
Contingency (15%)	0.41
Total Dry Weight	3.17

Performance - 12.9 t Max to GEO

Figure 3.6-1 Ground-Based Expendable Vehicle

Reusable Initial Concept—This STV, a space-based reusable version (Figure 3.6-2), can also be built from the common set of elements and subsystems. Two common tanksets, three engines, an aerobrake, and a core vehicle with limited subsystems form the basis for this vehicle. The dry weight of the vehicle is about 12 mt with an assembled length of over 14 m and width of over 18 m. The extra propellant tanksets provide an enhanced performance capability for delivery and return of geosynchronous payloads. The payload can be either deliverable cargo or (for some missions) a crew module with crew.



Mass Properties	
Components	Mass (t)
Structure	2.3
Propellant Tanks	0.8
Propulsion System	4.3
Main Engines	1.1
RCS System	0.1
GN&C	0.2
Communication & Data Handling	0.2
Electrical Power	0.4
Thermal Control System	0.5
Aerobrake	1.3
Contingency (15%)	1.5
Total Dry Weight	12.1

Performance:
Satisfies DRM E - 1(4.0 t GEO Delivery & Return) and DRM E - 4 (3.5 t GEO Delivery & Return) with
2 Partially Filled Drop Tanksets

Figure 3.6-2 Space-Based Reusable Vehicle

4.0 STV OPERATIONS

The LTS operations concept identifies the ground processing requirements to prepare elements for launch to LEO, the Earth-To-Orbit (ETO) transportation of the configuration elements, assembly & checkout of the system at LEO, flight operations from LEO to LLO, decent and ascent and LLO rendezvous and docking, flight operations from LLO to LEO, and post flight checkout and refurbishment of the system. Figure 4.0-1 shows an overview of the elements required to perform the lunar mission.

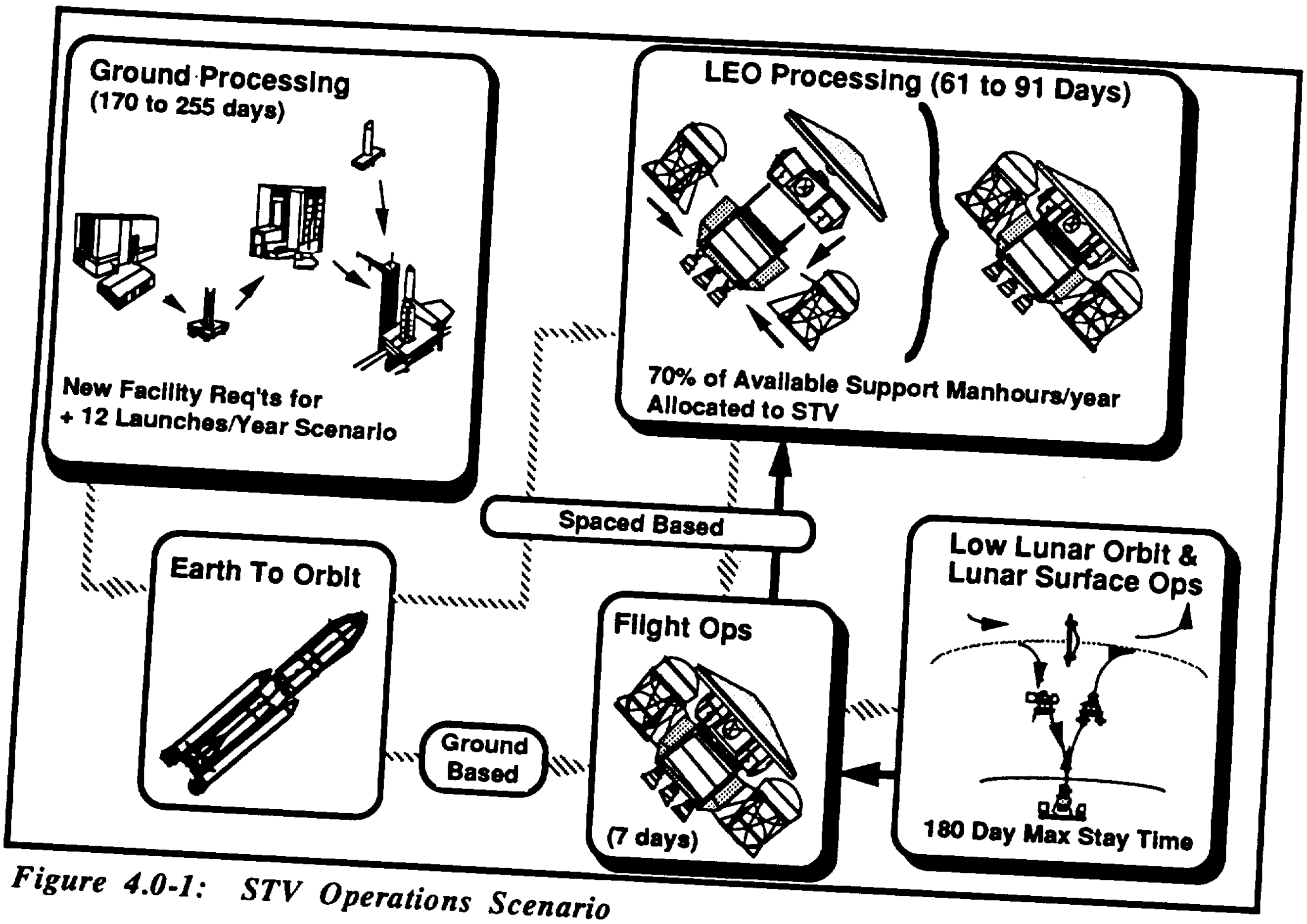


Figure 4.0-1: STV Operations Scenario

4.1 Ground Operations

The processing flow for the present STS shuttle orbiter is used as the basis for the development of the LTS/STV ground operations scenario. The LTS/STV vehicle has a modular configuration and consists of the crew module, core vehicle module, aerobrake module, TLI/LOI/RET Tankset modules, and cargo modules. These modules will be processed individually on the ground, manifested and carried to orbit in the payload shroud of the HLLV, and assembled in orbit at space station.

4.1.1 LTS/STV Ground Operations.

The LTS/STV is considered a payload for the HLLV while simultaneously carrying cargo modules of its own. It is shown that stand alone processing for STV modules and vertical integration into the HLLV payload shroud will be performed in a new combined STV Processing & Integration Facility (SPIF). Processing of LTS/STV at KSC begins with the receipt of system modules by air and/or barge. These components are then transferred to the SPIF for stand alone processing and subsequent installation into the HLLV's P/L Shroud. The integrated STV/shroud is then transferred to the VAB for mate and integration into the HLLV. After interface testing is complete in the VAB the entire stack is moved to launch pad LC-39C for final HLLV checkout, servicing and launch.

LTS/STV ground processing takes 50 days of initial stand-alone processing of the basic vehicle with subsequent supporting processing at 20-30 day intervals for tank module flights. The minimum launch interval would be constrained by the launch vehicle and not by LTS/STV. Installation and integration of LTS/STV would occur in the VAB and would not impact any other shuttle processing. Also, loading of the cryogenic propellants could occur the day before launch and have no close-out or impact on the final countdown.

4.1.2 ETO Processing and Requirements.

The baseline concept is capable of supporting one lunar mission per year consistent with 'Option-5', - requiring an initial Heavy Lift Launch Vehicle (HLLV) manifest of 3 launches with final STV assembly at SSF. It is planned that STV will be processed and launched at KSC Launch Complex-39 (LC-39) as a payload on a 75 tonnes HLLV ETO launch vehicle. For the purpose of this study it has been assumed that the new HLLV is planned to co-reside with STS shuttle, however, it will have its own dedicated launch pad, LC-39C. Accordingly, processing will be in concert with the existing STS shuttle program and will share integrated processing facilities, supporting services, and range services. Wherever possible, existing facilities are used as shown on Figure 4.1.2-1. New facilities are identified only where the vehicle design is incompatible with existing facilities or where planned rate usage has saturated facility capacity.

