SPACE TRANSFER VEHICLE CONCEPTS AND REQUIREMENTS STUDY

Phase I Final Report
Volume I
Executive Summary
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FOREWORD

This final report of the first phase of the Space Transfer Vehicle (STV) Concept and Requirements Study was prepared by Boeing for the National Aeronautics and Space Administration's George C. Marshall Space Flight Center in accordance with Contract NAS8-37855. The study was conducted under the direction of the NASA Contracting Officer Technical Representative (COTR), Mr Donald Saxton from August 1989 to November 1990, and Ms Cynthia Frost from December 1990 to April 1991.

This final report is organized into the following seven documents:

Volume I EXECUTIVE SUMMARY

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

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

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

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

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1.0 INTRODUCTION

This section provides a description of the study in terms of background, objectives, and issues. Use of trade names, names of manufacturers, or recommendations in this report does not constitute an official endorsement, either expressed or implied, by the National Aeronautics and Space Administration (NASA).

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

The study objectives are to define preferred STV concepts capable of accommodating future exploration missions in a cost-effective manner, determine the technology development (if any) required to perform these missions, and develop a decision database of various programmatic approaches for the development of the STV family of vehicles.

Special emphasis was given to examining space basing (stationing reusable vehicles at a space station), examining the piloted lunar mission as a primary design mission, and restricting trade studies to the high-performance, near-term cryogenics (LO2/LH2) as vehicle propellant.

The study progressed through three distinct 6-month phases. The first phase concentrated on supporting a NASA 3 month definition of exploration requirements (the "90-day study") and during this phase developed and optimized the space-based point-of-departure (POD) 2.5-stage lunar vehicle. The second phase developed a broad decision database of 95 different vehicle options and transportation architectures. The final phase chose the three most cost-effective architectures and developed point designs to carry to the end of the study. These reference vehicle designs (two are illustrated in figures 1.0-1 & 1.0-2) are mutually exclusive and correspond to different national choices about launch vehicles and in-space reusability. There is, however, potential for evolution between concepts.

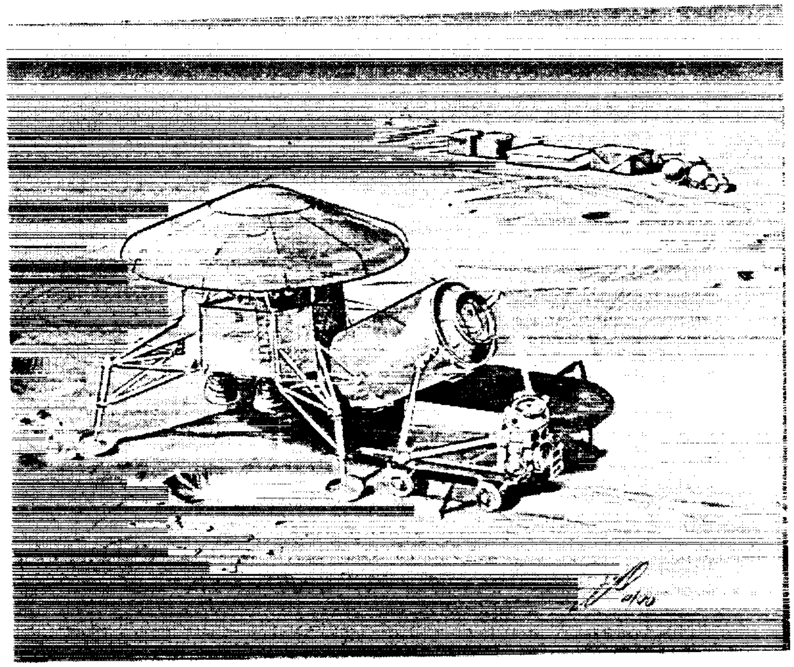


Figure 1.0-1. Highly Reusable Space-Based Lunar Vehicle Offloading Lunar Payload

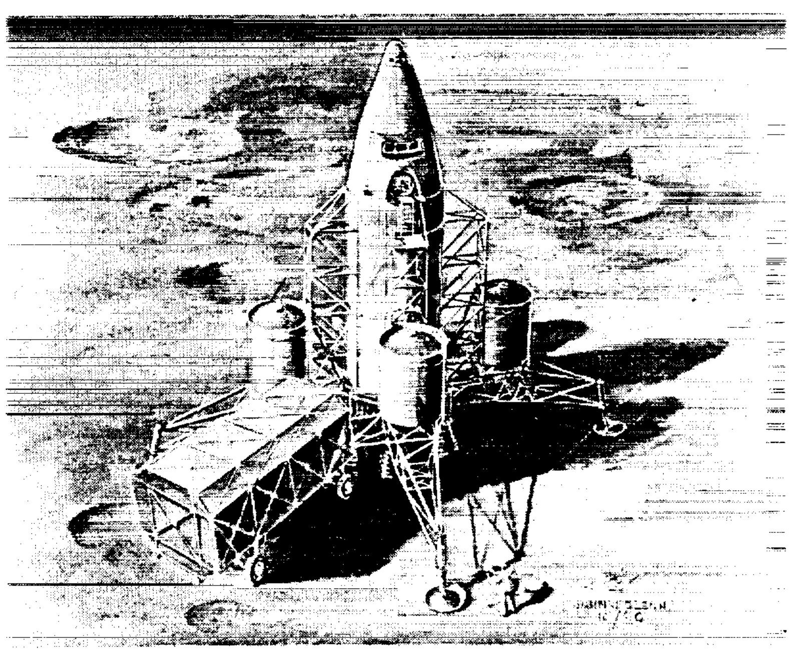


Figure 1.0-2. Ground-Based Lunar Vehicle Reuses Only the Crew Module

2.0 SUMMARY OF KEY FINDINGS, CONCLUSIONS, AND RECOMMENDATIONS

The study addressed a range of questions from very general (Why does the country need a transfer vehicle?), to subsystem specific (For what specific impulse should the engine be designed?). The key findings and conclusions from these questions are presented in the following sections.

2.1 KEY FINDINGS

Why and when do we need an STV? NASA needs an completely new upper stage to perform the next piloted mission, whether that be a lunar landing, an Earth orbit transfer to recover sample spacecraft, or a Mars transfer. Existing upper stage systems do not have the reliability/redundancy, design process, or performance capability to be a man-rated upper stage. In addition to NASA's next piloted mission, other high-energy national missions need to be performed that an STV could perform, rather than redesigning and significantly growing existing stages to accomplish those missions.

The time frame for the piloted missions are driven by national goals and priorities. Recent support for "mission to planet Earth" by the President and the Augustine committee indicates that a lunar mission could be launched by 2005 if the transfer vehicle was ready. Other national high-energy missions that occur before that date could use the vehicle in an unpiloted, protoflight mode. The Civil Needs Data Base (CNDB) is a collection of all these potential STV missions.

What are the missions that need an STV? The STV mission requirements fall into three main categories: unpiloted delivery, piloted or unpiloted servicing, and piloted or unpiloted lunar outpost support. The performance energy required for these three mission categories are shown in figure 2.1-1.

Unpiloted delivery missions are the shortest duration missions and require only two or three main propulsion system burns a few hours apart to place the

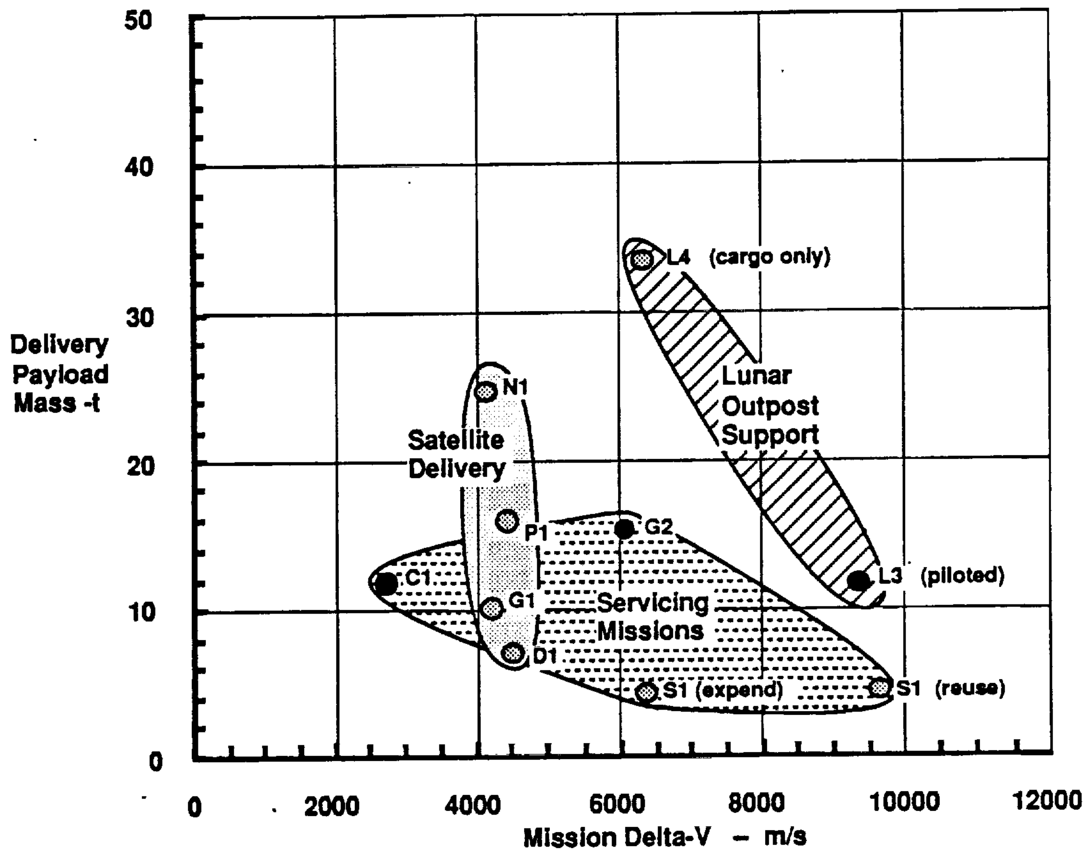


Figure 2.1-1. STV Mission Energy Requirement

payload in the final orbit. Most of these missions are to various Earth orbits (e.g., GEO and molynia) or to planetary boost trajectories.

Servicing missions require the vehicle to match orbits, rendezvous, and dock with a platform or spacecraft in LEO polar, GEO, or high-energy parking orbit. A high-value front end is carried round trip by the STV for these missions; sometimes being a piloted crew cab with arms or simply a telerobotic servicer with end effectors, spare parts, and consumables. In some cases the spacecraft contains samples that must be returned to Earth. These missions require a duration of 1 to 5 days and may involve significant teleoperation from the ground, in addition to the precise docking capability.

The lunar outpost support missions have the most demanding requirements in terms of duration, propulsive energy, and reliability/redundancy. The mission requirements call for operation in two modes: piloted reusable (transport crew and limited cargo from LEO to the lunar outpost and return the crew) and cargo expendable (transport cargo from LEO to the lunar outpost where the empty vehicle remains). The vehicle must be man rated, and the requirements specify two failure tolerance during all mission phases.

What should the space transfer vehicle look like? This study examined a wide variety of different staging concepts, launch vehicles, engines, crew modules, trajectory options, and technology levels. The goal was not to recommend one "best" vehicle, but to develop a decision database for a variety of infrastructure and technology conditions. This phase of the study did result in three "reference vehicle concepts" chosen which correspond to different national infrastructure requirements as described below.

Figures 2.1-2, 2.1-3, and 2.1-4 show the Earth-to-orbit (ETO) launcher requirements for the three vehicle concepts. The launch vehicle, more than any other factor, strongly affects the configuration of the concepts. The three concepts were developed expressly to optimally use the three ranges of launch vehicle capability under consideration by NASA and the Air Force. The space-based concept shown in Figure 2.1-2 provides the capability to mount a very large mission to the Moon with a very modest ETO launch capability. The lunar transfer vehicle is assembled at the Space Station from elements launched in shuttle and shuttle-derived type launch vehicles. Transfer stage reusability and other benefits of space basing are emphasized, and the crew module and most of the stage remain at the Space Station between flights.

The ground-based concept shown in Figure 2.1-3 is similar to the concept Apollo used to launch to the Moon. A single very large (twice the size of Saturn V) ETO launcher boosts the vehicle directly into an Earth phasing orbit for the translunar injection burn. This very large launcher, like Saturn V before it, would be developed solely to perform the lunar and later Mars missions. The range of ETO payloads represented by these two concepts spans from 70 (metric) tons for space-based concept to 225 tons of launcher capacity for the ground-based concept. The third concept, which fall between these two concepts, is a ground-based Earth orbit rendezvous (EOR) concept that requires a launch vehicle with about 120 tons of capacity (Figure 2.1-4). The transfer vehicle in this case is similar to the ground-based case, except the

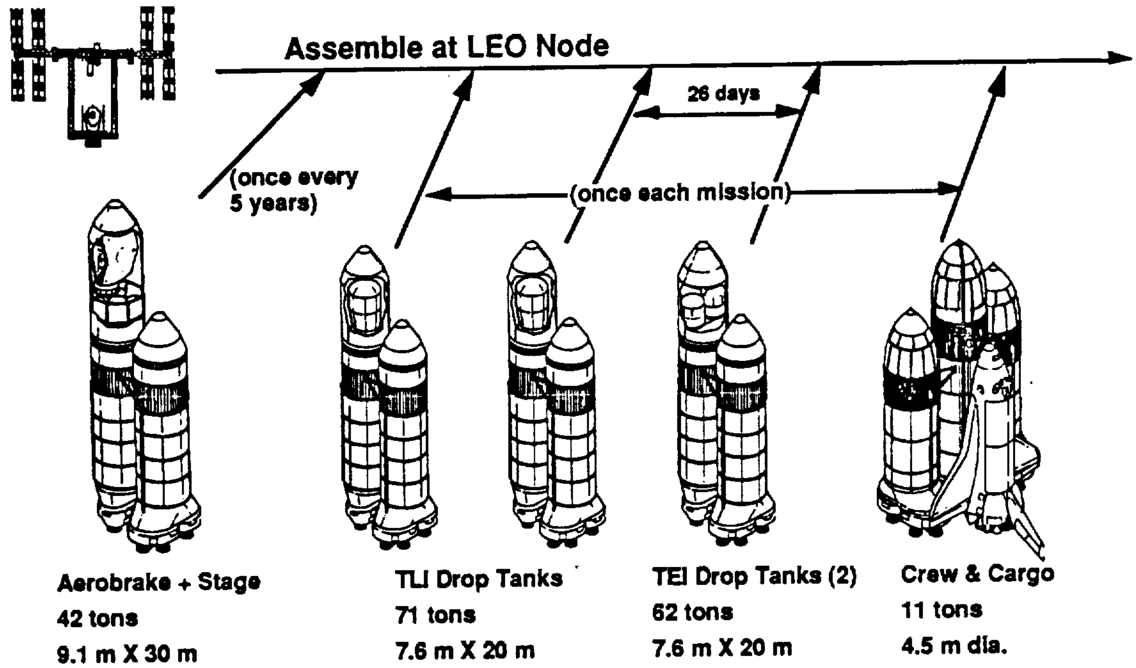


Figure 2.1-2. Space-Based Earth to Orbit Concept

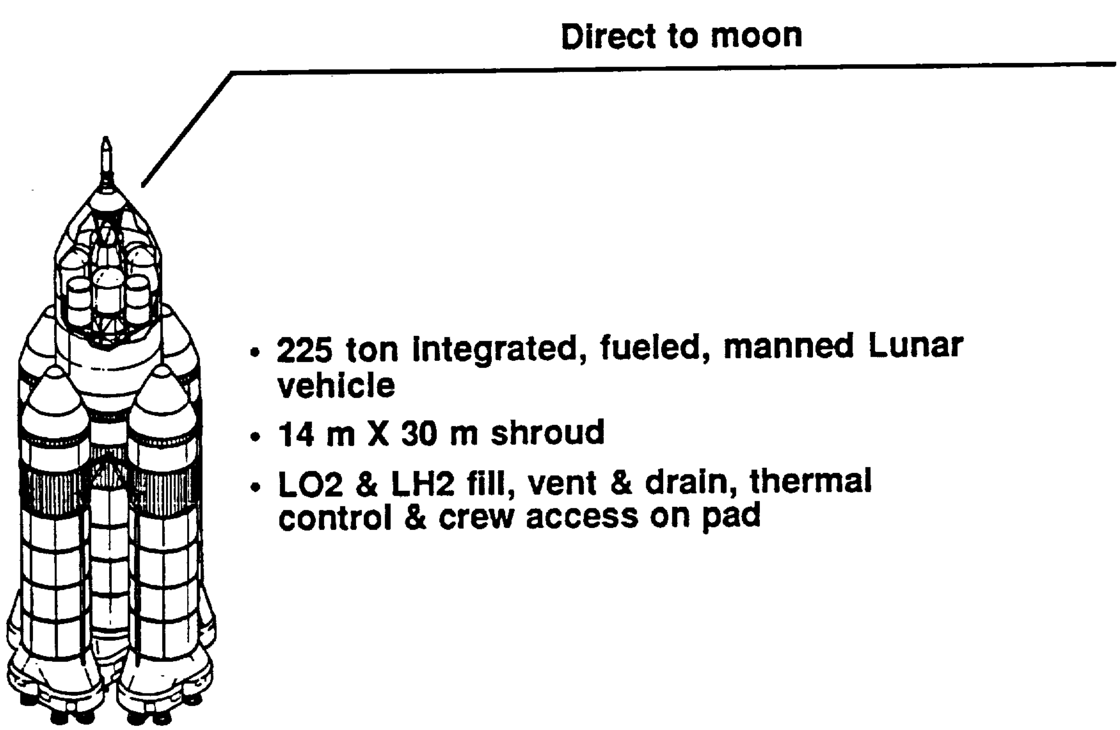


Figure 2.1-3. Ground-Based Launch Concept

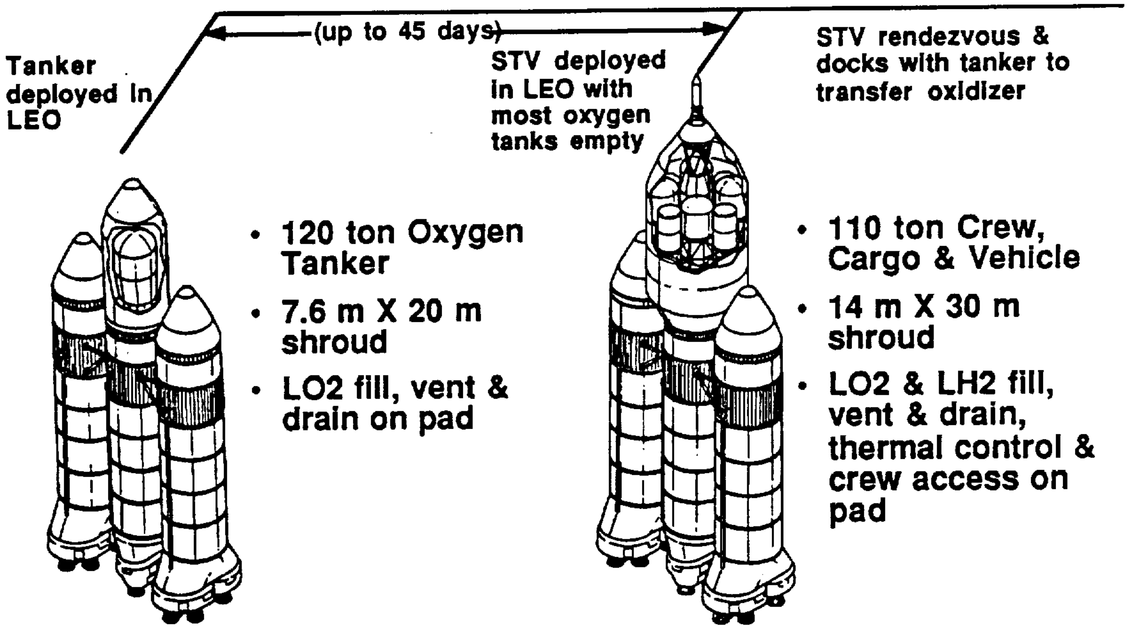


Figure 2.1-4. Ground-Based Multilaunch ETO Concept

oxygen propellant is offloaded and boosted to orbit on an earlier flight to reduce launcher requirements. No on-orbit integration of flight elements is required in the EOR concept. Only rendezvous and docking with the oxygen tanker is required, as shown in Figure 2.1-5.

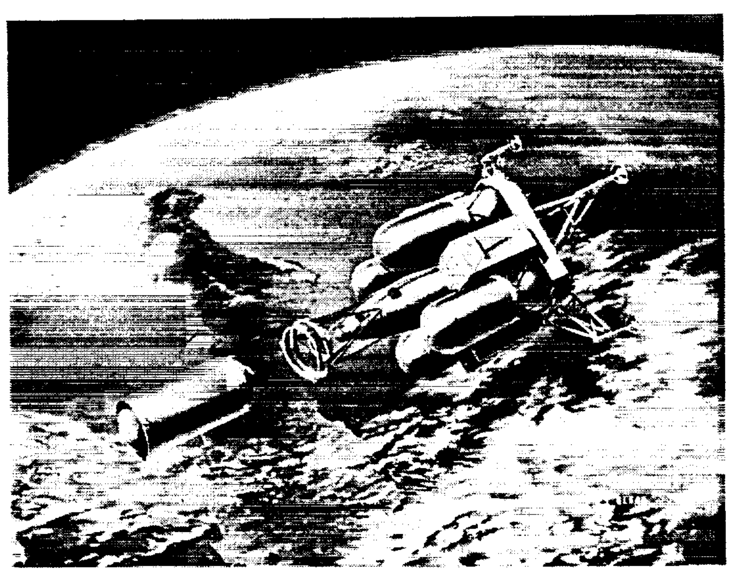
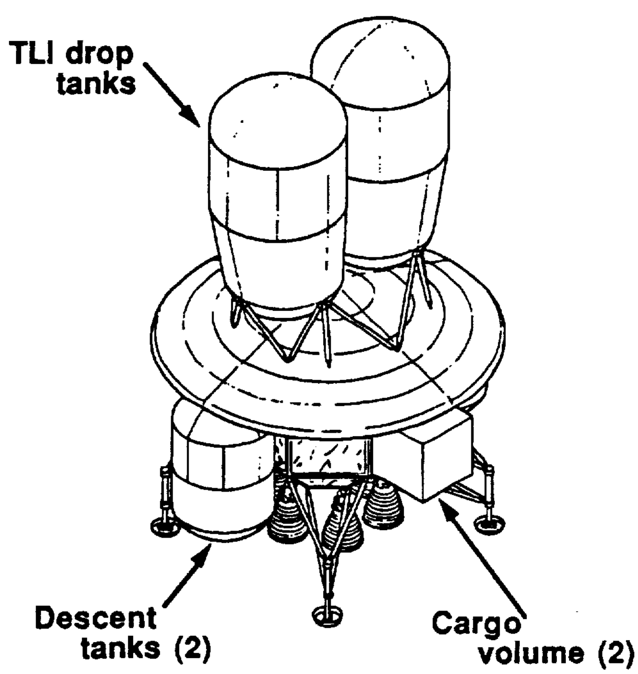


Figure 2.1-5. Ground-Based Vehicle Docking With Oxygen

Figure 2.1-6 shows the major components and features of the space-based reference concept, and Figure 2.1-7 shows the common features of the two ground-based reference concepts. The ground-based concepts are similar at this configuration level; the only difference is that the oxygen tanks are launched empty in the multilaunch concept, which requires an orbital tanker.

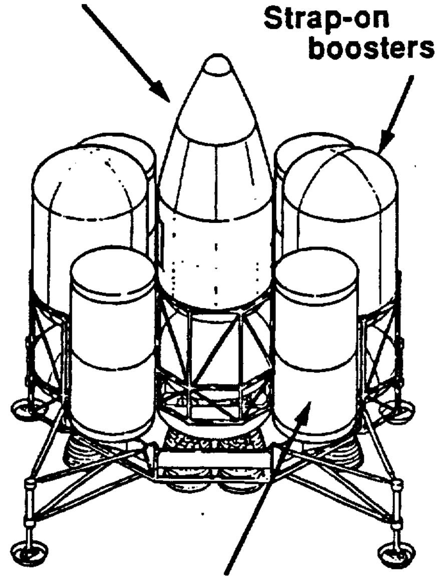
A common denominator throughout the study was the mission model; the study did not depart from the civil needs database (CNDB) FY'89 and the lunar option 5 of the NASA 90-day study. (Sensitivity to different flight frequency and varying payload size will be examined in the next phase of this study). A second common denominator was the use of a consistent set of trade study criteria used for all architecture analyses: cost, margins & risk, non lunar mission capture and benefits to the Mars Program. These evaluation criteria are discussed in additional detail in section 3.2.



Features

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

Figure 2.1-6. Space-Based Vehicle Concept



Features

- 13.7 m dia launch shroud
- Six 15klb thrust engines
- Cargo envelope: 4.5 m dia x 12.2 m (piloted), 4.5 m dia x ∞ (expendable)
- Self unloadable
- Crew module land (returns to ground) recovery
- Strap-on boosters usable as separate upper stage

Figure 2.1-7. Ground-Based Vehicle Concept

Can a single vehicle "family" satisfy the STV requirements? Yes, elements of a single vehicle designed for the lunar mission can satisfy the propulsive requirements of all potential missions in the database except for the piloted Mars transfer. (Mars transfer requires an order of magnitude higher propulsive energy, and two to four times longer mission duration. Mars ascent and earth reentry vehicles, however, could strongly utilize lunar systems.)

An example of vehicle elements "capturing" these mission classes is shown in Figure 2.1-8, where the ground-based vehicle is broken into main elements. Corresponding performance for the booster strap-on and the complete vehicle is shown in relation to the mission performance requirements. The booster propellant load allows it to capture all the delivery missions and about half of the servicing missions. The complete vehicle captures the remaining servicing missions and the lunar missions. Thus, the booster strap-on developed for lunar orbit insertion and lunar landing can be used as a standalone element (with the addition of the avionics/RCS pallet) to capture more than half the missions in the CNDB mission model.

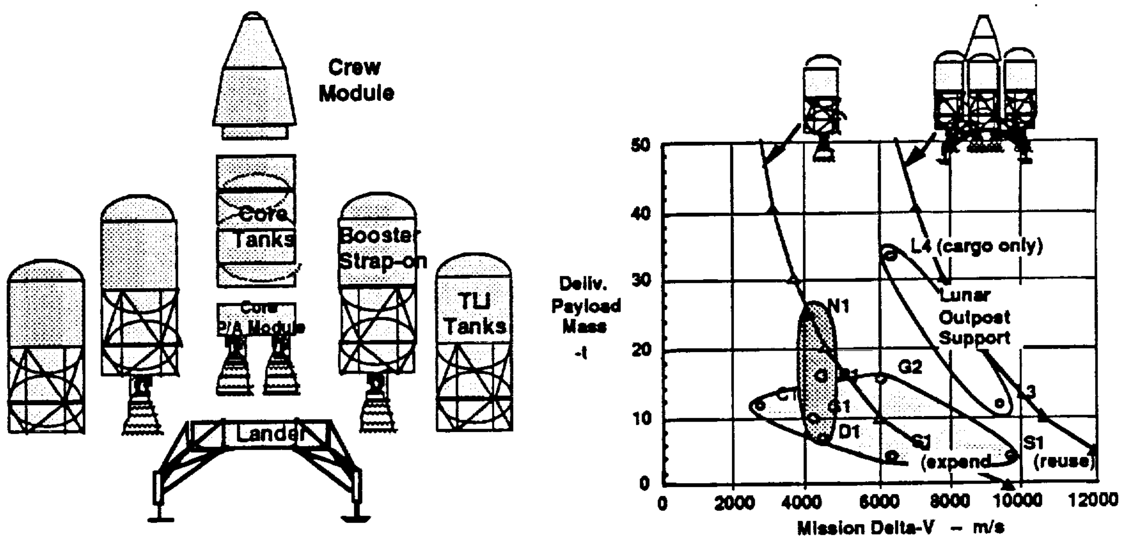


Figure 2.1-8. Ground-Based Concept Elements and Performance

Why don't we do the lunar mission like Apollo? The Apollo program goals and objectives are significantly different than the STV goals. President Kennedy stated Apollo's goal when he said, "...to land a man on the Moon and return him safely; and do it within this decade." The STV goal, which is much more operationally oriented, is "to provide a cost-effective transportation system capable of supporting an exploration program resulting in a manned outpost on the Moon." The mission requirements comparison between Apollo and STV in Figure 2.1-9 also show wide differences. The STV payload is larger by a factor of between twenty to fifty, surface duration is increased by two orders of magnitude, and subsystem component failure tolerance has increased for some systems.

The Apollo program consisted of a three-stage launch vehicle and a three-stage transfer vehicle (stage 3 (SIVB) could be called either). Apollo had a crash program schedule which attempted to land a crew and small payload on the Moon with as short a development time as possible. To meet the important schedule requirements the program utilized launch vehicle (C-series) & engine (F-1, J-2) development programs already underway, and operated without an orbital infrastructure (i.e., space stations) that could have been used for assembly - which put distinct performance limitations on the design. The Apollo

Requirement:	Apollo	STV
Crew on Lunar surface	2	4-6
Mission duration	12 days max	200-365 days
Duration on lunar surface	3 days max	190-360 days
Cargo mass to LS	0.7 tonnes	13/32 tonnes
Cargo returned to earth	0.2 t	0.5 t
"Engine Out" and return	0*	2 failed and rtn

* LEM ascent burn and CSM TEI burn

Duration, cargo mass delivered, and MPS failure tolerance requirements are significant differences between STV and Apollc

Figure 2.1-9. Apollo-STV Requirements Comparison

design was thus schedule and performance optimized, with less regard to program cost.

The STV design was cost and margins and risk optimized, which resulted in a reduced number of stages and eliminated splitting up the Earth return vehicle into a lander and a lunar orbiter. Both of these design decisions reduce cost, but increase the propellant required to fly the mission. It was determined that launching the extra propellant is cheaper than designing the vehicle to fly like Apollo; even when the propellant requirements are almost twice as high.

The STV is an all cryogenic vehicle, like the space shuttle. Apollo used storable MMH/NTO for the descent and ascent engines. Cryogenic fluid management was not as well understood as it is today and was not used for landing because of uncertainty in system reliability and boiloff on the lunar surface. The STV uses a high-performance cryogenic engine (ASE or RL10) and includes in the development program sufficient testing to demonstrate required reliability of engines and insulation. The vehicle additionally carries extra engines to provide sufficient margin in the case of engine failures.

How sensitive are the designs to changes in the mission requirements? Any design (including these specific vehicle designs) is quite sensitive to mission requirements changes if these changes affect the basic selection criteria. Key mission requirements that are very design sensitive include payload mass, the program goal to be cost effective (i.e., cost is a selection criteria), and the redundancy requirement for all life critical systems (two failure tolerance). Significant changes in these three factors would cause significant changes in the design and operations of the vehicle.

For example, the lunar vehicle has six 15-20k lbf engines, four are optimum considering the gravity loss curve for the most important first burn. In addition, a conservative reading of the two failure tolerance requirement requires the ability to land on the Moon (abort to surface rule) after two engines have failed. Starting with at least six engines allows the vehicle to land safely with two engines out; the two opposing engines are shut down and the two remaining engines allow vehicle center-of-gravity capture and vertical attitude touchdown. (The large landing mass of the reference concepts (126,000 to 198,000 lb) also requires the thrust of a minimum of two engines just to hover.) The abort to surface rule was chosen on a cost minimum basis because of the high-value payload and the once-yearly flight. One very large engine could have been the answer, if cost and redundancy were not included in the selection criteria. If the operational flight rule was made to never land with more than one engine out, a four-engine vehicle would be optimum.

Other mission requirement changes would have less of an effect on the vehicle system. Changes in duration or crew number would change the size slightly. A viewing requirement to observe the two front landing pads and the horizon simultaneously definitely restricts the landing configuration. The launch escape requirement necessitates the crew module location atop the stack in the launch vehicle shroud; there are a series of requirements that could change the system design slightly if they were modified. Mission requirements are discussed in detail in Volume II, Book 2, and concept response to those requirements are contained in Volume II, Book 1.

How sensitive are the designs to changes in the launch vehicle? The launch vehicle selection is critical to the transfer vehicle program. During the study it was recognized that the existing launch vehicles (STS and Titan IV) were not large enough or economic enough to support a major new initiative such as a lunar outpost. It is a fact that a new launch vehicle of some kind (larger and more economic) would be required. The STV designs presented above provide for any of three major classes of vehicles being available: (1) a vehicle launching 40 to 71 tons, with a payload envelope at least 15 to 30 feet in diameter; (2) a heavy lift vehicle capable of launching 120 tons, with an envelope 45 feet in diameter; and (3) a very heavy lift vehicle, which will probably only be used for lunar and Mars support, capable of launching 225 tons, with an envelope 45 feet in diameter.

Fundamentally, if the launch vehicle decreases in capability one of three things must happen: either the mission payload decreases, the transfer vehicle is broken into increasingly smaller pieces and assembled in orbit, or the mission design or propulsion technology is changed to give greater value to higher performance at the "expense" of making the program cost more. As to the first result, throughout the study payload to the lunar surface was held constant and not compromised. (Future studies will examine the effect of decreased payload in the early phases of the lunar program.) The three concepts required varying amounts of on-orbit assembly; the largest booster system requiring none to the space-based concept requiring five launches before the first flight was fully assembled.

Having a fixed launch vehicle with a "not to exceed" payload capability would change the nature of the study from a clean sheet design to a performance constrained design, which would probably affect some subsystem technology choices and would certainly result in a higher cost program. The magnitude of increased costs in a performance constrained case will be examined in the next study phase.