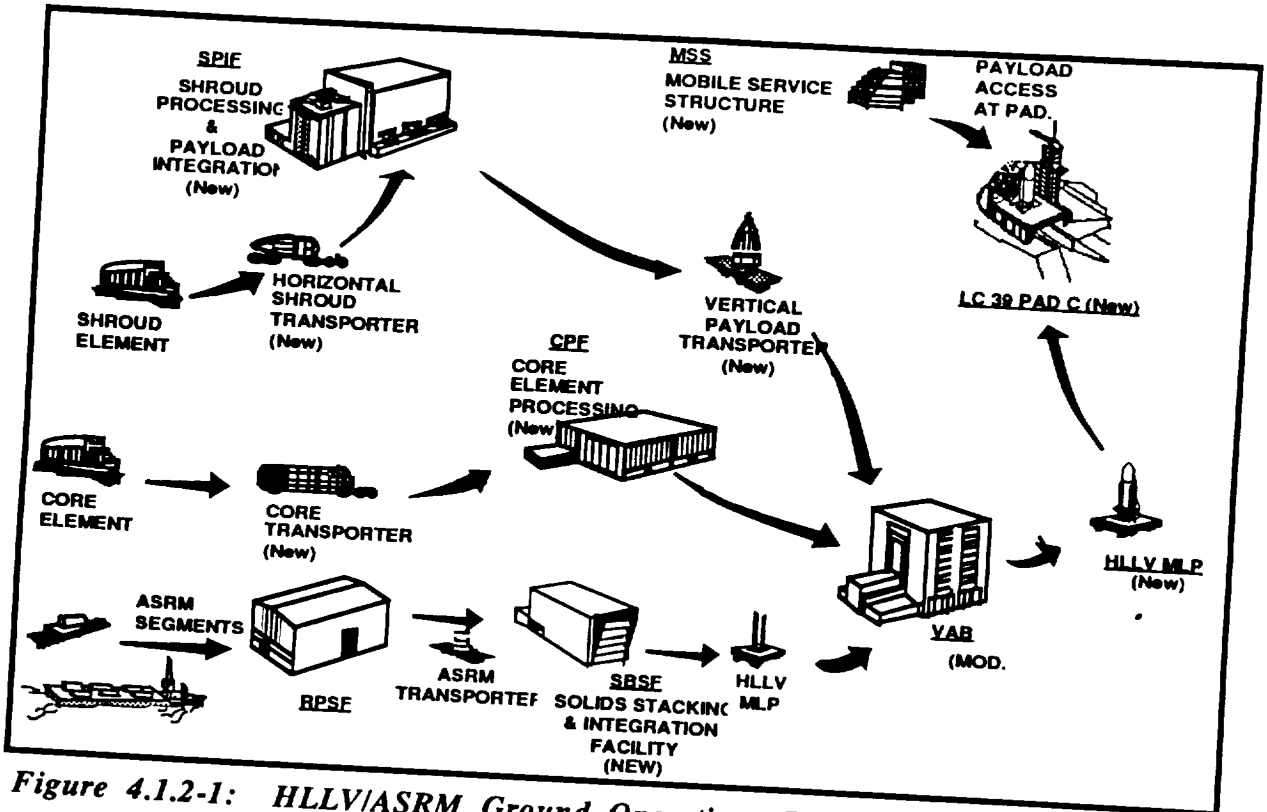


Figure 4.1.2-1: HLLV/ASRM Ground Operations Flow

Processing and launch of the LTS elements as shown on Figure 4.1.2-2 is conducted in six primary tasks and four secondary tasks that involve the processing of the ETO vehicle itself.

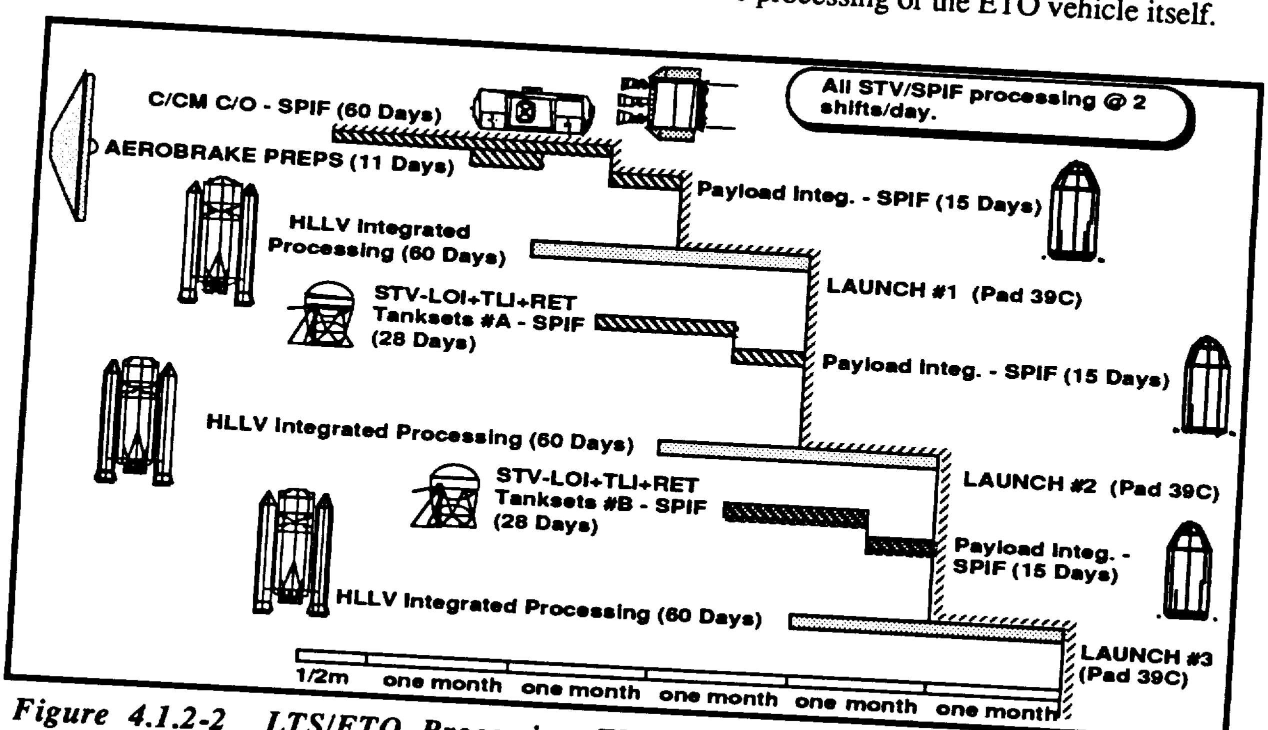


Figure 4.1.2-2 LTS/ETO Processing Flow

After receipt, the LTS/STV elements are checked out and integrated into the ETO fairing/shroud, a seventy-five day task. The integrated payload element is the transported to the Vehicle Assembly Building (VAB) for assembly onto the ETO booster element, a ten day task. The completed ETO vehicle is then transferred to the launch pad, where it is processed for launch. The total ground time requirements for the LTS is eighty-five days to launch. To support an initial mission, three ETO flights are required, for a steady state mission, two ETO flights are required. Prior to mating of STV the HLLV is stacked onto the MLP along with its two boosters at the VAB.

The boosters and the HLLV core vehicle have previously been prepared and checked out in their own stand-alone facilities. The Payload Shroud (PLS) containing the LTS/STV is transferred vertically from the SPIF to VAB's transfer isle. The shroud assembly is then hoisted from the transfer isle onto the top of the HLLV stack in the integration cell. Subsequent to the PLS/LTS/STV mate the entire HLLV undergoes interface and integration testing, ordnance installed and is prepared for roll-out to the launch pad.

Roll out to the launch pad and 'hard-down' takes about 8 hours. After connections to the facility are complete interface checks are made followed by final checkout of the launch vehicle and payload including communications and instrumentation verification. Final servicing (fluids, power, etc.) of all systems is performed just prior to start of the launch countdown. During the launch countdown after all systems power-up, final confidence checks are performed on critical systems and liquid propellants are loaded. LTS/STV propellants will be loaded first and the HLLV last. After propellants are loaded they will be continuously monitored and vented through pad facilities; at launch the LTS/STV will be locked up and no venting permitted until after booster burnout - above 75,000 feet.