How sensitive are the designs to changes in the space base infrastructure? Changes in the space base could affect that vehicle's design, but only slightly. The vehicle is already designed to be as autonomous as possible. The aerobrake is designed to be deployable and therefore not requiring EVA astronauts to assemble or verify connections. The drop tanks and core stage are designed to be assembled using a remote manipulator (potentially ground controlled) and checkout is performed by onboard systems (this checkout must be done again after 6 months in lunar orbit or on the Moon, so it must be an onboard function).

The Space Station could still change significantly without requiring much change in the transfer vehicle. Some node is required in the space based design to provide 1) a stable structure on which to base vehicle assembly, integration and fueling, 2) provisions for crew ingress and final checkout, 3) a meteoroid/debris-free storage environment, and 4) hardware required for repair/refurbish operations. Figure 2.1-10 shows a variety of node options and operational capabilities. Vehicle interface requirements are discussed in detail in Volume II, Book 3, and the node accommodations to those requirements are contained in Volume II, Book 1.

What type of propulsion system is most desirable for a new STV? One of the ground rules of the STV statement of work was that "only cryogenic LOX/LH STV concepts will be considered." Within that ground rule two general alternatives presented themselves: a derivative of the Pratt & Whitney RL10 engine or the advanced space engine (ASE), which is a new engine under study by the NASA-Lewis Research Center (LeRC). The RL10 was developed 30 years ago and continues to be used on all the Centaur derivatives. The engine was considered for lunar excursion module (LEM) propulsion in the 1960's when a cryogenic LEM was still under study and significant throttling tests proved the engine could be modified to perform the landing mission. The

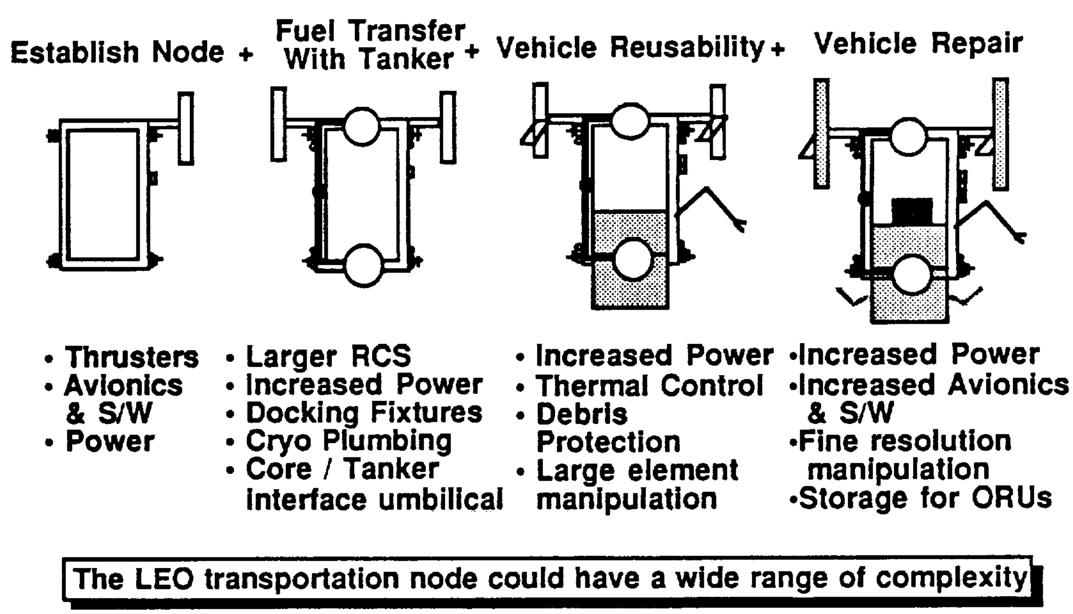


Figure 2.1-10. Varying LEO Node Requirements

ASE is a clean sheet design that is currently in a brassboard testbed stage. This engine is being designed for space maintenance and throttleability. The ASE would be smaller, more efficient (higher Isp), and lighter weight than the RL10, but cost more ($1 billion and more than 3 years) in development.

The ASE was chosen for the lunar transfer vehicle design because it provided the lowest life cycle cost over the life of the program, and the size permitted better integration of the six engines required to meet the strict redundancy requirement interpretation. If this requirement is relaxed or reinterpreted, if launched mass is not a constraint, or if front-end funding is more important than lowest life cycle cost then the RL10 engine would be a fine choice for the main engine.

What is the value and role of aeroassist to the program? In the broadest sense, both ground- and space-based concepts use aeroassist systems in the final velocity change maneuver. Like Apollo and the shuttle, the ground-based concepts use a heat shield and the Earth's atmosphere to slow down to terminal descent speeds for an Earth landing. The type of thermal protection system (TPS) chosen (from single use ablators to transpiration cooled ceramics) is a function of aerodynamic characteristics, development cost and turnaround requirements. All concepts that reenter to the Earth's surface must use the atmosphere to slow down.

The real trade of whether or not to use the Earth's atmosphere to slow down comes in the space-based concept. The choices are between carrying the additional propellant to perform one more burn in low Earth orbit (LEO) with the same engines used for the other 3 to 5 mission burns or to carry an aerobrake (Figure 2.1-11), which is a TPS covered structure that transfers the air loads and shields the rest of the vehicle from the high temperatures of the aeromaneuver. The aerobrake, therefore, is one more element of the main propulsion system and must be evaluated on its propulsive efficiency.

Previous studies have examined various types of aeroassist devices, from lightweight inflatable ballutes, to fabric-covered deployable umbrella aerobrakes, to rigid tile covered shells. Early in this study, during the 90-day

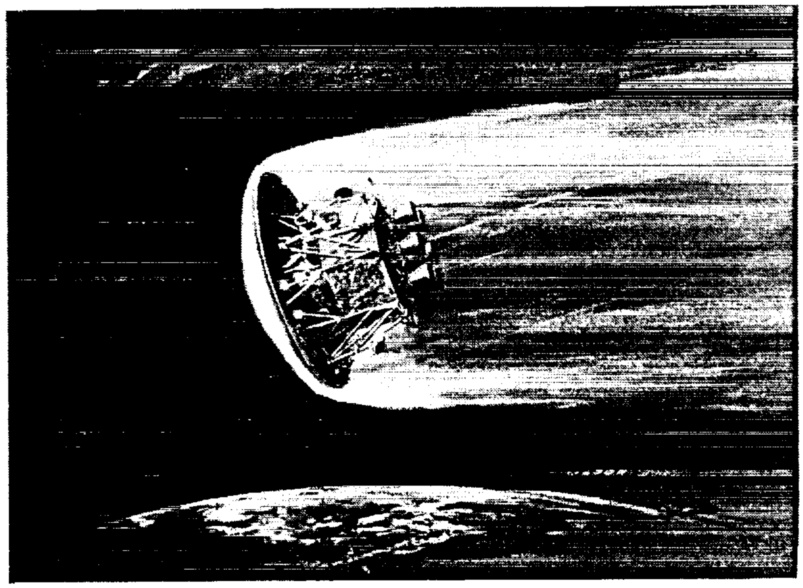
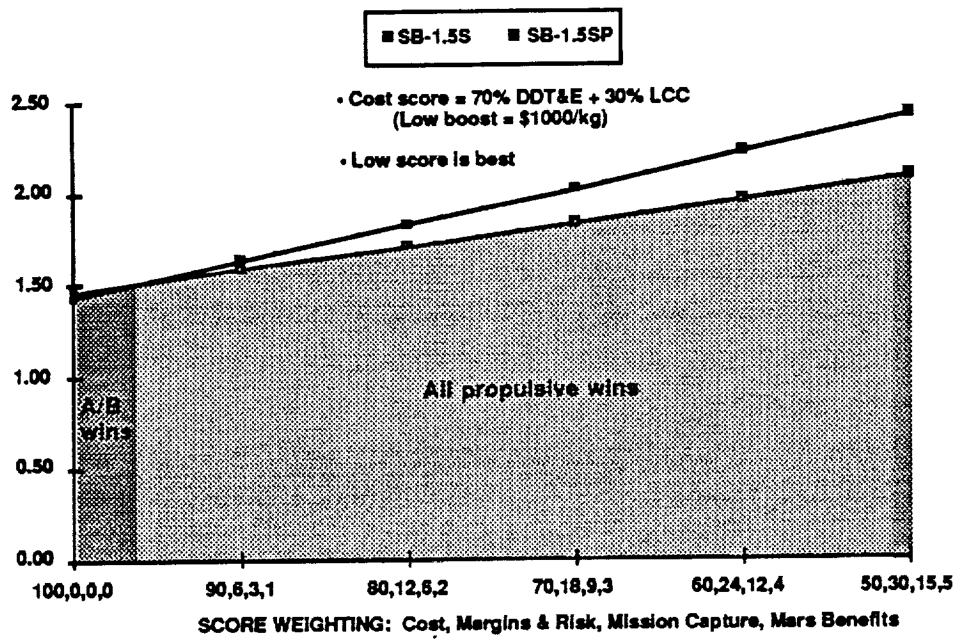


Figure 2.1-11. Aeromaneuver in Earth's Upper Atmosphere

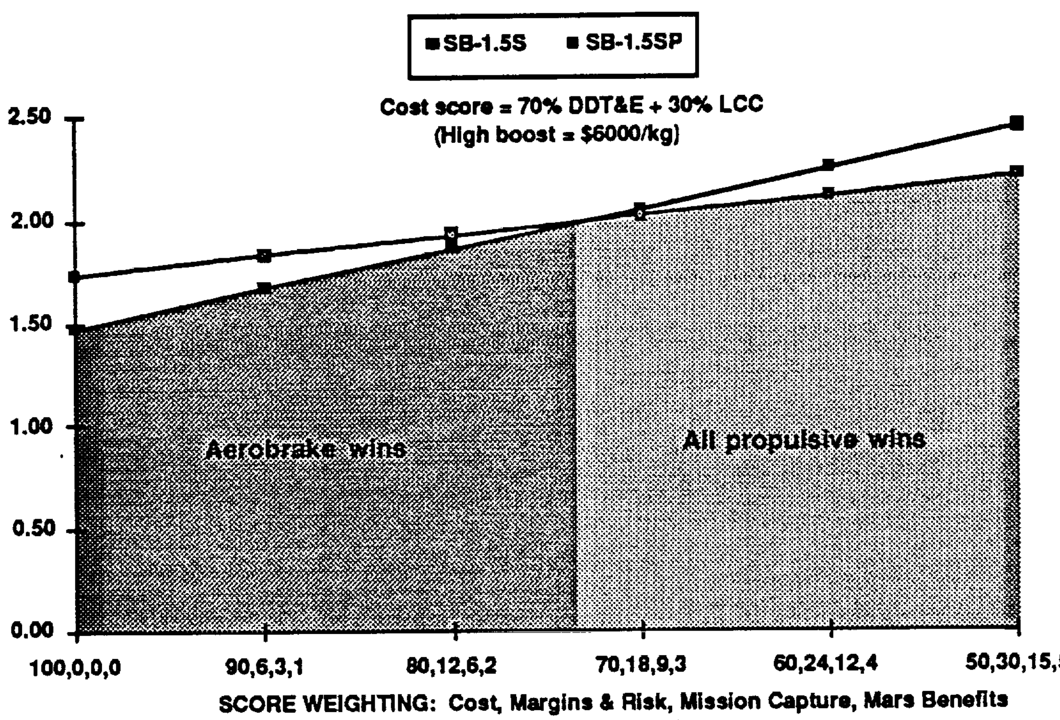
exercise, the MSFC inhouse team chose the rigid shell as the aeroassist device of choice. The shell is the lowest risk design because it has the best understood structural and TPS concepts, with test plans already in place (the aeroassist flight experiment (AFE) will test this concept in the mid-1990's). While the rigid shell provides a lower risk development and operation for piloted missions, it is the most massive of the three aeroassist device choices mentioned above and thus the least competitive from a performance standpoint.

When comparing even this heaviest of aerobrake designs with either all-propulsive options, the aerobrake will win the life cycle cost analysis. This means it saves more money in reduced propellant boosting costs than it did to develop and test, and it saves at a greater rate than either engine option. If the choice was made on life cycle cost alone, the aerobrake would beat the all-propulsive options. However, as described earlier, there were four evaluation criteria to score options against: cost (development and life cycle), margins and risk, other mission capture, and benefits to the Mars program. The aerobrake wins in the life cycle cost and benefits to Mars categories, loses in the development cost and margins and risk categories, and provides no additional mission capture benefits. Thus, depending on how these criteria are weighted and what it costs to boost propellant to orbit, either aerobraking or all-propulsive options could be the best answer. Figures 2.1-12 and 2.1-13 show the effect of



Aerobrake wins when cost is used exclusively as the selection criteria

Figure 2.1-12. Low Boost Cost Effect on Aerobrake Selection



Aerobrake wins when cost is > 70% of the selection criteria

Figure 2.1-13. High Boost Cost Effect on Aerobrake Selection changing the weighting mix of evaluation criteria for low boost cost and high boost cost, respectively. As expected, when the cost score is emphasized aerobraking has the lowest score (wins); and when launch vehicle boost costs are higher, aerobraking wins more often. (This same relationship is seen just as strongly with an increasing Mars benefit score; for more trade study data see Volume II, Book 2, System Architecture Study.)

Should the STV be based on the ground, in space, or use both modes? There are two main reasons for establishing a transportation node in LEO: to decouple the transfer vehicle mission from the launch vehicle capacity and to enable reuse of more transfer vehicle components than can be economically reentered to the Earth's surface. Regarding the first reason, this study has shown that a space base is advisable if launch vehicle capacity is limited to 70 tons and the cargo delivered to the Moon is greater than 30 to 40 tons per mission. However, if a larger launch vehicle exists or the cargo can be taken in smaller elements, then a LEO node is not absolutely required.

Space basing may, however, still be advisable if the lunar missions are frequent enough to make core stage reuse more cost effective. At less than one reusable flight per year, even the ground-based plan of recovering only the crew module requires a new module only every 5 to 10 years. Building a new stage every 5 years is not very cost effective, and building the entire fleet plus extensive stores at one time entails programmatic risks.

The next phase of this study will examine other lunar cargo capacities and mission frequencies. Use of the lunar vehicle core for non-lunar servicing missions showed promise for space-based concepts when those servicing missions were in the mission model. The latest edition of the civil needs database (CNDB '90) removes all servicing missions that could have used the lunar transfer vehicle core stage.

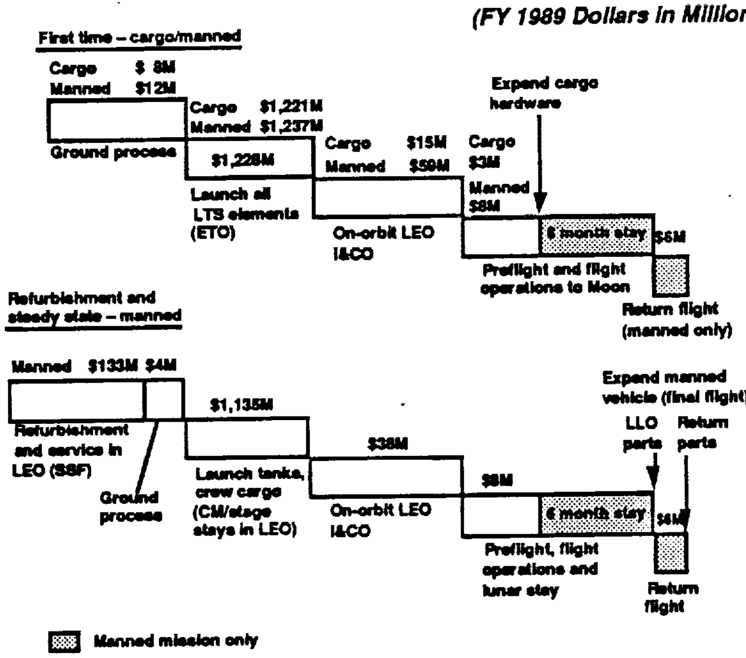
In summary, a space-based concept is required if the lunar mission definition remains a single large flight once per year and the largest launch vehicle capacity is in the 70-ton range. If either of these conditions changes, space basing may still win economically if there are sufficient missions to support vehicle reusability.

How cost effective is stage reusability for the space-based STV missions? There are four major cost factors that must be examined to assess the cost effectiveness of reusability: development costs, production costs, launch costs, and refurbishment costs. In-space reusability strives to reduce production and launch costs, but must endure slightly higher development costs (design for reuse) and incurs a refurbish cost that single-use systems do not. We would expect that reusability is more attractive when Earth-to-orbit launch costs are more expensive.

In reality, reusability of the lunar transfer vehicle elements do not save significant production costs at the current mission model rate for reasons discussed in the previous question. For even a modest number of four reuses, the mission model rate of around one flight per year requires production of a vehicle only once every 5 to 6 years, which is a very inefficient method of maintaining manufacturing capability. Alternately, building the six vehicles and spares required over the 30-year program life would mean using the equivalent of a 25 year old mothballed Gemini capsule for orbit or reentry operations today. Using hardware whose production line shut down 25 years before would be too great a risk to be a realistic option. (However, given a continued production demand, a system such as Atlas Centaur with decades old design and technology can still remain a viable program.) So, the main factors to be examined in the reusability trade for the reference mission, is launch costs versus refurbish costs.

Launch costs of a space-based reusable lunar transfer vehicle are indeed lower than an equivalent single-use system. (This is not true for pure payload delivery missions, as there is no reuse.) The core stage (and aerobrake) only need to be launched from Earth every 5 to 6 years. An example of this delta launch cost is shown in Figure 2.1-14, with the first time or expendable "ETO" launch costing $102 million (9%) more than the reusable case ($1,237M -$1,135M). For the reusable case, additional flights are required to launch orbital support hardware, such as vehicle debris protection enclosures and additional space-based reboost propellant, but the net balance is in favor of the reusable system on a launched mass basis over the life of the program.

(FY 1989 Dollars in Millions)



Summary (by mission type)

	Cargo	Manned
LTV	$ 26M	$ 27M
CM	0	58
Subtotal	$ 26	$ 85
ETO	1,221	1,237
Total*	$1,247M	$1,322M

	Manned only
LTV	$ 134M
CM	55
Subtotal	$ 189
ETO	1,135
Total*	$1,324M

* Note: Dollars exclude fee and NASA program support factors (at 10% and 5%) and mission control costs

Figure 2.1-14. O&S Flight Cost Buildup – SB2-1.5S

The refurbish cost of a reusable, space-based system includes two major elements: facilitization and operations. All space-based concepts require some assembly on orbit, which requires facilities such as debris protection, assembly fixtures, and remote manipulation systems. The additional facilities required for "turnaround operations" to refurbish used equipment for reflight may not be significant depending on the design philosophy established. In the built-in redundancy or even the "remove and replace LRU" philosophy there is very little extra equipment required for refurbish operations over what would normally be required of on-orbit test and checkout after spacecraft assembly. The extra operations is the main cost element. The study has estimated turnaround (refurbishment) operations at 864 labor-hours, which would cost $133 million. The crew module requires the most refurbishment between flights (done IVA), and for reusable systems these operations are performed (expensively) on orbit by a few astronauts, instead of on the ground by numerous technicians.

As Figure 2.1-14 shows, two major cost elements affecting reusability (launch costs and on-orbit operations and support costs) essentially cancel one another. If production costs are removed as an issue because of the very low flight rate, there is no cost advantage or disadvantage to reusing space-based vehicles.

What kind of program schedule is appropriate? The current mission model calls for lunar missions and other delivery and servicing missions to begin in the same year. With this mission model there is no opportunity for vehicle "evolution" or phased development of various vehicle elements (i.e., developing the booster first and using it as a protoflight testbed before lunar flights commence). The development program is shown in Figure 2.1-15. Note the risk reducing technology development effort starting early in the program with cryogenic fluid management, propulsion, and aerobraking "national test beds". Other program options are being examined in a future phase of this study that either have no other missions to capture besides the lunar mission (other Air Force or NASA vehicles are developed for the other missions) or the lunar mission is accomplished gradually, slowly increasing the amount of cargo to the lunar surface or gradually increasing the functionality of the vehicle from purely unpiloted to piloted over several years.

Does this program minimize development costs? Minimizing development and life cycle costs was given high priority in the trade studies. Final vehicle concepts were selected because they reduce these costs, while achieving other overall objectives, such as developing technology and infrastructure for eventual Mars programs. However, if the vehicles had been designed with only front-end funding minimization as the sole criteria, the new advance space engine (ASE) and aerobraking would have been delayed or eliminated.

The ASE provides greater performance (Isp) and lighter weight and can be designed for space-based maintenance. However, the engine will cost $1 billion more in development than modifying the existing cryogenic engine (the RL10). If launch vehicle size restrictions require the greater performance (ASE provides for 5% to 9% lower initial mass in low Earth orbit (IMLEO) than the RL10), then the ASE would be required to boost the annual 41 tons of cargo to the lunar surface. Most likely the cargo capacity would be lowered at the outset and improved when the ASE eventually replaced the RL10 engine. If the

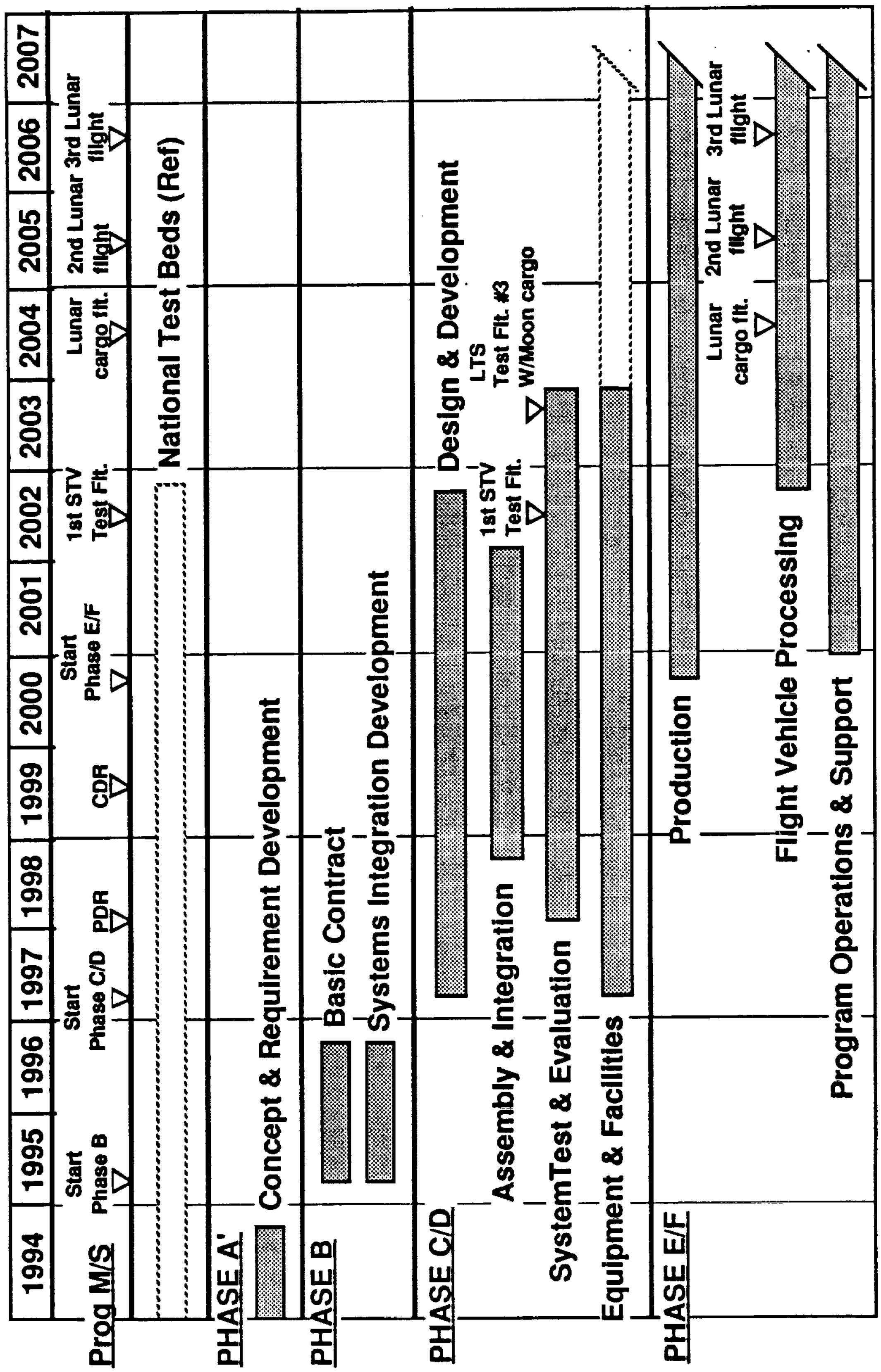


Figure 2.1-15 Program Master Schedule

vehicle is to be space based, then either the RL10 would require additional modifications to the vehicle interface to simplify remove and replace or an ASE would be required with the "clean sheet" interface developed expressly for the space-based design case. At this point in the STV program, without a firm launch vehicle to pose performance restrictions, or a decision to definitely space base the vehicle; the ASE simply provides a lower recurring cost (greater performance) and is not required to perform the mission.

The aerobrake is one entire vehicle element that could be eliminated to reduce front-end funding. The aerobrake pertains only to the space-based vehicle concept and is assembled and maintained at the Space Station between flights. The advantage of the aerobrake is that it eliminates the final Earth orbit burn and thus reduces the amount of propellant and tankage needed to be carried. During the trade studies, aerobraking won over "all propulsive" when examining the program life cycle costs. Breakeven time (where the advantage in recurring cost equalled the cost of development) varied between the first and fifth missions when propellant boost to LEO varied from $6,000/kg to $1,000/kg, respectively. This is a very quick and attractive payback, even for the "cheap boost" case. Clearly however, when examining front-end costs, the aerobrake is one additional element to develop and test. (The design, development, test, and evaluation (DDT&E) difference between an aerobraked stage and an all-propulsive stage is $0.5 to $1 billion.)

The aerobrake (like the ASE) is therefore a cost-enhancing element, not a required element for the space-based lunar mission. The aerobrake is required for the Mars mission scenarios, even in the nuclear thermal vehicle when it is used only on the Mars lander, and will need to be developed for any Mars program. The important questions are (1) how closely tied together will the Mars and lunar programs be, (2) whether the lunar vehicle is space based, and if so, and (3) whether recurring costs, not just development costs are considered important in program selection.

These are two additional elements (the advanced space engine and the aerobrake) that could be delayed or eliminated to reduce the development cost, however at substantial penalty in overall life cycle costs. Even with these two elements, the front end costs of the transfer system defined in the three reference concepts are significantly lower than equivalent year dollars for the corresponding Apollo costs. Figure 2.1-16 shows the comparison between Apollo (which was performance driven, and required significant technology development) and the cost driven STV concepts.

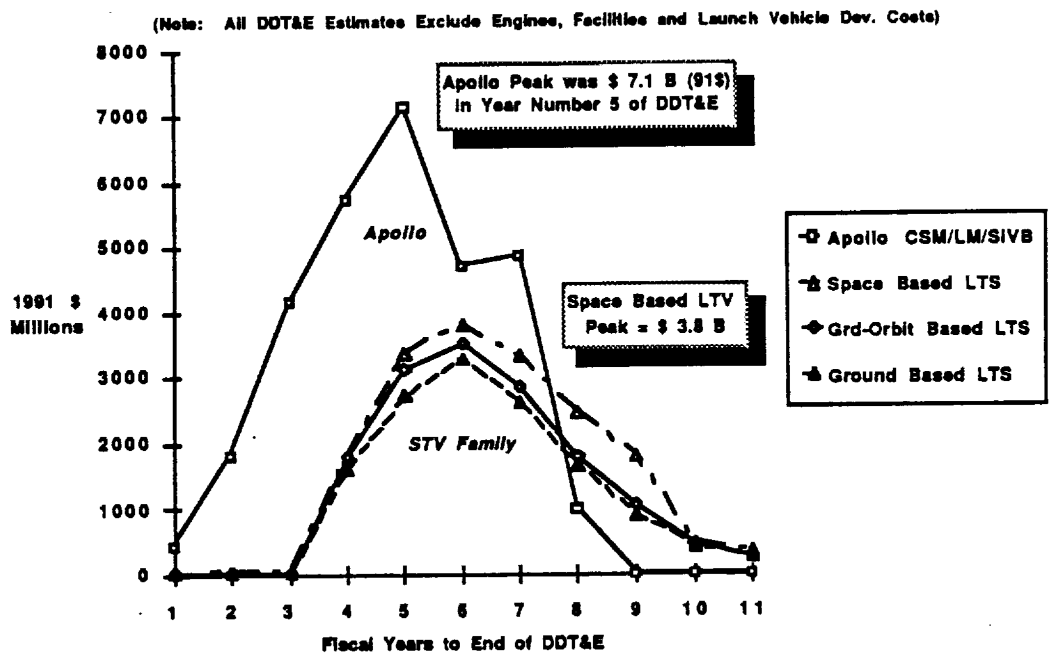


Figure 2.1-16. Comparison of Apollo Hardware DDT&E to Boeing LTS Vehicle DDT&E Estimates

2.2 MAJOR CONCLUSIONS

The launch vehicle is key to lunar transfer system design. The next transfer vehicle will be designed to accomplish missions using whatever new launch vehicle is developed. Decision on a launch vehicle should precede or be done in concert with transfer vehicle design. Accordingly, as the nation proceeds with new launcher development, the STV and lunar mission requirements need to be accommodated in the new launcher. This study has developed a data base where the lowest cost and risk options were identified for three launch vehicle sizes. The data base provides insight into the sensitivity of the vehicle design to various launch vehicle factors, such as the following:

- Launch vehicle boost cost affects staging location and vehicle design (Lunar orbit rendezvous versus direct descent, for example).
- Launch vehicle shroud size and payload capability affects STV stage design and operations (drop tanks, inflight turn and redock maneuvers, top versus side mount of vehicle elements)
- Launch vehicle operations affects STV design and operations ("ship & shoot", pad access, P/A module recovery with direct injection)
- Launch vehicle design for manrating affects STV mission operations & design (launch escape systems versus separate crew launch)
- All above launch vehicle factors affects STV costs

Current technology, with cost enhancing updates in some areas, is sufficient to return to the moon. Some subsystem development in the Reaction Control System (RCS), Thermal Protection System (TPS) and avionics & software were shown to be cost effective, however the missions could still be performed with less efficient alternate systems or operational scenarios. The space based design requires the most advancement in technology, because of the longer orbital lifetime and the additional on-orbit vehicle assembly, checkout and refurbishment required. All concepts were designed to avoid requiring major advancement in cryo fluid management (tank exchange and propulsive settling are baselined for fluid transfer), however highly efficient cryo insulation systems are required in all cases.

Either ground or space based concepts will work well, depending on launch vehicle and mission requirements. Any of the three reference concepts could perform the required mission. The choice between ground and space basing can be made on the basis of other national decisions, such as the launch vehicle capability and the lunar outpost payload sizes. For the mission model and infrastructure costs used, the space based case had 10-20% higher DDT&E and cost 20-25% more over the program life than the ground based case. However, the space based case develops many of the systems, technologies and operations that will eventually be required for a space assembled Mars transfer system. If the Mars and Lunar missions are tightly coupled in time, the space based cost differences may be less severe.

2.3 RECOMMENDATIONS FOR FUTURE STUDY

As stated above, the reference designs were derived for a fixed payload size and mission model - with few other design constraints. With the latest civil needs data base eliminating all manned mission but the lunar mission - future study needs to concentrate on the "lunar mission" as the driving need for the agency's next upper stage development. Additional study could provide resolution as to the "best" design for this upper stage system:

- Examine alternate lunar outpost designs, varying flight frequency, payload sizes, and unsupported lunar staytimes
- Fix launch vehicle size to that currently envisioned for other national requirements and examine how "performance constraining" increases overall system costs
- Examine the fundamental goals and all hardware elements of the lunar program (i.e., payloads, lunar outpost, launch vehicle and transfer vehicle) with various overall program cost ceilings and identify optimum cost allocations and program schedules between the elements
- Examine options involving other programs, including common development program with the Mars mission (storable propellants could be attractive options for ascent vehicles), common earth re-entry vehicles (with ACRV/PLS or Mars return vehicle), or evolution of some stages from an NLS (National Launch System) upper stage.

3.0 STUDY RESULTS

3.1 MISSION ANALYSES

Mission analysis activities consisted of the development of design reference missions (DRM) with associated mission timelines and performance requirements. In addition, trajectory programs were developed and analyses conducted, primarily for the Earth/Moon trajectories.

3.1.2 MISSION MODEL ANALYSIS

3.1.2.1 Mission Model Overview

The missions developed for the STV Concepts and Requirements study were taken from a number of different data sources (Figure 3.1-1). A detailed discussion of the DRM development process, including individual DRM requirements and design drivers, DRM selection rationale, mission timelines, and discussions of the source databases is included in volume II, section 1-1.2.