4.2 Space Operations

The space operations for the LTS/STV consists primarily of two phases. The first involves the activities that take place in Low Earth Orbit (LEO) followed secondly by the inflight operations that support the transport of the vehicle from LEO to it's destination. In the case of manned missions, the system is returned to LEO for refurbishment and preparation for the next mission.

4.2.1 Low Earth Orbit Operations

The LEO node has been identified as the transportation node for the lunar exploration missions. The primary element of the LEO will be Space Station Freedom (SSF) and its proximity operations support equipment. A general overview of the defined operations in LEO initiate with the ETO system delivering LTS hardware elements to a SSF parking location. This point in LEO has been defined as being approximately 20 miles from SSF. Elements of SSF Proximity Operations SE transport these elements back to SSF, where they are received and readied for assembly and checkout. Following the completion of the assembly activity, the system undergoes a final flight readiness verification test. The system is then transferred from SSF to its TLI station again using SSF Proximity Operations SE.

Figure 4.2.1-1 defines the complete set of timelines for the processing of LTS elements for both the first flight and steady state scenarios. For the initial flight mission, there are six primary activities performed at LEO (SSF). The hardware delivery phase (16.5 days), receives the LTS components at SSF where an element level checkout is conducted. The assembly phase (17.5 days) assembles the LTS components into an operational configuration. This is followed by the verification phase (16 days) that ensures the flight readiness of the system. With the system

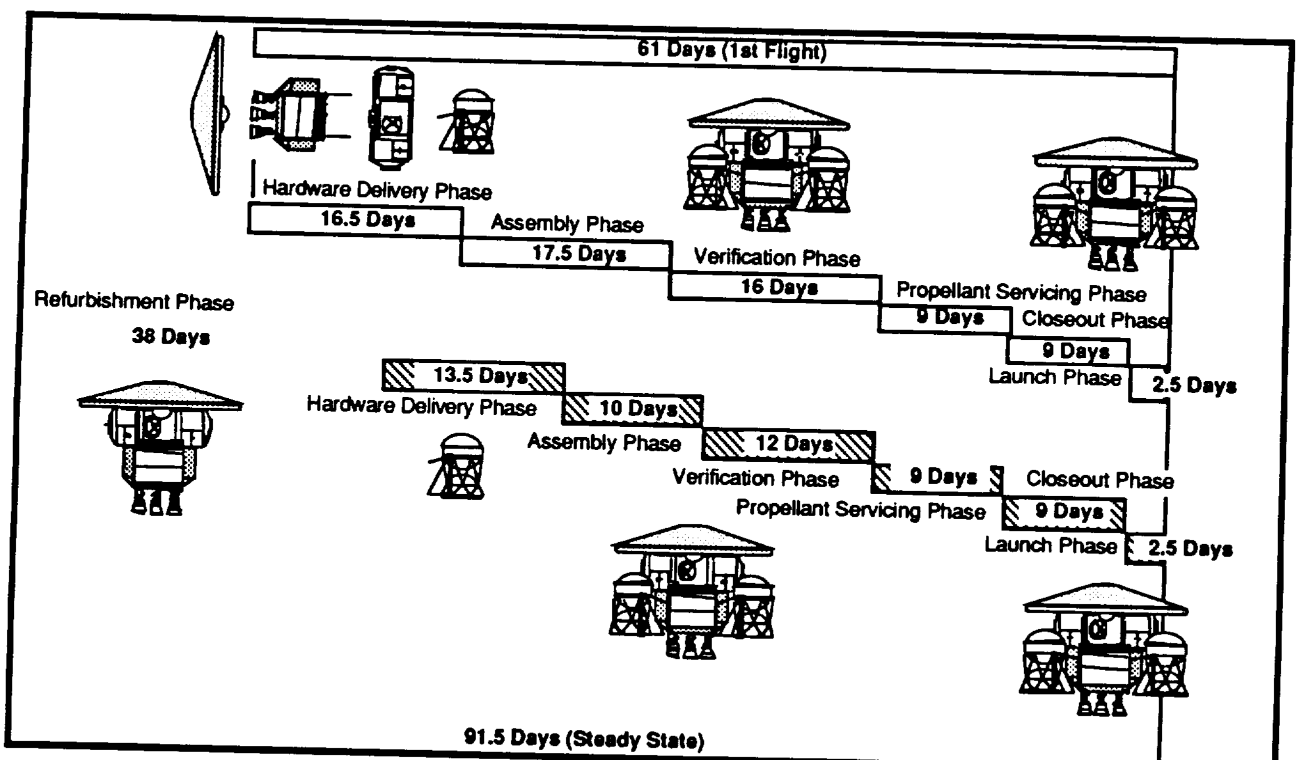


Figure 4.2.1-1 LTS Processing Timelines

mission ready, the propellant servicing phase (9 days) assembles the drop tanks to the mission vehicle. The closeout phase (9 days) provides final launch readiness, and is followed by the launch phase (2.5 days). The launch phase delivers the mission crew, transport the LTS to the injection burn location, and initiates TLI. Total processing time for an initial flight mission is 61 days, although due to the KSC and SSF constraints, the actual time required to process the LTS is 265 days.

4.2.2 Space Flight Operations

Once the processing activities at the LEO node have been completed and the LTS transferred away from the node to a remote location, the initial phase of the space flight activates begin. Space flight operations encompass those functions that make up the outbound mission from LEO to low lunar orbit, the rendezvous and docking and station keeping activities in LLO prior to descent and following ascent, descent and ascent to the lunar surface from LLO, and the inbound mission from LLO to LEO and recovery by the LEO node. Figure 4.2.2-1 shows the complete space flight