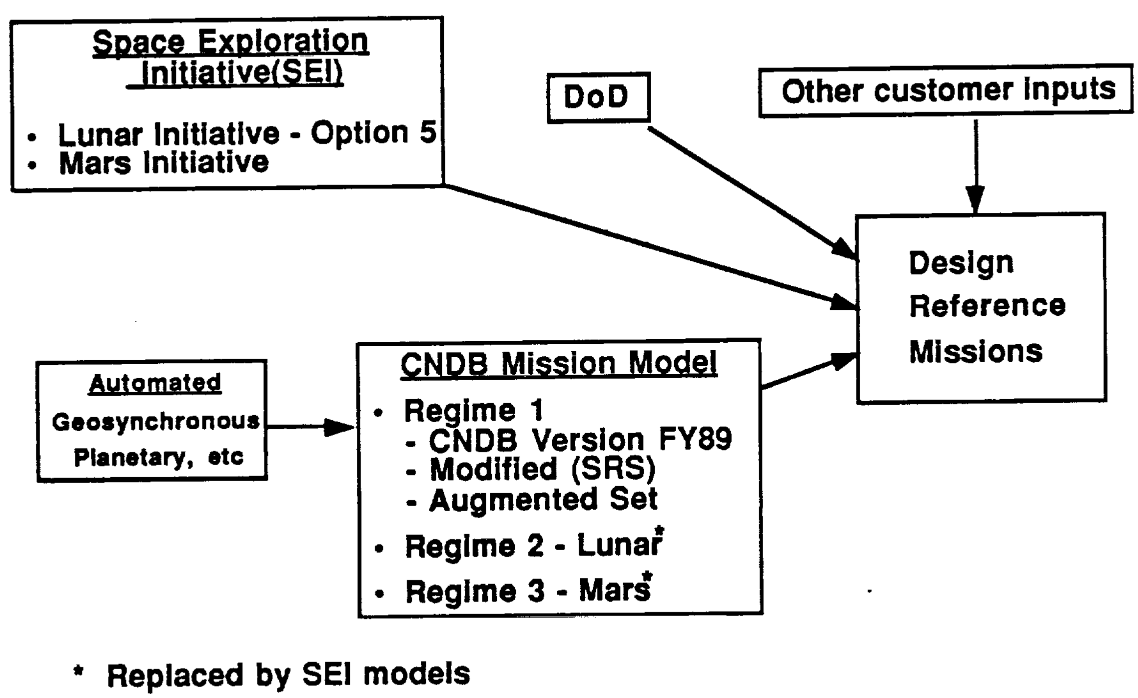


Figure 3.1-1. STV Mission Model

The top-level mission model was supplied at the beginning of the study (August 1989) and consisted of line items taken directly from the 1989 civil needs database (CNDB). The CNDB-based model was further appended with a DoD model supplied by MSFC. The MSFC mission model for STV Concepts and Requirements studies was delivered with NASA HQ approval.

At about the same time as the creation of the STV mission model, a number of scenarios were being developed in support of the Human Exploration Initiative (HEI), which is now called the Space Exploration Initiative (SEI). The lunar and Mars portions of the CNDB were replaced by the SEI Option 5. SEI Option 5 eventually became Reference Approach E in NASA's 90-Day Study on Human Exploration of the Moon and Mars. All of these sources, in addition to the inputs from MSFC, were used to build a set of design reference missions for the STV study.

The design reference missions (DRM) selected for the STV are listed in Figure 3.1-2. The DRMs are divided into two categories: primary and evolutionary missions. The primary DRMs cover a range of lunar missions, both piloted and cargo-expendable. The four lunar DRMs, L1 through L4, were intended to provide sufficient detail to define vehicle and operational concepts for the STV. The lunar DRMs were taken from Option 5 of the lunar Initiative and were based on an informational data book written by NASA-JSC (Initial Study Period Results Summary - Planet Surface Systems - Conceptual Design and Development Requirements) defining the mission manifest and planetary surface systems to be taken as cargo by the STV. The lunar DRMs provided a basis for vehicle designs that met the primary objective of the STV program; to provide a transportation system capable of supporting a human exploration program to the Moon.

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

Figure 3.1-2 STV Design Reference Missions

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

The goal in creating a set of design reference missions was to capture all of the worst case requirements from the large quantity of missions included in the STV mission model in a much smaller and manageable mission set. The design reference missions are not necessarily identical to specific missions in the model but, in some cases, are a mosaic composed of the driving elements of two or more missions from the model which envelope the requirements for that class of missions..

The CNDB FY90 was released toward the end of the study activity. A brief analysis of the new version of the database showed that all of the non-lunar and Mars missions that drove STV requirements were eliminated. The only traditional upper stage missions that remained in the CNDB FY90 were geosynchronous and planetary delivery. Elimination of these driving missions should allow a more graceful evolutionary path for the STV program. The mission model used for the STV study required the majority of the total STV capabilities to be available within a short time after initial flight.

3.1.3 TRAJECTORY ANALYSIS

The performance and trajectory analyses performed during the study can be broken down into four categories: (1) creation of an Earth-to-Moon trajectory database, (2) Development of the Lunar mission survey (LMS) program, (3) Boeing Lunar Trajectory (BOLT) multiphase trajectory program development, and (4) detailed analysis of specific performance and trajectory analysis issues. The LMS uses data from the Earth-to-Moon database with the BOLT program using data from the LMS program. With this process, comprehensive trajectory information can be readily developed including: actual delta V requirements, which include finite burn losses; trip times; Earth/Sun/Moon geometry for communications and thermal design; injection windows with associated delta V penalties for window extension; and pointing accuracies.

Copies of the database and program codes with descriptions and analysis results, were provided to MSFC as completed in 1989 and 1990 in working group and program review meetings. A summary of the four areas of work is given in the following paragraphs with detailed discussions presented in volume II, section 1-1.3.

3.1.3.1 Earth-to-Moon Trajectory Database

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

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

The trajectories were generated by integration, including Moon and Earth perturbations. Results, which completely define Earth-to-Moon (and by symmetry, Moon-to-Earth) trajectories, are stored in four ASCII files of 91 trajectories each for automated lookup by programs such as the LMS program.

3.1.3.2 Lunar Mission Survey Program

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

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

3.1.3.3 Boeing Lunar Trajectory Program

BOLT is a three-degree-of-freedom point mass trajectory simulation program used to rapidly analyze lunar missions. The LMS program, run in advance, supplies approximate times and ΔV values that are input to the BOLT program for further refinement and analysis. All phases of a mission, including launch, Earth orbit, thrusting, translunar and trans-Earth coast, lunar orbit, descent, stay time, ascent, and aerobrake can be included in the same BOLT trajectory. Analysis can be by explicit forwarding, search, optimization, or a combination of the three.

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

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

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

3.1.3.4 Performance and Trajectory Analysis

Several analyses were performed as needed during the period of the STV study. These analyses included: the effects of Earth orbit departure delays, including TLI and midcourse correction burn impacts; injection window considerations from Space Station Freedom (SSF); assessments of mission aborts and free return; lunar orbit stability; descent and ascent from the lunar surface; analyses of the Trans-Earth injection (TEI) burn and aerobrake lift to drag (L/D) requirements; and different near-moon trajectory options. The following sections address specifically some of the abort considerations and the near-moon trajectory option analyses. These, as well as the remaining analyses, are discussed in detail in volume II, section 1-1.3.4.

Mission Abort/Free Return Analysis

An analysis of a LEO node based STV showed that, in general, a free return to the LEO node is not a viable option. Free return is where, after the transfer burn to the moon, lunar swingby and Earth return are accomplished with no further main propulsion system burns. Free return is particularly of interest in the event of system failure(s) precluding a propulsive burn for Earth return. Figure 3.1-3 shows the transfer geometry from a LEO node to the moon with Figure 3.1-4 showing that a free return path does not, in general, arrive at Earth in the plane of the LEO node (e.g. SSF). This is due to the regression of the LEO node orbit plane during the 6 days after departure from the LEO base. The heavy shading in the figure shows the final LEO node orbit plane. Note that these free return issues are only applicable to STV concepts that use a LEO node. Both ground-based options (GB-1.5S and GO-1.5S) have a free return capability as the ballistic reentry crew module can return the crew to Earth.

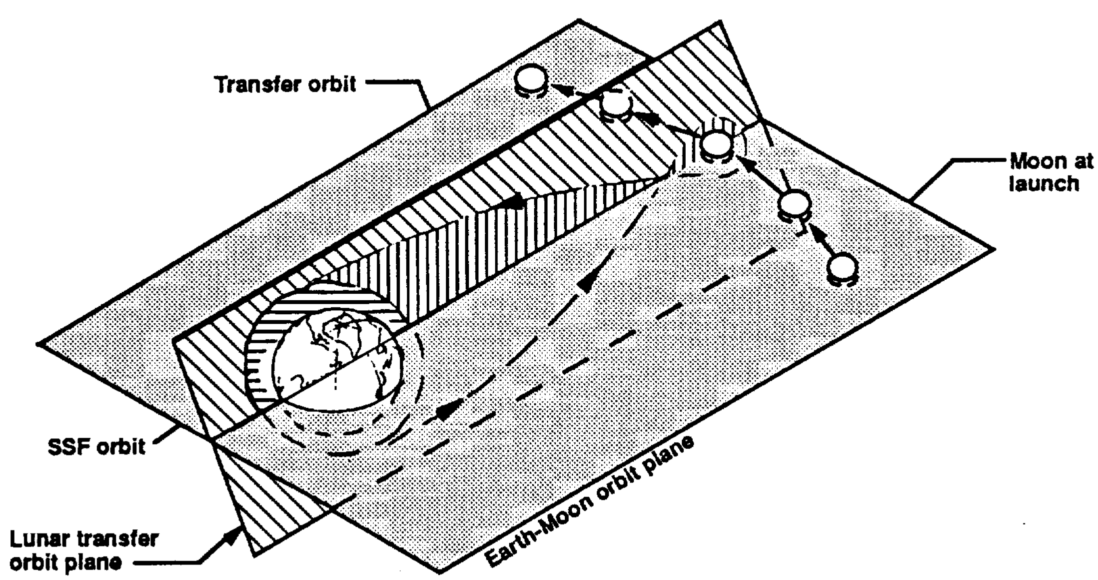


Figure 3.1-3. Lunar Transfer Geometry From SSF

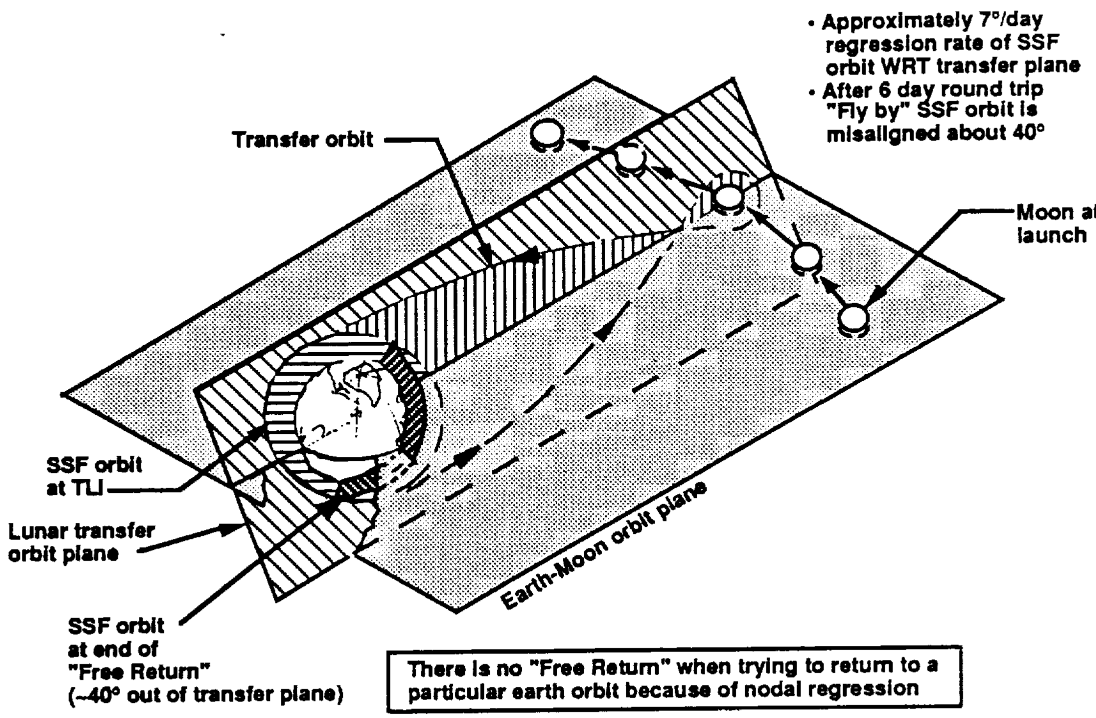


Figure 3.1-4. Impact of SSF on "Free Return"

Special situations can be found supporting free and near-free return to a LEO node which depend on a near-zero angle between the orbit planes of the Moon and the LEO node. However, such situations occur only every 19 years, when the Moon's orbit inclination is near its maximum of 28.5 degrees.

In the event of a need for earliest possible return to the Space Station (or other LEO node), the overriding problem is the potentially large (up to 57-degree) angle of the Moon out of the plane of the Space Station's orbit. (Nominal mission event times are based on the passages of the Moon through this plane, and the opportunities average about 9 days apart.) Figure 3.1-5 reflects this worst case condition in the three upper solid "ΔV required" lines. Available ΔV is shown as dashed lines, decreasing in three phases with the nominal burn expenditures. Even a so called "free return" from translunar trajectory cannot avoid the requirement for high ΔV because the Moon is, in general, out of the plane at the time of flyby. Note that the data presented in Figure 3.1-5 was generated for the 90-day study reference vehicle (2.5 stage, LEV/LTV scenario,

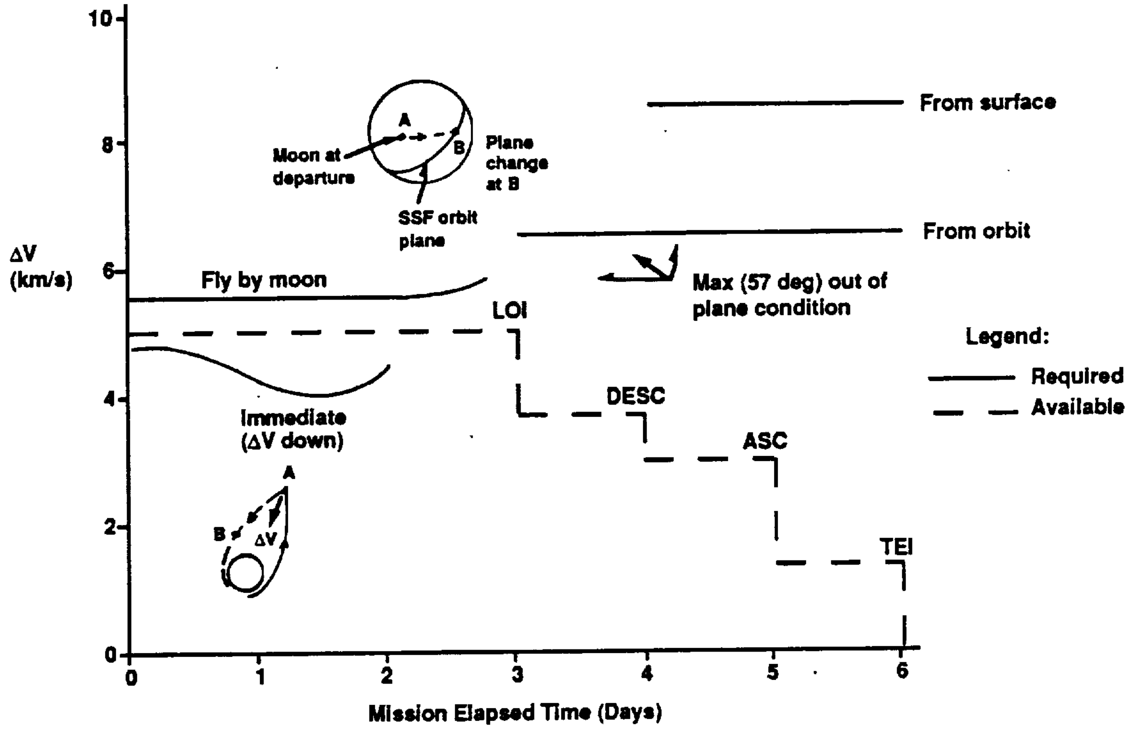


Figure 3.1-5. Abort Scenario Capability

using LOR) and is presented here to provide visibility into concerns which must be addressed.

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

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

Trajectory Options

Two options for targeting of the near-Moon portion of the trajectory were initially studied with a third option, subsequently selected, being identified later in the study. These three options are shown in Figure 3.1-6. Figures depicting these options in connection with abort considerations are presented in volume II, section 3.1.3.4.3.