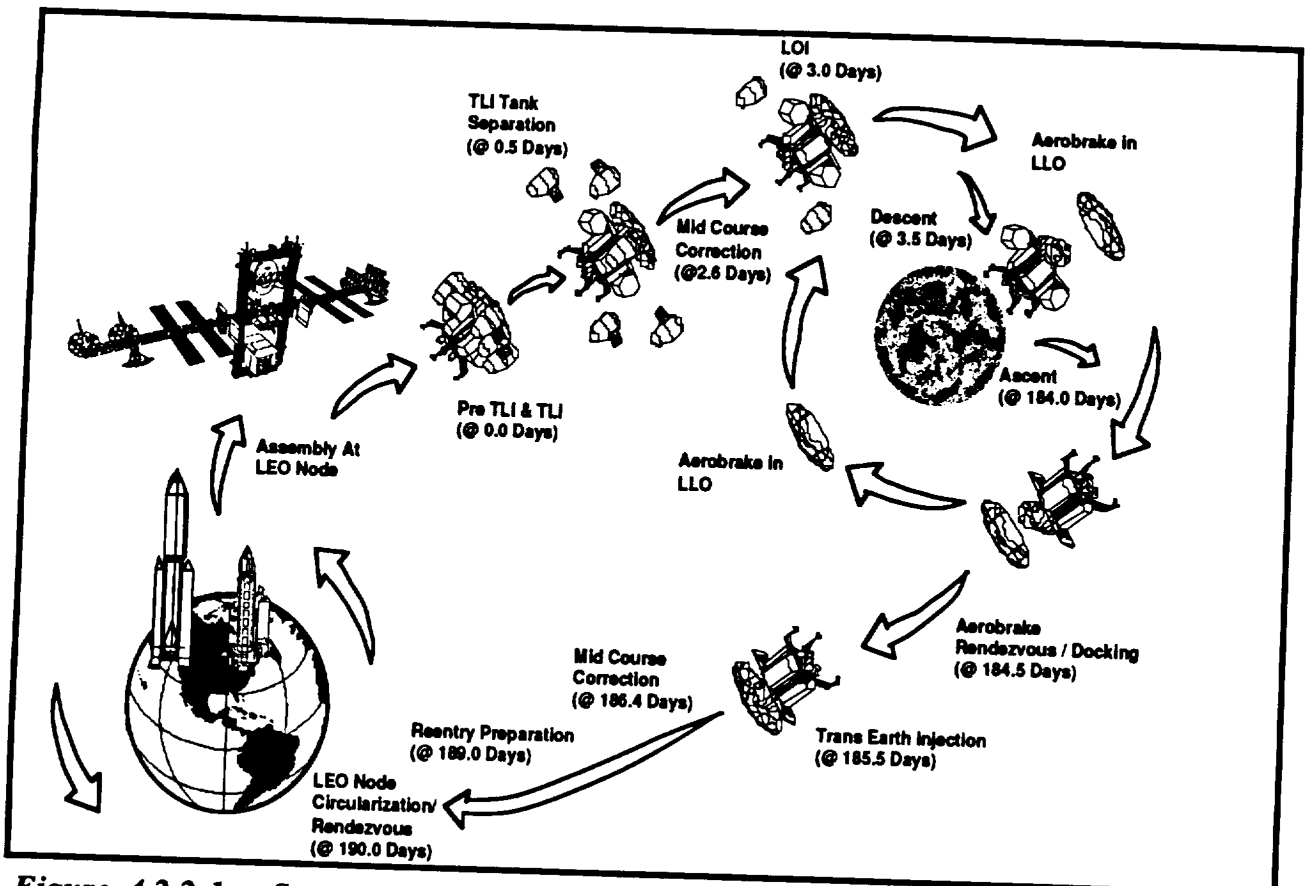


Figure 4.2.2-1: Space Flight Operational Functions and Timelines

architecture that has been defined for the LTS mission. Although the figure represents a piloted mission, the reusable cargo mission uses the same mission functions and the expendable cargo missions follow the same functions through descent to the lunar surface.

Figure 4.2.2-5 shows the overall mission timeline for a piloted mission, starting with receipt of hardware in LEO, the initial mission, system refurbishment, conduct of a steady state mission including return to the LEO node. Details of the LEO processing phases of this timeline have been defined in section 4.2.1, Ground Processing.

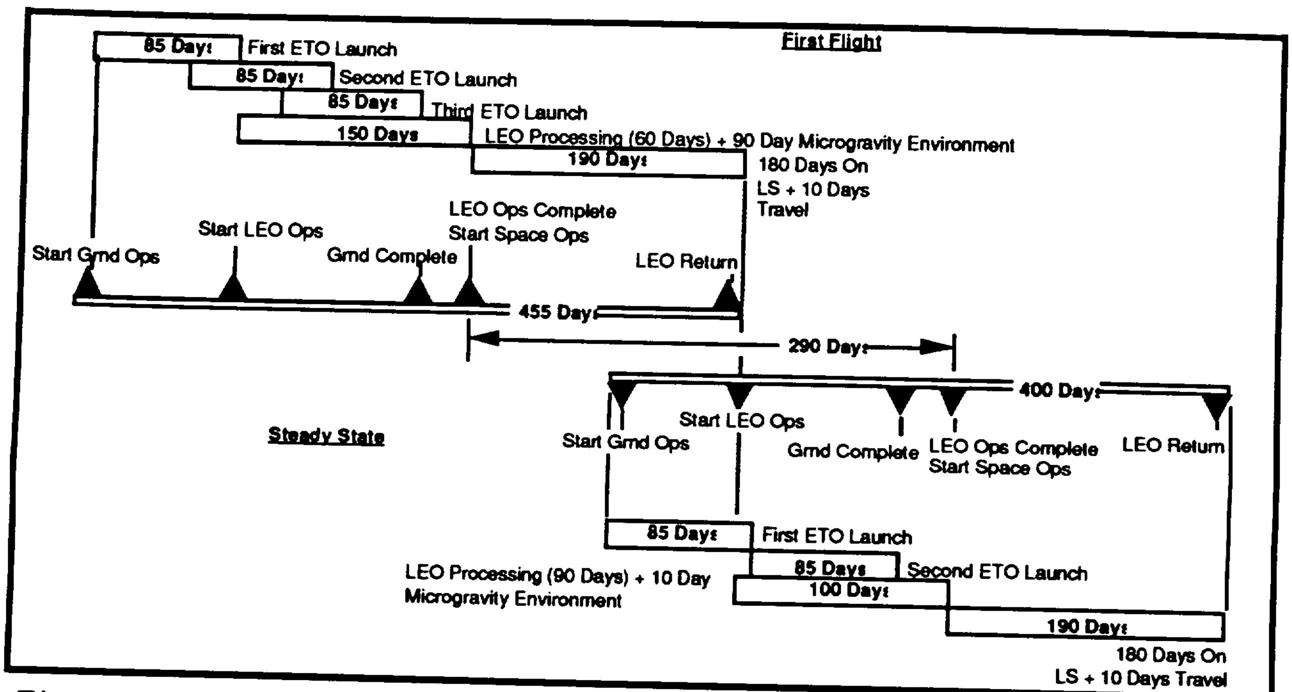


Figure 4.2.2-5: Overall LTS Mission Timeline

4.3 SURFACE OPERATIONS

The LTS operations on the lunar surface are limited to cargo and crew loading and unloading, station-keeping monitoring, and unscheduled maintenance of mission critical elements. Once the cargo has been delivered, it must be unloaded by surface support equipment or by the LTS to transportation equipment, because deliveries are made in both cargo and piloted configurations, both unloading systems will be used. The large cargo platforms require surface loading/unloading equipment to be available, as unloading of these platforms is not feasible with the current piloted system configuration. This surface unloader/loader has been defined as the Lunar Excursion Vehicle Payload Unloader (LEVPU) by Planetary Support Systems (PSS) inputs to the "Option 5" SEI Lunar Outpost Initiative. Figure 4.3-1 shows the LEVPU unloading cargo from the cargo

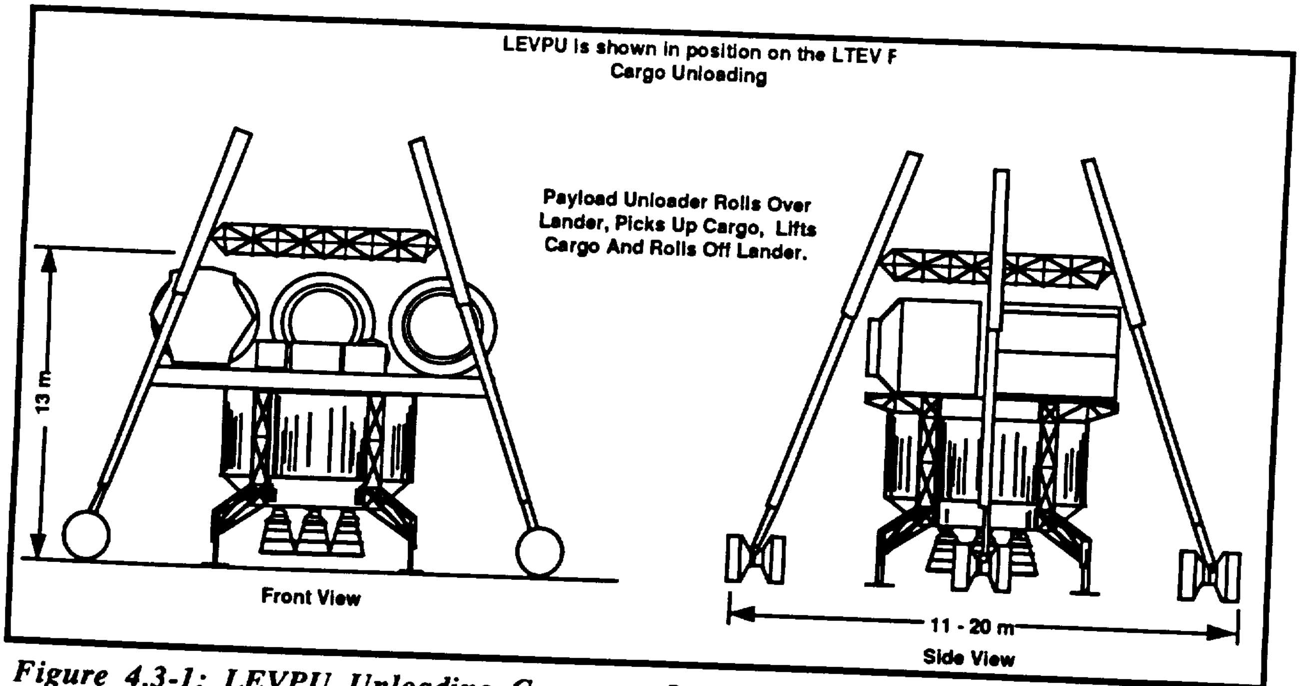


Figure 4.3-1: LEVPU Unloading Cargo on Lunar Surface

configuration on the lunar surface. The vehicle configuration is sized to allow the payload unloader to roll over and straddle the vehicle with its cargo. Once positioned over the vehicle, the unloader picks up a piece of cargo, lifts it, and proceeds to roll away from the vehicle. After the cargo has been deposited in its position on the lunar surface or on a transporter, the unloader proceeds back to the vehicle to unload another piece of cargo. Cargo unloading of the piloted vehicle on the Lunar surface can be accomplished without the use of the LEVPU, as shown in Figure 4.3-2. The cargo is supported by supports extending from the sides of the core. Once the vehicle has landed on the lunar surface, the cargo can be lowered directly to the surface or onto a transporter by using a hoist mounted on the cargo support structure. These hoists allow cargo to be lowered directly to the lunar surface. The spacing between the legs of the core allows the cargo to be lowered directly to the surface.

After landing, connection of the surface umbilicals for transferring of propellant and data management will be made by surface support equipment. Details of this function as well as the equipment to conduct it, have not been defined at this time; however, it is known that the interfaces to the LTS will be compatible with those used at SSF and KSC.

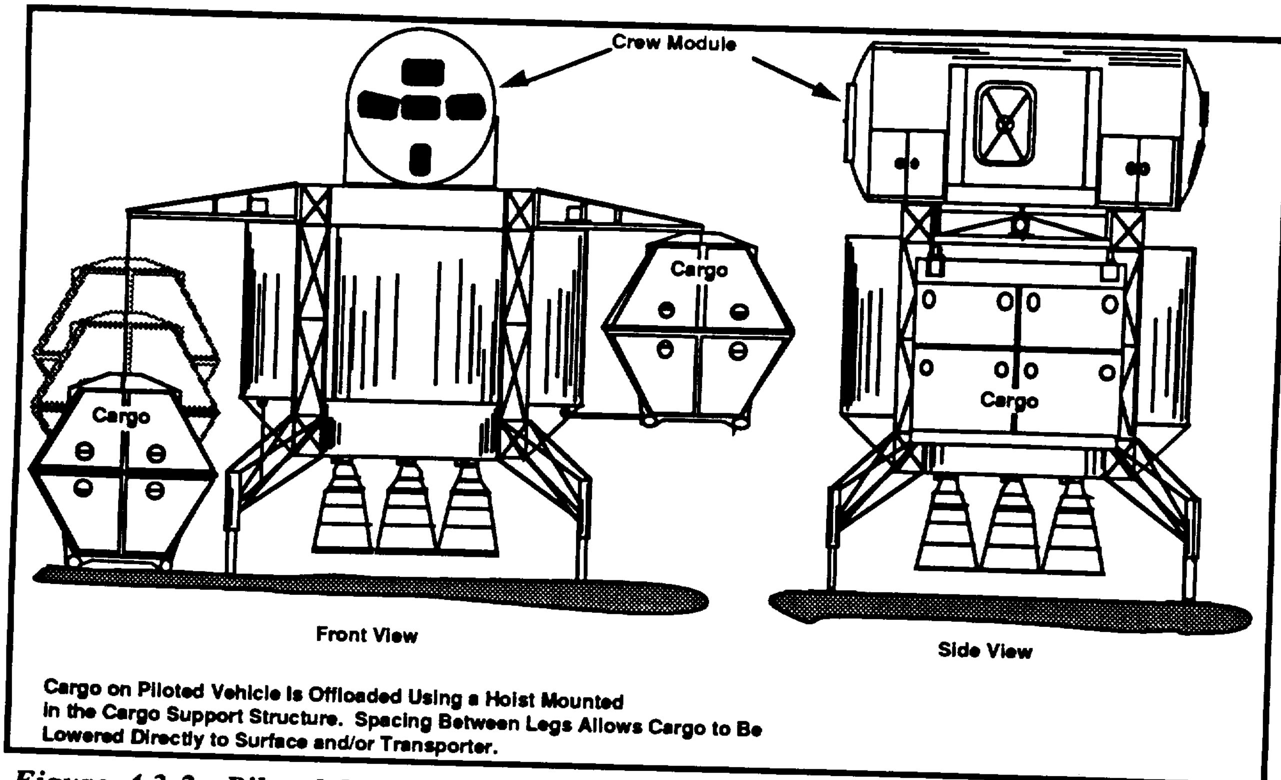


Figure 4.3-2: Piloted Vehicle Unloading Cargo on Lunar Surface

4.4 Interfaces

The LTS will interface with several of the primary space infrastructure elements during the execution of a single lunar mission. These elements include the ground processing facilities at KSC, the ETO system during transport into LEO, SSF during assembly, verification, and refurbishment, PSS cargo during transfer between LEO and the lunar surface, and the lunar outpost facilities throughout the duration of the surface stay time. Discussed in this section will be the principle interfaces as defined for each of these support nodes.

The STV interfaces for both ground processing and the HLLV are identified in Tables 4.4-1 and 4.4-2. Envelope dimensions indicate the handling size but do not include accessibility requirements or GSE allowances. Vertical transporters, handling dollies, and tractors are required for each of the STV modules and requires (or shares) an HLLV payload shroud vertical transporter. Electrical power will interface with the ground system only during stand alone processing in the SPIF using drag on cables.

Table 4.4-1: KSC Ground Processing Interfaces

Interface	Core	Crew Module	Aerobrake
Envelope	8.5 m	3.7 m dia x 8.5 r	8.5 m dia envelop
Handling	hooks & fittings to I/F w/ vertical transporter dollies and tractors	hooks & fittings to I/F w/ vertical transporter dollies and tractors	hooks & fittings to I/F
Electrical	drag on cables - SPIF ASE thru HLLV on pad	drag on cables - SPIF ASE thru HLLV on pad	drag on cables - SPIF ASE thru HLLV on pad
Mechanical	Handling-Gnd / HLLV	Handling-Gnd / HLLV	Handling-Gnd / HLLV
Propellants	N/A	Life support fluids loaded in SPIF	NA
Pneumatics	loaded in SPIF	loaded in SPIF	loaded in SPIF
Environmental Control	HLLV shroud purg	HLLV shroud purg	HLLV shroud purg
Safety	High pressure gasses cryo handling	High pressure gasses cryo handling	No unusual safety requirements
Security	normal NASA requirements	normal NASA requirements	normal NASA requirements
Communications	ground I/Fs thru fiber optical, RF or IR links	ground I/Fs thru fiber optical, RF or IR links	ground I/Fs thru fiber optical, RF or IR links
Cabling	electrical and instrumentatio	electrical and instrumentatio	electrical and instrumentatio
Operational Constraints	Pos pressure on tank Maintain clean systen	Pos pressure on tank Maintain clean systen Maintain cabin ai	Pos pressure on tank Maintain clean systen

Table 4.4-2: KSC Ground Processing Interfaces

Interface	TLI/LOI Tanks	Return Pallets
Envelope	4.6 m dia x 8.7 m ea.	4.6 m x 2.7 m x 2.6 m (pallet)
Handling	hooks & fittings to I/F w/ vertical transporter dollies and tractors	hooks & fittings to I/F
Electrical	drag on cables - SPIF ASE thru HLLV on pad	drag on cables - SPIF ASE thru HLLV on pad
Mechanical	Handling-Gnd / HLLV	Handling-Gnd / HLLV
Propellants	filled thru umbilicals on HLLV shroud	filled thru umbilicals on HLLV shroud on pad
Pneumatics	loaded in SPIF	loaded in SPIF
Environmental Control	HLLV shroud purge	HLLV shroud purge
Safety	High pressure gasses cryo handling	High pressure gasses cryo handling
Security	normal NASA requirements	normal NASA requirements
Communications	ground I/Fs thru fiber optical, RF or IR links	ground I/Fs thru fiber optical, RF or IR links
Cabling	electrical and instrumentation	electrical and instrumentation
Operational Constraints	Pos pressure on tanks Maintain clean system	Pos pressure on tanks Maintain clean system

Because SSF conducts many of the same types of functions performed at KSC, similar interfaces are found. These interfaces provide an unpressurized area which provides meteoroid protection, and active and passive thermal control for the STV. A teleoperator manipulator dedicated to STV is planned along with an interface with SSF electrical power. Communications and tracking are provided by SSF for the monitoring of critical operations and support of overall mission functions.