The first option addressed is the lunar orbit rendezvous (LOR) approach used on the Apollo missions. In this option, a Lunar Orbit Insertion (LOI) burn establishes a low-inclination circular lunar orbit from the initial free-return approach path. In general, this approach is used in conjunction with a LLO

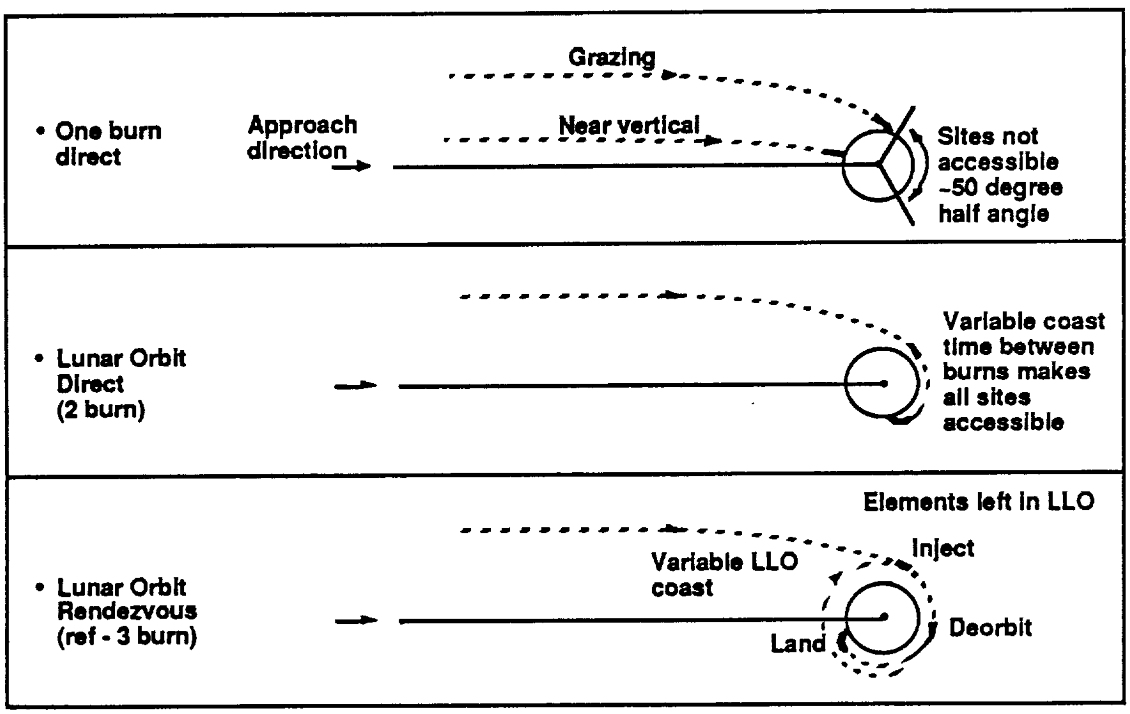


Figure 3.1-6. Lunar Approach Trajectory Options

node. Operationally, a Lunar excursion vehicle can be stored in LLO with a transfer vehicle bringing the crew and cargo from Earth and transferring crew and cargo to the excursion vehicle for the Lunar portion of the mission. Upon conclusion of the Lunar portion of the mission, the excursion vehicle returns the crew to the transfer vehicle which returns the crew to Earth while the excursion vehicle remains in LLO. Alternatively, for single stage vehicle, return propellant, the aerobrake, or other elements can be stored in LLO to be picked up prior to return.

The benefit of the LOR approach is that performance is optimized as only mass required for the lunar portion of the mission is taken to and from the Lunar surface. There is a significant risk penalty associated with the LOR option as elements required for Earth return are left in LLO and successful rendezvous, docking, and mating of several interfaces is required before the crew can be safely returned to Earth. Additionally, as discussed before, better performance does not necessarily equate to lower costs. The higher propellant costs for other approaches are generally offset by the higher development costs for the elements left in LLO which require station-keeping and rendezvous and docking capabilities.

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

Detriments with the direct approach are that direct landing and ascent incurs high gravity losses, especially over sites requiring a nearly vertical trajectory, which results in the worst performance of the three options. The direct approach is a fairly risky approach as the vehicle is on an impact path for approximately one day, after the retargeting for the landing site. In addition, due to the approach geometry, there are substantial portions of the lunar surface (primarily on the dark side but including some areas on the near side) which are unavailable using the direct option. For these reasons, this approach was not selected.

The third option, called the lunar orbit direct (LOD) approach, was developed later in the study and subsequently selected as the lunar approach trajectory for the downselected vehicles. In this option, the STV departs from the free-return path about 1 day prior to arrival to target to a LLO (possibly high inclination) passing over the landing site. The LLO is elliptical having a minimum periapsis altitude (about 5 km), with the orbit oriented to put periapsis over the landing site. The descent is a single burn following a fractional or multiple orbit coast. Similarly, ascent is a single burn to a low periapsis, possibly with a high inclination orbit from which TEI may occur after a fractional or after multiple revolutions. There is not necessarily a relation between the lunar orbits used for descent and ascent in the LOD option. The goal is to be able to land after only a fractional orbit, however, initial use of multiple orbits may be desired to prove out navigation techniques.

The benefit of the LOD approach is that less risk is incurred, compared to both the LOR option, as no elements required for return to Earth are left in LLO, and to the direct approach as the STV is not on a direct impact path until just prior to landing. As all vehicle elements are taken to the lunar surface, this approach has poorer performance than the LOR approach (although about the same ΔV requirements), however, the simplicity of the vehicle design offsets the higher propellant costs, especially as Earth to Orbit transportation costs decrease.

3.2 SYSTEM TRADE STUDIES

The system trade studies and analyses necessary for definition of the STV system were identified using a team approach. Issues were brainstormed and documented and the preferred approach, such as a trade study or ground rule, for addressing the issues was then identified. An influence matrix was developed showing the interactions of the trade studies with one another in order to identify the priority for conducting the trades and identifying the interactions of trade results with other issues. Figure 3.2-1 shows a tree of the trades and analyses conducted for each STV program area such as vehicle design or operations.

The System Architecture Trade Study was a major effort of the STV study and combined several architecture trades into an overall architecture trade study.

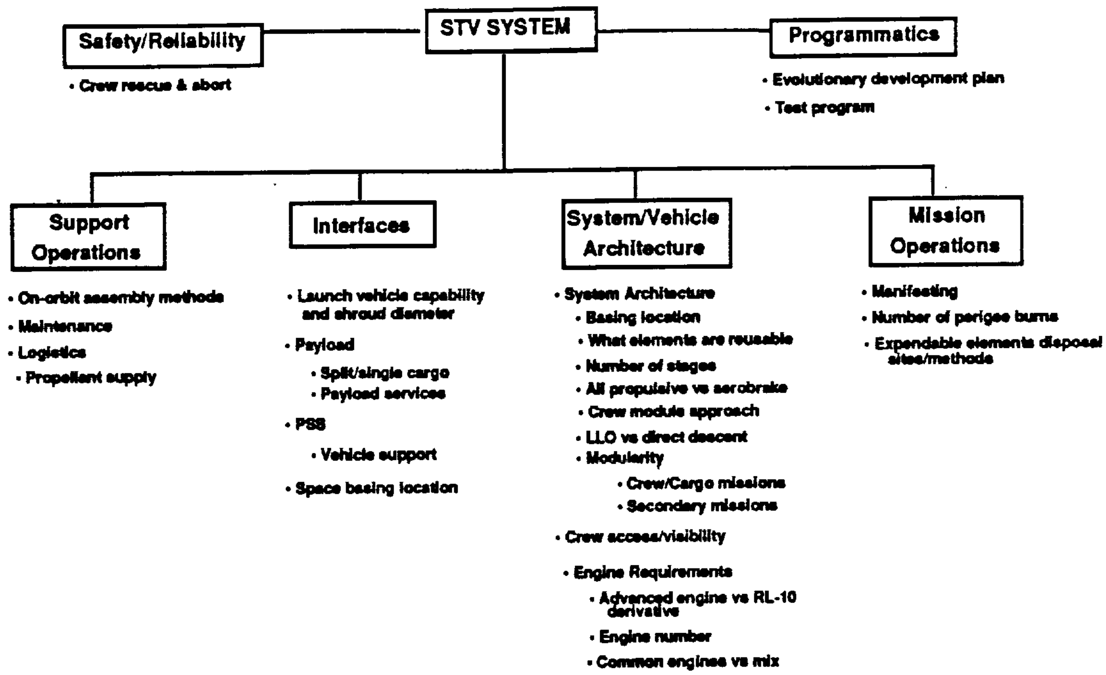


Figure 3.2-1. System Trade Studies

Several of the architecture trades were interdependent. It was therefore felt that a combined trade could account for the interactions by providing evaluations of one trade across different options of other interdependent trades. In this method, the best combination of architectural options could be determined.
The six trades that were combined in the System Architecture Trade Study consisted of trades addressing (1) the number of stages, (2) basing options, (3) the number and operational use of crew modules, (4) use of a low lunar orbit (LLO) transportation node versus a direct descent to the lunar surface, and for the space-based options only: (5) use of an aerobrake versus main propulsion for Earth capture upon return, and (6) use of propellant tankers versus drop tanks for supply. Evaluation criteria and criteria weighting against which the options were evaluated consisted of cost (50% weighting), margins and risk (30% weighting), other mission capture (15% weighting), and benefits to Mars (5% weighting).

The options defined for the six architecture trades were combined in a matrix resulting in over 400 possible architectures. Ground rules and assumptions were applied to reduce these combinations to 94 architectures for which performance and mission scenarios were developed. Figure 3.2-2 presents the scrubbed matrix used to identify the options to be evaluated. Based on this work, 29 scenarios were selected and initially assessed against the cost and margins and risk evaluation criteria to determine trending. Based on the observed trends, 13 additional scenarios were initially included with one being added later. The resulting 43 scenarios were fully evaluated against the four evaluation criteria to determine the preferred architectures. Figure 3.2-3 provides an overview of the System Architecture Trade Study process with section 2-1.1, Volume II discussing both the process and results in detail.

Mission scenarios were developed for each of the 94 architectural options identified with the reduced matrix. For each option an overview and timeline was developed, the mission phases and operations in which each generic flight element was involved were defined, and the characteristics and requirements for each scenario were identified.

Orbital Options

		GB1	GB2	SB1	SB2	SG1	SG2	GO1	GO2	EC1	EC2	
		E	*E*	*LEO*	*LEO*	*LEO*	*LEO*	*LEO*	*LEO*	*E*	*LEO*	
		LS	*LLOLS*		*LLO*	*LS*	*LLOLS*		*LLO*	*LS*	*LS*	
		E	*LLO*	*LEO*	*LLO*	*E/*	*LLO*	*E*	*LLO*			
			E		*LEO*	*LEOB/*			*E*			
							LEO					
Basing	Grnd based single lnch	X	X							X		
	Grnd bsd, on-orbit assy							X	X		X	
	Space based			X	X							
	Comb space/grnd bsd					X	X					
Recovery	All-propulsive return				X							
	Aerobrake return			X	X							
	Re-entry return	X	X					X	X			
	Aerobrake P/A, Re-enter CC					X	X					
	All prop P/A return, Re-enter CC						X					
Stages	1.5 stage	X	X	X	X	X	X	X	X	X	X	
	2 stage	X	X	X	X	X	X	X	X	X	X	
	2.5 stage	X	X	X	X	X	X	X	X			
	3 stage	X	X	X	X	X	X	X	X			
	3.5 stage		X		X		X		X			
	4 stage		X		X		X		X			
Crew Cabs	1 CC	X	X	X	X	X	X	X	X			
	2 CC		X		X		X		X			
	Hybrid CC		X		X		X		X			
	Number of scenarios	4	18	4	19	4	19	4	18	2	2	= 94
	Trending selection	2	9	2	12	1	8	4	4	0	0	= 42

▭ - APOLLO ▒ - 90 DAY REFERENCE

Figure 3.2-2. Scenario Matrix

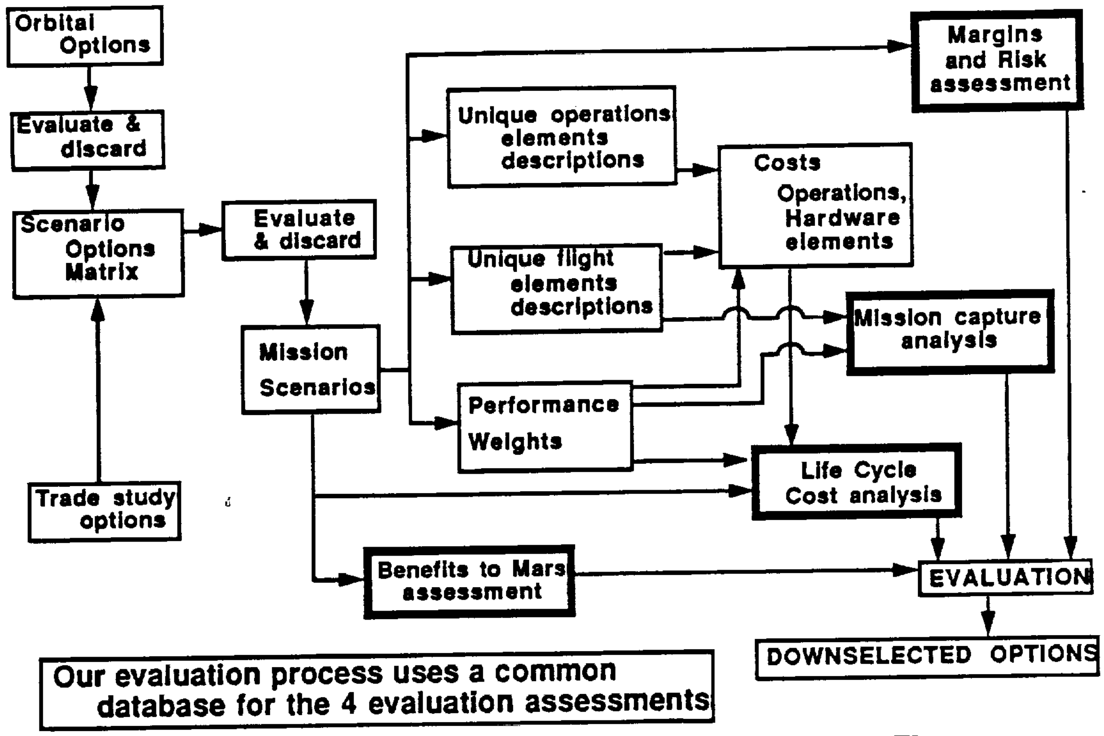


Figure 3.2-3. System Architecture Evaluation Flow

Using the mission scenarios, unique flight elements were identified and characterized as shown in Figure 3.2-4. A functional split was made between flight elements to distinguish mass and subsystem definitions, as well as unique hardware and operations. In summary, an analysis of the 94 mission scenarios yielded a total of 546 flight elements. Analysis of these flight elements with respect to unique mission functions resulted in 33 functionally unique flight elements. General categories of the flight elements identified and characterized included aerobrakes, transfer stages, ascent stages, lander stages, drop tank modules, transfer crew modules, and excursion crew modules. An additional effort conducted as part of the flight element definition was an avionics functional and location definition. For each of the flight elements, the avionics functions associated with that flight element was identified. The ultimate goal was to identify concept differences that distinguished hardware and operations costs.

To support (1) the cost and margins and risk assessments and (2) the subsystem design task, operations flows were developed for the mission scenarios. Operations were defined from the start of KSC processing of a new

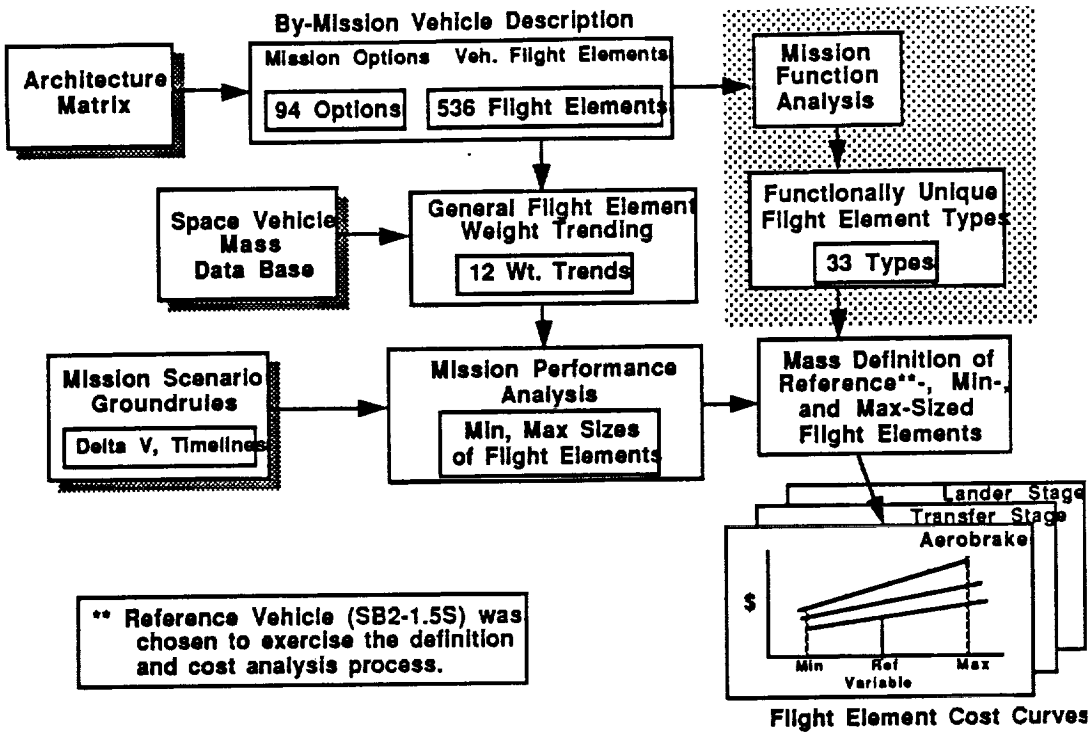


Figure 3.2-4. Flight Element Definition Process

vehicle to the end of the mission of its second flight. This covers all major events, except final disposal, in the vehicle's life, including refurbishment for reflight.

A diverse source of inputs was considered in developing the operations flows. Studies have been performed in the past by several major contractors whose primary purpose was to define on-orbit operations of an OTV (STV or lunar vehicle). Operations were defined at a major task description level, with a ROM estimate of task duration hours assigned.