During transportation of the crew and cargo, or just cargo to and from the lunar surface, interfaces between the LTS and the cargo exist. To minimize the impact to the LTS, the interfaces shown in Table 4.4-6 include only the physical attachments of the cargo to the vehicle and electrical to provide monitoring of the health cargo itself. Handling attachments for placing the cargo on the STV will be provided by the cargo. No liquid or pneumatic interfaces will be supplied by the STV to the cargo although minimal electrical power for monitoring and statusing is provided. Environmental control and meteoroid protection, if required, is supplied by the cargo. Communications support will be provided by STV for health and status monitoring only.

5.0 PROGRAMMATICS

5.1 PROJECT PLANNING AND CONTROL

During the initial phase of the Space Transfer Vehicle Concepts and Requirements Study contract, the project planning, project finance, and project data management activities were combined into a single functional task. This task provided management with the tools required to control the business management aspects of the contract. The study plan (DR-1) was updated after negotiations, submitted and approved by NASA/MSFC. This study plan was then used to monitor program schedule and cost performance. The STV Study Program Master Schedule (Figure 5.1-1) and program technical status were then reported to NASA/MSFC in the monthly program progress report (DR-3). The monthly program financial status was reported to NASA/MSFC via the NASA form 533M, and an estimate to complete was provided to NASA/MSFC on a quarterly basis in the NASA form 533Q.

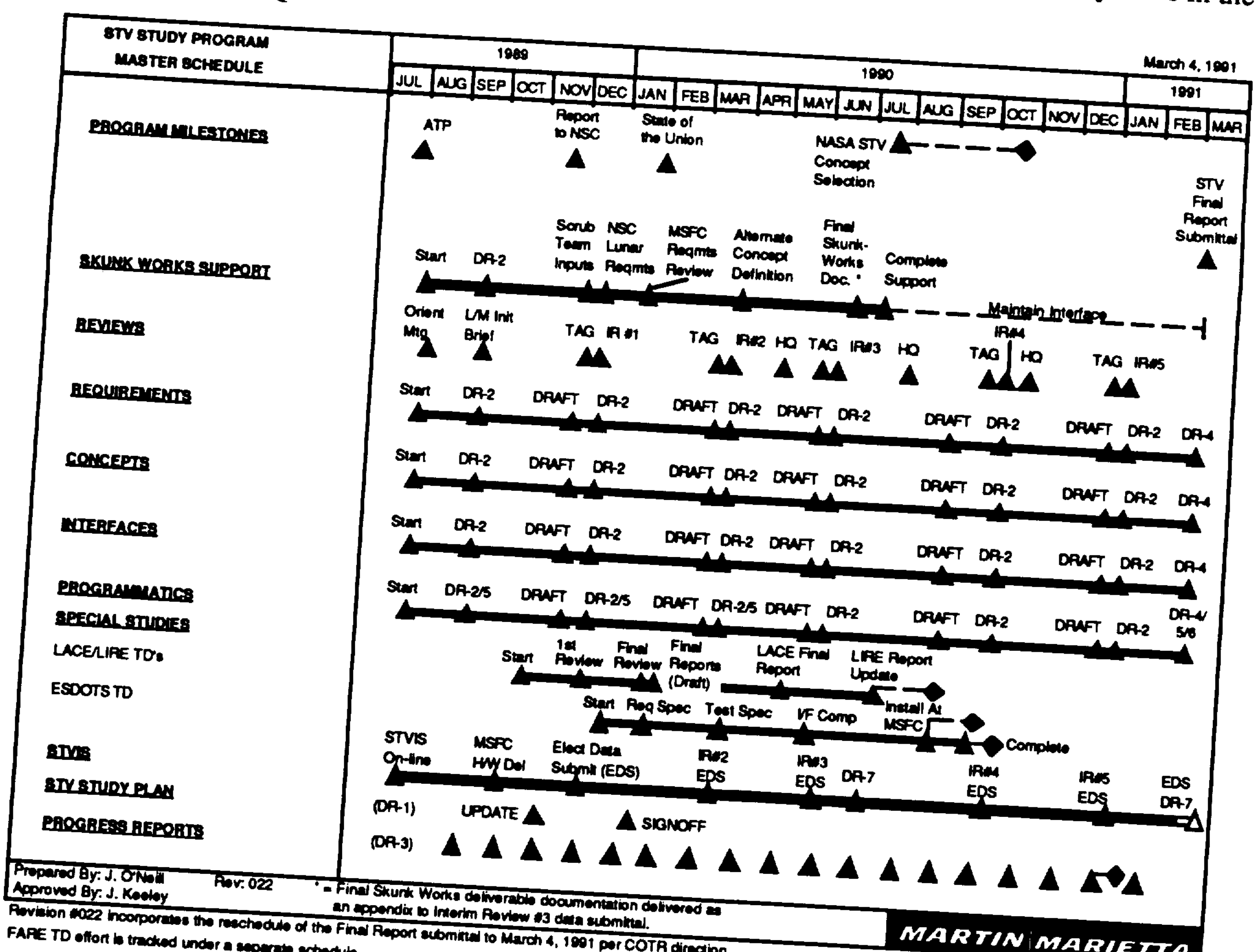


Figure 5.1-1 STV Study Program Master Schedule

The development of the summary phase C/D and phase E/F planning data was accomplished during this study phase. Based on the direction taken in the Space Transfer Vehicle (STV) basic defined tasks contract activities, detailed project logic network models were developed for the Lunar Transportation System (LTS) as the major emphasis and STV programs. The network models have been developed to the subsystem level, based on the current depth of conceptual maturity, and are directly traceable to the major work breakdown structure (WBS) element. Both the required critical path analysis and risk assessments have been accomplished and are documented in this final report. Incremental delivery of the project planning data has been accomplished with inclusion in the performance review documentation (DR-2) submittals at the quarterly Interim Review (IR) meetings held at NASA/MSFC.

5.1.1 Summary Master Schedules

The HLLV/STV Program Schedule (Figure 5.1.1-1) illustrates the interrelationship between the HLLV development program and the development program of an STV/HLLV upper stage. The HLLV schedule data reflects the sequencing of the anticipated major milestones for PDR, CDR, and test flight. The schedule then shows the time phasing requirements to implement an almost parallel program for an STV as an HLLV upper stage with the phased progression to the Lunar