System Architecture Trade Study Evaluation. In general, the scenarios were evaluated against each of the four evaluation criteria and then normalized to a 1 to 5 scale with 1 being the "best" score and 5 being the "worst" score. The total score was then developed as a summation of the score for each criterion times the weighting of each criterion.

Cost Assessment. The cost score was based on a combination of 70% DDT&E costs and 30% life cycle costs. This approach was based on the belief that the DDT&E costs, being the driver behind the level of funding required to obtain a new program start, should be strongly emphasized. All scenarios met the basic mission requirements, so an affordable funding profile at the beginning of the program, which would facilitate a program start, was seen as a valid discriminator.

Figure 3.2-5 shows the process used to develop the life cycle costs for each scenario. A life cycle cost model then used cost elements associated with each flight element, boost costs per flight element per mission, and the number of each kind of flight (steady state, replacement, and expendable cargo) from the Option 5 mission model along with the non-recurring costs/scenario to determine the overall life cycle cost for each scenario.

Margins and Risk Assessment. All of the STV system concepts and each of the subsystems met safety requirements with margins for all contingencies. In

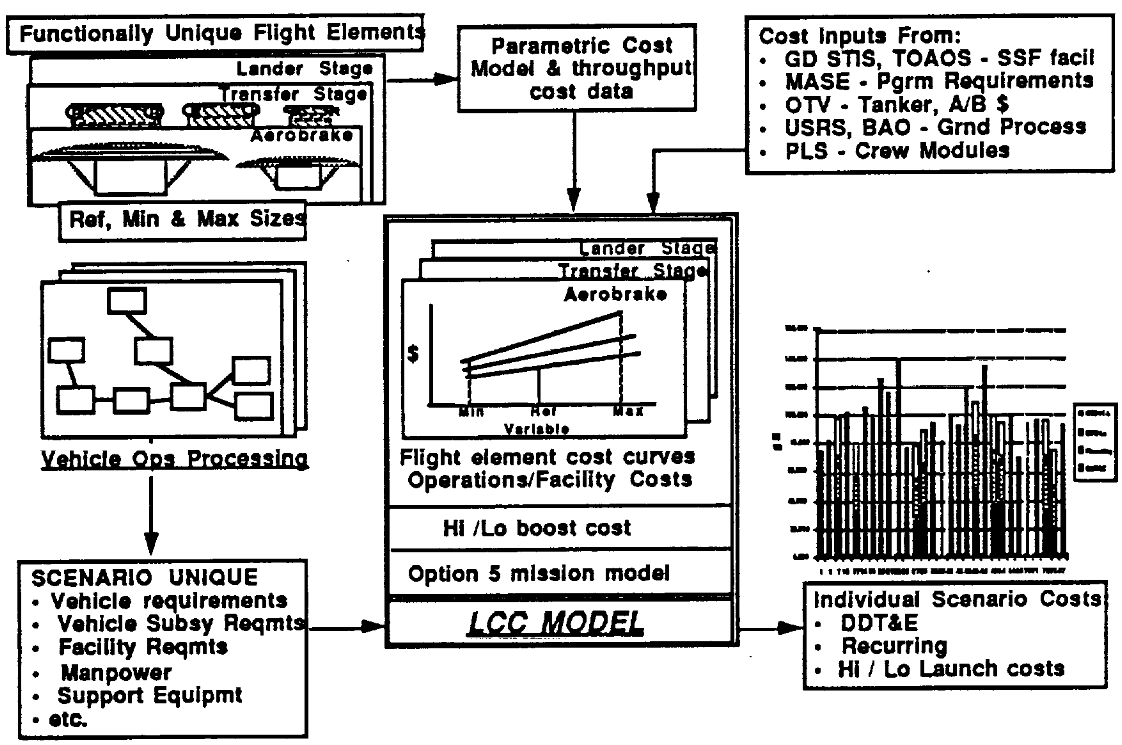


Figure 3.2-5. Cost Definition Process

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

The risk area was broken into equal weighting between technical and programmatics risk. Technical risks dealt with the risk during the operational phase and included such things as mission success, performance and operation, and safety and reliability. In general, the programmatic risk dealt with the anticipated risk associated with the FSD program phase (i.e., cost and schedule).

Mission Capture Assessment. The purpose of this analysis was to determine how well the STV concepts designed for the lunar missions could capture other NASA and DoD missions identified as design reference missions. Concepts were scored both by stage efficiency (how efficient the lunar-sized stage can perform the other missions; required propellant mass/total start mass, excluding payload) and by Earth-to-orbit launched mass.

Benefits to Mars Assessment. Mars mission benefit was one of the evaluation criteria for STV concept selection with a 5% weighting of the total evaluation criteria. The purpose of this analysis was to determine how much the STV concepts, operations, and subsystems, designed for the lunar missions, can benefit the Mars missions and vehicle designs as they are projected at the current time. To determine the overall benefit of each of the lunar vehicle concepts, specific benefits were weighted independently, scored, and then combined.

System Architecture Trade Study Results. The System Architecture Trade Study resulted in a downselect to three architecture options for further definition. All of the scenarios were 1.5-stage vehicles using a single crew module and the lunar orbit direct trajectory approach for lunar landing. The main difference in the three scenarios was in the basing and launch infrastructure. One scenario was ground based with a single launch, one was ground based with on-orbit assembly by way of rendezvous and docking, and the final scenario was a space-based architecture. For the space-based case, drop tanks were used instead of propellant tankers and an aerobrake was used for return to SSF.

One of the findings in this trade was that better performance did not necessarily equate to lower costs. Better performing systems tend to have higher development and operations costs that outweigh the higher propellant delivery costs associated with lower performing systems.

Number of Stages. The results of the scenarios compared for the number of stages trade strongly indicated that fewer stages were preferred, with the single-stage scenarios (with drop tanks) being the clear winners. Although the single stages, in general, did not have the best performance, the reduction in operational complexity and development costs for the fewer stage vehicles outweighed the performance penalties.

Crew Module Approach. The single and hybrid crew module approaches were close, with the dual crew module losing. In general, the single crew modules had the lowest cost and the hybrid crew modules had less risk because of the presence of two independently pressurized volumes available for the majority of the mission. The dual crew modules had the highest costs because of the LLO basing of the crew module and the higher costs associated with development of two elements. Note that the hybrid and dual crew modules were options only when a LLO node was used for mass storage during the missions (i.e., LOR lunar approach trajectory option). Based on the generally better scores for the single crew module, along with the results of the lunar approach trajectory trade, the single crew module was selected.

CAMUS Incorporated, a consulting company formed by astronauts William Pogue and Gerry Carr, which was under subcontract for this study, assessed the crew module options from a safety and abort perspective. Their assessment, in summary, was that: the single crew module was preferred for operational simplicity. Also, undesirable risk was introduced by the other crew module options that required rendezvous and docking, possible long storage periods in orbit, and on-orbit mating of multiple interfaces.

Basing Location. In general, the two ground-based options scored better than the two space-based options because of generally lower costs and reduced risks. The ground-based, orbital assembly option scored best on costs because all refurbishment operations took place on the ground. The ground-based, single launch concept also had ground refurbishment; however, this option incurred a $7 billion penalty for development of the large booster (≈ 260 metric tons). The lunar and Mars missions were seen as the only missions benefiting from this size booster with the lunar missions having the initial requirement and thus a share of the development costs (primarily facilities modifications). Note that the space-based scenarios, nominally based at Space Station Freedom, incurred a $4.5 billion cost for modifications to SSF. This cost estimate, taken from the General Dynamics Space Transportation Infrastructure Study (STIS), was broken down into top-level elements and examined and accepted as a reasonable estimate.

One of the intentions of the study was to develop and provide a decision database with basing being seen as a primary issue in the definition of the STV. For these reasons, the three basing options were retained in the downselected scenarios to allow more detailed definition of the impacts and costs of the different basing approaches. The different basing approaches depend, in many respects, on other space transportation infrastructure considerations. For example, the ground-based, single-launch concept requires booster capability on the order of 260 metric tons; the ground-based, orbital assembly concept requires booster capability on the order of 125 metric tons; and the space-based concept requires a 71 metric ton booster. By carrying the three options, a database is available in response to other infrastructure decisions. An examination of the top 10 scores reveals that, if the LOR approach is not used, the top three scenarios were selected for further definition.

Lunar Approach Trajectory. At the time the System Architecture Trade Study was being conducted, only two lunar approach options had been identified. After the trade was nearly complete, the two-burn lunar orbit direct (LOD) approach was identified. An assessment of this approach showed that, in terms of the evaluation criteria used, LOD was similar to the direct approach. The direct approach was seen to be preferred over the LOR approach and the differences between the direct approach and LOD only favored LOD.

CAMUS was also asked to assess the lunar approach trajectory options in terms of safety and abort considerations. Their assessment was that the LOD approach appears feasible and worth pursuing, initial use of the fractional orbit approach may be optimistic, and the initial use of multiple orbits with growth to the fractional orbit approach may be desirable. Also, leaving elements required for Earth return in LLO for up to 6 months during the missions (LOR approach) introduces risk and is not the preferred approach. The LOD approach builds on Apollo experience instead of duplicating it. If a multiple orbit LOD scenario is initially selected, accommodations for growth to the fractional orbit approach should be guaranteed (i.e., not precluded by configuration, propulsion, and so forth).

Aerobraked Versus All-Propulsive Return. Cost slightly favored all-propulsive return, influenced both by the 70% DDT&E component of the cost scoring (effectively penalizing the aerobrake) and the low boost cost of $1000/kg, which favored the all-propulsive approach with the required additional propellant available in LEO at a relatively low cost. Margins and risk, somewhat obviously, also favored the all-propulsive approach because this type of operation has been performed before whereas use of the aerobrake would entail an all new development. The benefits to Mars criterion favored, again somewhat obviously, the aerobrake approach as aerobraking is required for a Mars landing. Note that the margins and risk and benefits to Mars criteria tended in opposite directions as new technology and operational approaches obviously entail a higher level of risk than use of existing hardware and operational concepts. The relative weighting of the criteria was an important factor in the all-propulsive approach having the best scores.

The aerobrake was retained in the interest of developing the technical database and aerobrake details. Additionally, the evaluation methodology did not allow for higher weighted scoring based on mitigating factors. In this case, the lunar transportation system is the only SEI opportunity to prove out aerobraking, unlike other technology and operational areas that will benefit from development and operation of SSF, the lunar base, new ETO systems, and so forth.

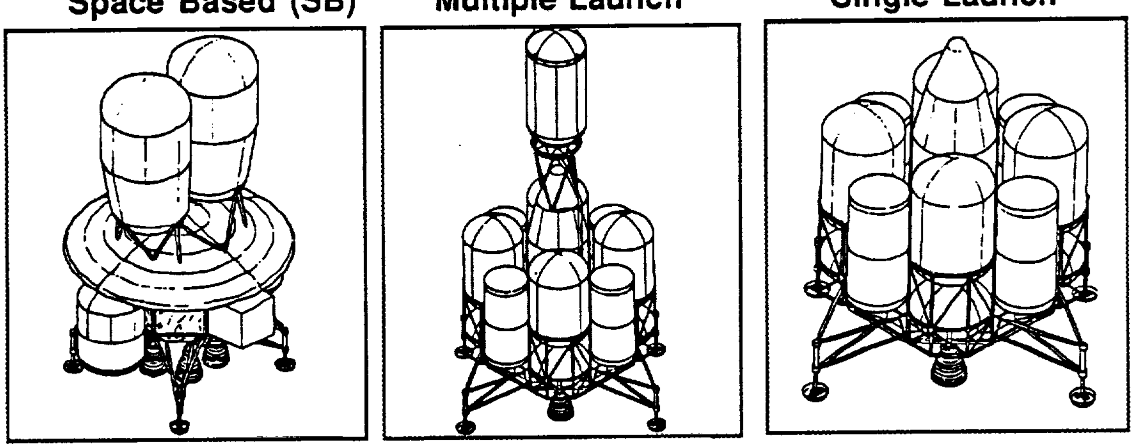
Drop tanks Versus Use of Propellant Tankers. Two scenario pairs were used to trade the use of drop tanks versus the use of propellant tankers. Note that this trade (along with the entire system architecture trade) was based on the lunar missions only, with the exception of the mission capture evaluation, which used the lunar transportation system optimized elements as required to perform the non-lunar design reference missions. Based on the lunar missions, the use of drop tanks was slightly favored over the use of propellant tankers and was selected as the baseline for the space-based vehicles.

3.3 RECOMMENDED APPROACHES

3.3.1 Vehicle Descriptions

The purpose of this STV study was to identify and study a transportation system from Earth orbit to the lunar surface and look at its applicability to other orbital transportation needs. This section provides an overview of the STV concepts resulting from system architecture trades, mission analyses, and subsystem trades. The selected concepts are shown in Figure 3.3.1.1 and include a space-based, single-stage vehicle with expendable drop tanks; a ground-based, single-stage multiple-launch vehicle with expendable drop tanks and lunar lander; and a ground-based, single-stage single-launch vehicle with expendable drop tanks and lunar lander. The two ground-based concepts are similar in design, but the multiple-launch concept includes a LO2 tanker for filling vehicle LO2 tanks on orbit. All concepts have a single crew module for piloted missions and use a lunar-surface-direct transfer, requiring no

Space Based (SB)	Ground Based Multiple Launch	Ground Based Single Launch



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

Figure 3.3.1-1. STV Concepts

rendezvous in lunar orbit. The space-based core vehicle uses an aerobraking maneuver to return the crew module and core stage to the Space Station or other LEO node; however, on the ground-based vehicle only the crew module returns to the ground and is recovered. These three concepts satisfy current study requirements and were chosen to carry forward for further study.

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

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

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

The space-based vehicle is made up of the following subsystems, as shown in Figure 3.3.1-2:

1. Structures and Mechanisms - Includes a core stage with external load-bearing body structure and landing gear, a rigid aerobrake, a pressurized crew module, two sets of TLI drop tanks, and two sets of descent drop tanks.

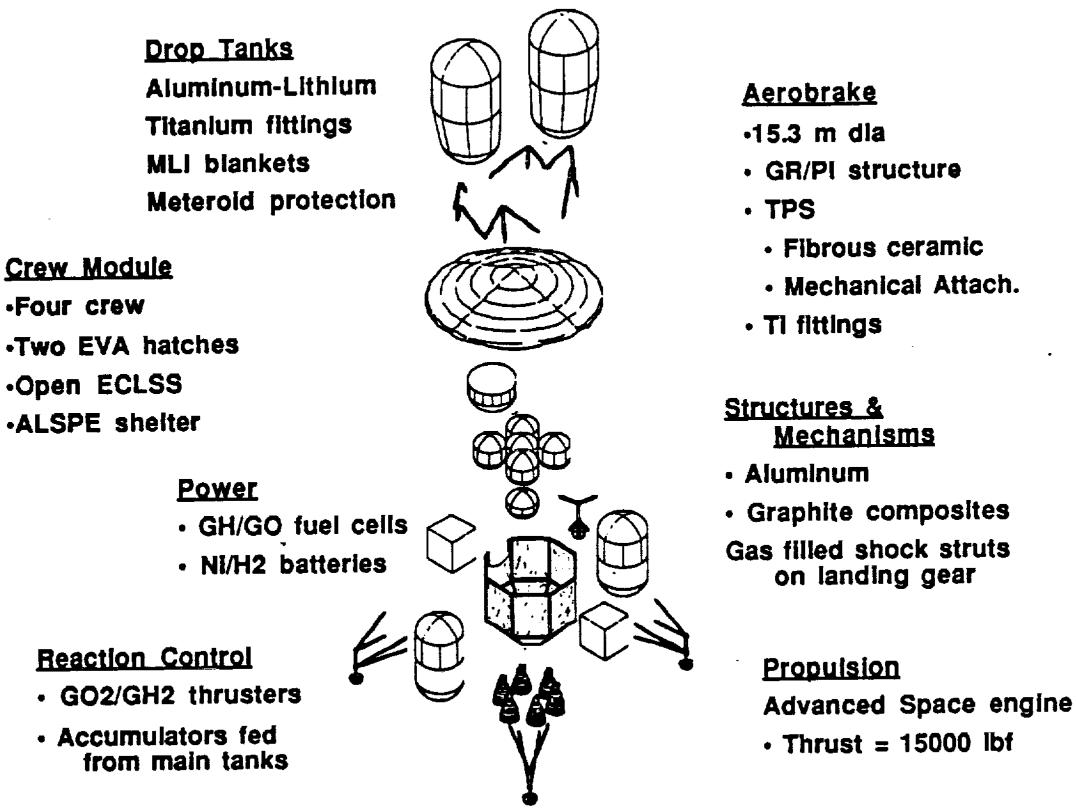


Figure 3.3.1-2. Space-Based Subsystem Overview

2. Main Tankage - The core stage has two cylindrical LO2 tanks and two cylindrical LH2 tanks with associated propellant acquisition devices. Each drop tank set has a single LO2 tank and a single LH2 tank with associated slosh baffles and propellant acquisition devices.
3. Protection - Includes thermal control and damage protection of the main cryogenic tanks, thermal control of avionics and power equipment, thermal and radiation protection of the crew during long-duration exposure in space, and thermal protection of the aerobrake during the aerobraking maneuver.
4. Main Propulsion - Consists of six 66,800N (15,000-lb) thrust advanced expander-cycle engines (Isp = 481 seconds) with electromechanical actuation and propellant delivery, pressurization, fill, and vent systems.
5. Reaction Control - Includes four GO2/GH2 (Isp = 410 second) thruster modules and associated accumulators, pressurization, and control.