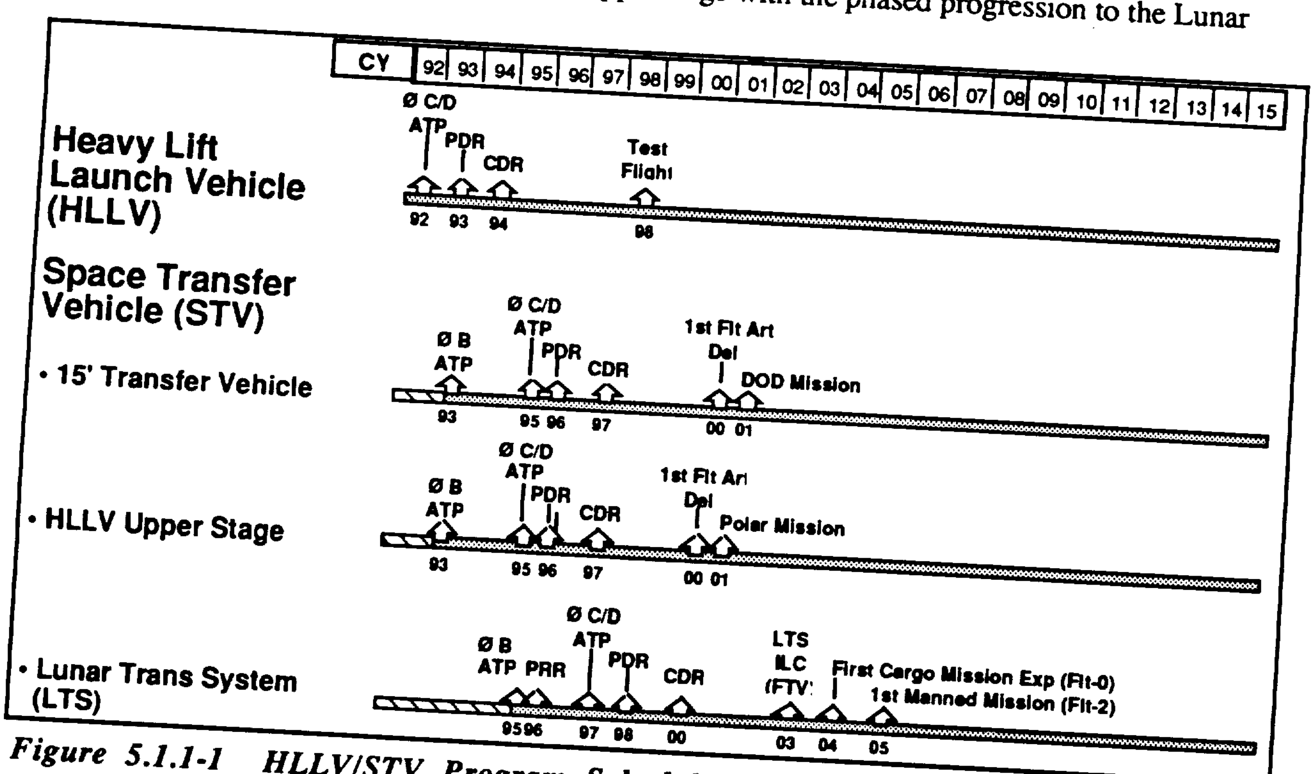


Figure 5.1.1-1 HLLV/STV Program Schedule

Transportation System (LTS). The fifteen foot diameter STV schedule is included to accommodate the interface for the Space Shuttle, an upgraded Titan IV, or other fifteen foot diameter payload class of vehicle as identified in the STV statement of work. The STV schedule for the fifteen foot diameter and the HLLV upper stage meets the early IOC dates for the NASA polar mission and the DoD missions from the CNDB-90. These STV systems are in service while the development of the LTS progresses through the first test flight launch in 2003. An expendable LTS cargo mission (payload unloader) to the lunar surface follows in 2004 and a reusable LTS cargo mission and the first piloted mission in 2005. This program phasing lowers peak funding requirements and provides integration of the mature STV design into the LTS. This sequencing also increases the ability to use common test beds and previous STV test articles through modifications and upgrades for LTS scenarios (schedule permitting) and provides early flight mission confidence using the STV prior to the LTS flights. The early STV flights will accomplish selected LTS test objectives and lower the development time, cost, and risk for the LTS program.

5.2 TEST PROGRAM

The STV/LTS test program has been developed to show an integrated approach of satisfying both the component and system test requirements of the ground and flight articles. To assure the success of this test program it has been divided into test phases which parallel the STV/LTS program phases B, C/D, and E/F. Figure 5.2-1 briefly describes each of these phases and the test intentions: a) technology verification and feasibility of STV/LTS design concepts during phase B; b) design development testing during phase C/D; c) component and system qualification program during phase C/D; d) systems level ground and flight testing during phase C/D; and e) acceptance and operational testing during phase C/D and phase E/F.

The STV/LTS phase B ground testing scenario has been established to provide technology verification and feasibility of design concepts. The main emphasis of this phase has been to address the technology/advanced development of the aerobrake, avionics/software, cryo-fluid management, cryo auxiliary propulsion, and alternative propulsion systems. This effort is further addressed in the technology/advanced development section of this final report via the roadmaps. The particular schedule driver, as it exists today, is the development of the "smart" aerobrake. Our test program has been established which requires the equivalent of an AFE II, whereby the LTS configuration aerobrake (although not full scale) is demonstrated using a "to be" scheduled STS flight in the 1997 timeframe. The development of the smart aerobrake also uses data gathered during the already scheduled AFE I, in the 1995 timeframe.

The following matrix represents the mission objectives accomplished by each flight article:

Mission Phase \ Test Article	Aeroassist Flight Experiment I	STV Flight Demo (Aerobrake)	Polar Servicing Mission (STV)	Flight Test Vehicle (FTV)	1st Cargo Flight (Flt-0)	2nd Cargo Flight (Flt-1)	1st Piloted Mission (Flt-2)
On-Orbit Assembly and Checkout			√	√	√	√	√
Rendezvous and Docking			√	√	√	√	√
Trans-Lunar Injection (TLI)				√	√	√	√
Descent				√	√	√	√
Ascent				√	EXP	√	√
Trans-Earth Injection (TEI)				√	EXP	√	√
Aeropass Maneuver	√*	√*	√	√	EXP	√	√

* Scaled Configuration Versions of Aerobrake - AFE I Mainly Dynamics Analysis/CFD Modeling, STV Demo Scaled Version of LTV Aerobrake (Rigid or Flexible Still To Be Determined).
Note: FTV To Be Reusable and May Require Refurbishment Prior To Next Usage.
EXP : Denotes That Unit Is An Expendable Unit

Figure 5.2-1 Mission Objectives Accomplished by Flight Article

The STV/LTS acceptance and operational test programs would be used to verify flight hardware performs in accordance with design and manufacturing documentation. STV/LTS test units will have an acceptance test performed verifying that the hardware is of known configuration (components, subsystems, and systems). The operational testing would consist of manufacturing in-line acceptance tests, systems operations testing (as practicable on ground and prior to LEO node departure), and launch processing tests (again as practicable at KSC and prior to LEO node departure). It is expected that much of the testing could and would be accomplished, via built-in-test (BIT) both at KSC and at the LEO node. Launch processing tests would include interface verification, RF verification, STV/LTS system functional, and booster integration and combined system test.

5.3 COST SUMMARY

Table 5.3-1 shows the STV top level cost by program phase and by major WBS element. It includes the production and launch of 22 vehicles with a LCC of $10,247.3 M. The DDT&E cost is $624.4 M, the production cost is $1205.4 M ($55 M average unit cost), and the operations cost is $8417.7.M.

Table 5.3-1 also shows the overall cost for the LTS program, including the production of 9 vehicles and launch of 25 missions, is $88,620.4 M. The DDT&E cost is $23,385.4 M, the production cost is $6,375.8 M ($708 M average unit cost), and the Integration and Operations cost is $58,859.2 M.

Table 5.3-1 Top Level Cost Summary

Element	DDT&E	Prod	Ops	LCC
Space Transfer Vehicle	451.8	871.9	6090.0	7413.7
Growth and Fee	172.6	333.3	2327.7	2833.6
TOTAL	624.4	1205.2	8417.7	10,247.3
Lunar Transportation System	16,918.7	4612.7	42,583.1	64,114.5
Growth and Fee	6466.7	1763.1	16,276.1	24,505.9
TOTAL	23,385.4	6375.8	58,859.2	88,620.4
STV/LTS TOTAL	24,009.8	7581.0	67,276.9	98,867.7

Costs Reported in Millions of 1991 Dollars

6.0 TECHNOLOGY/ADVANCED DEVELOPMENT

The objective of this task was to determine the technologies and advanced development concepts essential for the evolution of the next generation of lunar space transfer vehicles. The STV Technology and Advanced Development (TAD) effort has preliminarily identified the highest priority technologies and advanced concepts that are essential for the development of lunar STVs which can evolve into vehicles for Mars manned and cargo missions. In order to establish the status of each key TAD concept, development schedules have been defined for each area showing the current TAD maturity level and the existing/planned programs which will advance each TAD concept. A cost and performance benefits assessment is underway for each candidate TAD concept to quantify its value to the STV program. All candidate concepts will be prioritized and detailed development plans will be completed for those with the highest priority. A wide range of technologies have been identified and assessed to ensure the requirements for all STV concepts being evaluated are considered. All TAD concepts will be prioritized based upon their impact on STV cost, performance/safety and development schedule. Those that have a significant effect on any of these three criteria will be identified as "High" priority items. Those that have a moderate effect will be identified as "Medium" priority, and a "Low" priority will be assigned to those which have an insignificant effect on STV cost, performance or schedule. All the TAD concepts evaluated in this study will be listed according to their priority and a development plan established for the highest priority concepts.

Definitions of the seven TAD maturity levels illustrated in Figure 6.0-1 were derived from the NASA Space Systems Technology Model (January, 1984). They range from the observation of the basic principles (level 1) to an engineering model tested in space (level 7). To minimize program risk with resultant cost overruns, it is imperative that a maturity level 4 be reached by STV Preliminary Design Review and a maturity level of 6 (with 7 preferred) be obtained by the Critical Design Review (CDR), tentatively shown as the first quarter of 1997.
The twelve basic, top-level STV system requirements that drive the technologies and advanced development needs are summarized in Table 6.0-1. Although the first five listed have slightly more impact on almost all the major STV systems than the other seven, all twelve directly affect the selection of the key technologies and advanced development concepts.

Table 6.0-2 shows the ten key STV technology and advanced development areas essential for the development of lunar STVs that evolve into Mars vehicles. Early GEO vehicles will incorporate less advanced technology/development concepts and serve as test beds for the more advanced concepts required for sustained Lunar, Mars and planetary travel. In-depth development schedules have been prepared for each of the twelve TAD areas. These schedules show the current maturity level, the on-going programs (if any) that will be raising the maturity level, and the agency or program that is responsible for increasing the maturity. Only a portion of one schedule is shown here due to space limitations. Schedules for all TAD concepts are available upon request.

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

Technology Development

Advanced Development

Flight Systems

Figure 6.0-1 TAD Maturity Level Definitions

Table 6.0-1 STV Requirements That Drive Technology/Advanced Development

- Evolve For Mars Missions
- Manrated, Dual Fault Tolerant & High Reliability
- Withstand Space Environments, Long Duration
- Robust Design, Margins
- Minimum Space Assembly & EVA and No In-Flight Maintenance
- Cryogenic Propellant, 5 to 12 Months Propellant Storage
- In-Space Fluid Management & Transfer *
- Minimum In-Space Fluids
- Aeroassist GEO, LEO or Mars Return *
- Autonomous Rendezvous, Docking & Landing *
- In Situ Resources
- Low Life Cycle Costs and Acceptable Performance
- If Hardware Reused, 5 to 30 Year Service Life

* Not Required For All Concepts

Table 6.0-2 Key STV Technology/Advanced Development Areas

Area	GEO	Lunar	Mars
• Aerobraking	√	√	√
• Avionics	√	√	√
• Cryo Fluid Mgmt	√	√	√
• Cryo Space Engine	√	√	√
• Space & Ground Operations (Robotics, AI, etc.)	√	√	√
• Crew Module		√	√
• ECLSS		√	√
• Cryo Auxiliary Propulsion		√	√
• Alternative Propulsion		√	√
• In Situ Resources		√	√

To quantify the cost and performance benefits of each TAD concept, an analysis is being performed using the Zero Base Technology Concept (ZBTC) approach developed on the Advanced Launch System (ALS) program. In this approach, a reference ZBTC is defined and its Life Cycle Cost (LCC) and performance established. The cost and performance effects each TAD concept has on the ZBTC is then assessed. For our analysis, the Martin Marietta 90 Day Study vehicle reference concept was selected as the ZBTC. This reference vehicle was assumed to use existing technology and hardware such as RL-10A-4 engines, aluminum tanks and aluminum-mylar MLI. The non-recurring, recurring, and LCC for the ZBTC is shown in Figure 6.0-2. This analysis assumes five flights per vehicle.

When the cost and performance benefits analyses have been completed for each candidate TAD concept, they will be ranked against each other based upon the total LCC savings. To ensure each concept is assessed properly, data will also be derived as to the concept's total investment cost, recurring savings per flight, cost benefit (LCC divided by research and technology cost), and net present value for a 5% discount rate. All this information will be used to establish the "cost" ranking which will be integrated with the "performance" and "schedule" rankings to arrive at the high, medium and low priorities for all of the STV TAD concepts.

Results from the initial assessment of the TAD concepts show the potential high priority items to be aerobrake aerophysics; guidance/control and materials; avionics, power, software and fault tolerance system; cryogenic engine throttling and integrated modular engine; health and status monitoring; fault tolerance and space environmental effects. Our study results show that many of the potentially high and medium priority TAD concepts will not reach an adequate level of maturity to support the STV program without additional funding.

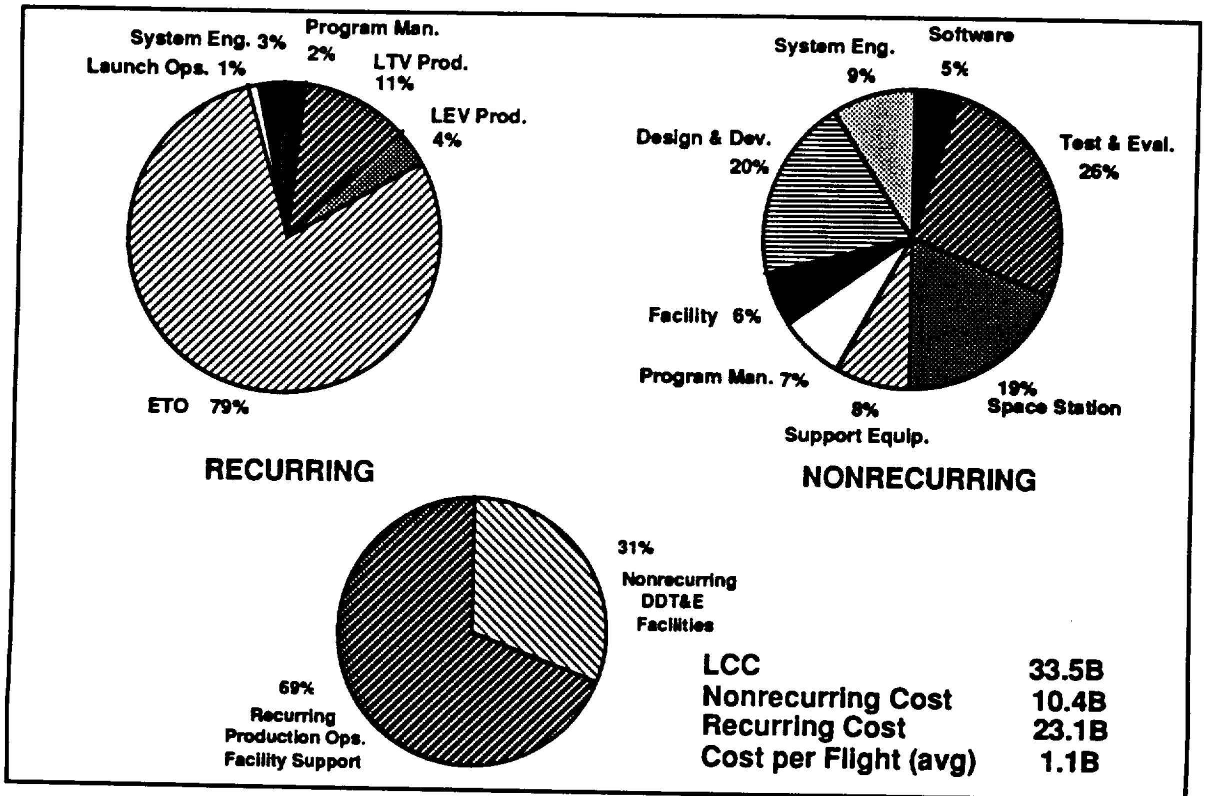


Figure 6.0-2 LCC of ZBTC: 90 Day Reference Configuration.