6. Electrical Power - Features redundant $O2/H2$ fuel cells fed from accumulators filled from the vehicle main propellant tanks, as well as distribution and control units and associated wire harnesses.
7. Guidance and Navigation - Provisions for lunar mission operations, including rendezvous, docking, and lunar landing, with built-in redundancy for piloted operations.
8. Communication and Data Handling - Provisions for communication, vehicle health maintenance, and data handling, with audio and video interfaces for piloted operations and instrumentation for drop tank monitoring and control.
9. Displays and Controls - Provisions on the crew module for limited crew control and status monitoring of the vehicle during critical phases of the mission.
10. Environmental Control - Provisions on the crew module for atmosphere supply and control, internal equipment cooling, and metabolic and equipment heat rejection.
11. Personnel Provisions - Food, water, and waste management systems as well as fire detection and crew furnishings on the crew module.

The current space-based vehicle concept can either deliver 9,870 kg of cargo to the lunar surface in a piloted mode or 52,683 kg in a cargo-delivery mode. With this cargo split, 418 tons of cargo can be delivered to the lunar surface in 21 piloted and 4 cargo-only missions. The size of the vehicle is common to both piloted and cargo-only missions.

Sequential mass properties for the space-based STV are shown in Figure 3.3.1-3.

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

ITEM	Mass - kg W3	W2	CENTER OF MASS - m Xcg	Ycg	Zcg	MOMENTS (KG-M^2) IXX	IYY	IZZ
Mass Properties Summary								
CARGO		9870	3.81	0.00	0.003	225E+05	3.295E+05	2.221E+04
CARGO 1	4935		3.81	0.00	5.58	7.588E+03	1.110E+04	1.110E+04
CARGO 2	4935		3.81	0.00	-5.58	7.588E+03	1.110E+04	1.110E+04
CREW MODULE		5584	6.35	0.00	2.169	079E+03	9.079E+03	9.079E+03
CREW MODULE	4493		6.35	0.00	2.16	8.806E+03	8.806E+03	8.806E+03
CREW	400		6.35	0.00	2.16	1.000E+02	1.000E+02	1.000E+02
EVA SUITS	400		6.35	0.00	2.16	1.000E+02	1.000E+02	1.000E+02
CREW CONSUMABLES - total	291		6.35	0.00	2.16	7.275E+01	7.275E+01	7.275E+01
STAGE- (D&A Lander) P/A MODULE		29974	4.24	0.00	-0.53	1.602E+05	1.696E+05	2.175E+05
STAGE INERT	8522		2.22	0.00	-0.04	1.025E+05	1.047E+05	1.107E+05
PROPELLANT	21452		5.04	0.00	-0.73	5.477E+04	1.343E+04	5.815E+04
AEROBRAKE		4109	7.87	0.00	0.00	1.220E+05	8.028E+04	8.028E+04
DROP TANK SET-TLI (2 SETS)		143856	9.94	0.00	0.00	1.491E+06	1.021E+06	2.476E+06
TANKSET INERT	4301		13.33	3.18	0.00	1.826E+04	5.975E+04	5.975E+04
PROPELLANT	67627		9.73	3.18	0.00	0.000E+00	3.982E+05	3.982E+05
TANKSET INERT	4301		13.33	-3.18	0.00	1.826E+04	5.975E+04	5.975E+04
PROPELLANT	67627		9.73	-3.18	0.00	0.000E+00	3.982E+05	3.982E+05
DROP TANK SET-Descent (2 SETS)		61618	4.71	0.00	0.00	2.746E+06	2.624E+05	2.703E+06
TANKSET INERT	2424		3.18	6.48	0.00	1.644E+04	2.145E+04	1.120E+04
PROPELLANT	28385		4.84	6.48	0.00	6.295E+04	1.036E+05	4.065E+04
TANKSET INERT	2424		3.18	-6.48	0.00	1.644E+04	2.145E+04	1.120E+04
PROPELLANT	28385		4.84	-6.48	0.00	6.295E+04	1.036E+05	4.065E+04
SEQUENCED MASS DATA								
LEO ASSEMBLY		255011	7.66	0.00	-0.02	4.886E+06	3.698E+06	7.300E+06
START TRANS-LUNAR INJECTION		249013	7.57	0.00	0.004	815E+06	3.614E+06	7.124E+06
PRIOR TO LUNAR ORBIT INSERTION		110373	4.73	0.00	0.00	3.284E+06	9.623E+05	2.990E+06
LLO OPERATIONS		87860	4.74	0.00	0.002	280E+06	8.663E+05	2.002E+06
LUNAR LANDING		58389	4.75	0.00	0.019	669E+05	7.408E+05	7.095E+05
BEGIN LUNAR ASCENT		42029	4.99	0.00	0.033	406E+05	3.738E+05	3.763E+05
START TRANS EARTH INJECTION		28476	5.35	0.00	0.302	619E+05	2.794E+05	2.666E+05
START AEROMANEUVER		20474	5.64	0.00	0.37	2.012E+05	2.076E+05	1.960E+05
EOM MASS		19050	5.75	0.00	0.431	924E+05	1.953E+05	1.827E+05

Figure 3.3.1-3. Space-Based Vehicle Sequential Mass and Fluid Inventory

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

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

The ground-based vehicle includes the following subsystems, as shown in Figure 3.3.1-4:

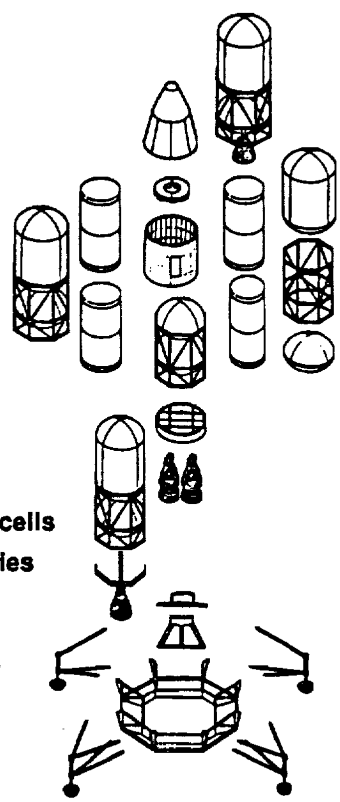


Figure 3.3.1-4. Ground-Based Vehicle Subsystem Overview

1. Structures and Mechanisms - Includes a core stage with external load-bearing body structure, a lunar lander with landing gear, a pressurized crew module with an external aerodynamic shell, two sets of TLI drop tanks, and two sets of delivery dropstages.
2. Main Tankage - The core tankset and each drop tank set has a single LO2 tank and a single LH2 tank with associated slosh baffles and propellant acquisition devices. The LO2 tanker is a single tank with internal stiffening and slosh baffling capable of withstanding launch conditions fully loaded.
3. Protection - Includes thermal control and damage protection of the main cryogenic tanks, thermal control of avionics and power equipment, thermal and radiation protection of the crew during long-duration exposure in space, and thermal protection of the crew module for the reentry maneuver.
4. Main Propulsion - Consists of a total of six 66,800N (15, 000-lb) thrust advanced expander-cycle engines (Isp = 481 second) with electromechanical actuation and propellant delivery, pressurization, fill, and vent systems.
5. Reaction Control - Includes four GO2/GH2 thruster modules (Isp = 410 second) on the delivery stages and four on the crew module, with associated accumulators, pressurization, and control.
6. Electrical Power - Features redundant O2/H2 fuel cells fed from accumulators filled from the vehicle main propellant tanks, as well as distribution and control units and associated wire harnesses.
7. Guidance and Navigation - Provisions for lunar mission operations, including rendezvous, docking, and lunar landing, with built-in redundancy for piloted operations.
8. Communication and Data Handling - Provisions for communication, vehicle health maintenance, and data handling, with audio and video interfaces for piloted operations and instrumentation for drop tank monitoring and control.
9. Displays and Controls - Provisions on the crew module for limited crew control and status monitoring of the vehicle during critical phases of the mission.
10. Environmental Control - Provisions on the crew module for atmosphere supply and control, internal equipment cooling, and metabolic and equipment heat rejection.

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

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

Sequential mass properties for the ground-based lunar piloted mission are shown in Figure 3.3.1-5

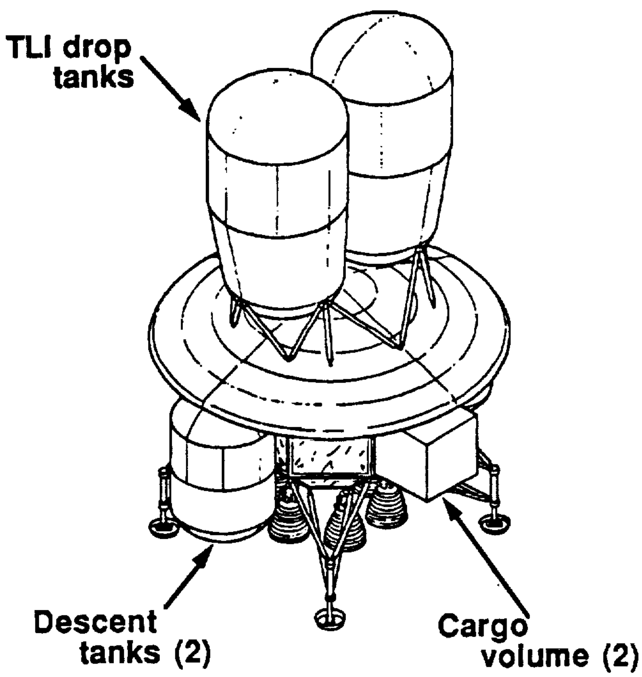
ITEM	Mass - kg W3	W2	CENTER OF MASS - m Xcg	Ycg	Zcg	MOMENTS (KG-M^2) IXX	IYY	IZZ
Mass Properties Summary								
CARGO		**11640**	**2.29**	**0.00**	**0.00**	**2.324E+05**	**1.838E+05**	**8.304E+04**
CARGO 1	5820		2.29	2.21	3.68	8.949E+03	1.310E+04	1.310E+04
CARGO 2	5820		2.29	-2.21	-3.68	8.949E+03	1.310E+04	1.310E+04
CREW MODULE		**11533**	**11.68**	**0.00**	**0.00**	**2.008E+04**	**2.008E+04**	**1.592E+04**
BALLISTIC RETURN CAB	8394		11.68	0.00	0.00	1.789E+04	1.789E+04	1.149E+04
EQUIPMENT PALLET	2031		11.68	0.00	0.00	1.911E+03	1.911E+03	4.153E+03
CREW	400		11.68	0.00	0.00	1.000E+02	1.000E+02	1.000E+02
EVA SUITS	400		11.68	0.00	0.00	1.000E+02	1.000E+02	1.000E+02
CREW CONSUMABLES - total	308		11.68	0.00	0.00	7.700E+01	7.700E+01	7.700E+01
TANKSET- ASCENT		**19834**	**3.11**	**0.00**	**0.00**	**6.257E+03**	**4.598E+04**	**4.598E+04**
STAGE INERT	2217		3.66	0.00	0.00	6.257E+03	1.409E+04	1.409E+04
PROPELLANT	17617		3.04	0.00	0.00	0.000E+00	3.115E+04	3.115E+04
ASCENT P/A MODULE		**1791**	**-0.01**	**0.00**	**0.00**	**3.721E+03**	**3.412E+03**	**3.412E+03**
STAGE INERT	1791		-0.01	0.00	0.00	3.721E+03	3.412E+03	3.412E+03
LANDER		**3686**	**-1.10**	**0.00**	**0.00**	**7.612E+04**	**4.966E+04**	**4.966E+04**
LANDER STRUCTURE	2926		-1.00	0.00	0.00	4.225E+04	3.186E+04	3.186E+04
LANDING GEAR (DEPLOYED)	760		-1.50	0.00	0.00	3.387E+04	1.765E+04	1.765E+04
DROP TANK SET-TLI (2 SETS)		**98394**	**2.58**	**0.00**	**0.00**	**1.759E+06**	**1.779E+06**	**8.528E+05**
TANKSET INERT	3150		3.70	2.03	-3.68	9.901E+03	2.894E+04	2.894E+04
PROPELLANT	46047		2.50	2.03	-3.68	-1.164E-10	1.900E+05	1.900E+05
TANKSET INERT	3150		3.70	-2.03	3.68	9.901E+03	2.894E+04	2.894E+04
PROPELLANT	46047		2.50	-2.03	3.68	-1.164E-10	1.900E+05	1.900E+05
STAGE - Descent (2 SETS)		**99766**	**2.52**	**0.00**	**0.00**	**2.105E+06**	**4.661E+05**	**2.550E+06**
TANKSET INERT	3877		2.82	4.57	0.00	1.063E+04	4.285E+04	4.285E+04
PROPELLANT	46006		2.50	4.57	0.00	0.000E+00	1.899E+05	1.899E+05
TANKSET INERT	3877		2.82	-4.57	0.00	1.063E+04	4.285E+04	4.285E+04
PROPELLANT	46006		2.50	-4.57	0.00	0.000E+00	1.899E+05	1.899E+05
TOTAL AT LEO ASSEMBLY		**246644**	**2.94**	**0.00**	**0.00**	**4.202E+06**	**3.541E+06**	**4.593E+06**
SEQUENCED MASS DATA								
LEO ASSEMBLY		**246644**	**2.94**	**0.00**	**0.00**	**4.202E+06**	**3.541E+06**	**4.593E+06**
START TRANS-LUNAR INJECTION		**247405**	**3.01**	**0.00**	**0.00**	**4.183E+06**	**3.674E+06**	**4.738E+06**
PRIOR TO LUNAR ORBIT INSERTION		**112122**	**3.54**	**0.00**	**0.00**	**1.644E+06**	**1.673E+06**	**2.857E+06**
LLO OPERATIONS		**89252**	**3.80**	**0.00**	**0.00**	**1.162E+06**	**1.536E+06**	**2.242E+06**
LUNAR LANDING		**59313**	**4.44**	**0.00**	**0.00**	**5.402E+05**	**1.325E+06**	**1.416E+06**
BEGIN LUNAR ASCENT		**33989**	**6.42**	**0.00**	**0.00**	**3.350E+04**	**7.268E+05**	**7.218E+05**
START TRANS EARTH INJECTION		**23006**	**7.99**	**0.00**	**0.00**	**3.004E+04**	**5.240E+05**	**5.191E+05**
START REENTRY		**11854**	**11.68**	**0.00**	**0.00**	**2.064E+04**	**2.067E+04**	**1.639E+04**

Figure 3.3.1-5. Ground-Based Vehicle Sequential Mass and Fluid Inventory

3.3.2 Operations Descriptions

Space-Based Vehicle. The selected space-based concept is a cryogenic vehicle with a reusable core stage and two pairs of expendable drop tanks, as shown in Figure 3.3.2-1. For piloted lunar missions, the core stage is flown with landing gear, a crew module, and a rigid space-assembled aerobrake. For unpiloted lunar cargo-delivery missions, the core stage is flown in an expendable mode without the crew module and aerobrake. The drop tanks for both missions include a pair of tanksets holding translunar injection (TLI) propellant and a pair of tanksets holding lunar-descent propellant. The vehicle has six main engines, allowing two engine-out capability during all mission phases.

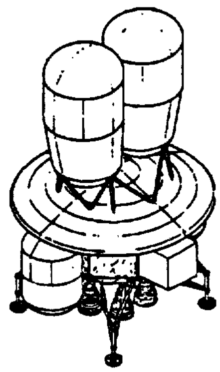
The lunar mission sequential configuration of the vehicle is depicted in Figure 3.3.2-2. The aerobrake must be launched in sections to fit in the launch shroud, assembled on orbit, and then attached to the core vehicle with the crew module.



Features

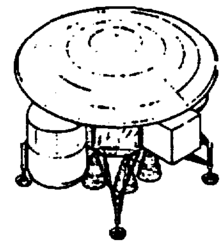
- **50 ft dia aerobrake**
- **Largest element 65 mt (TLI tankset)**
- **Reentry L/D >.2**
- **Asymmetric vehicle (offset crew module)**
- **Launchable in 30 ft shroud**
- **15 ft x ∞ cargo envelope (expendable missions)**
- **Recovery to SSF**
- **Reuse all high value elements**
- **Crew module fits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

Figure 3.3.2-1. Space-Based Vehicle



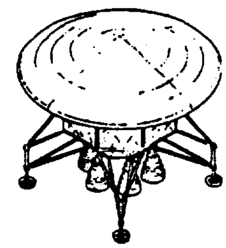
Mission start

Fully assembled configuration



Lunar arrival

TLI tanks discarded



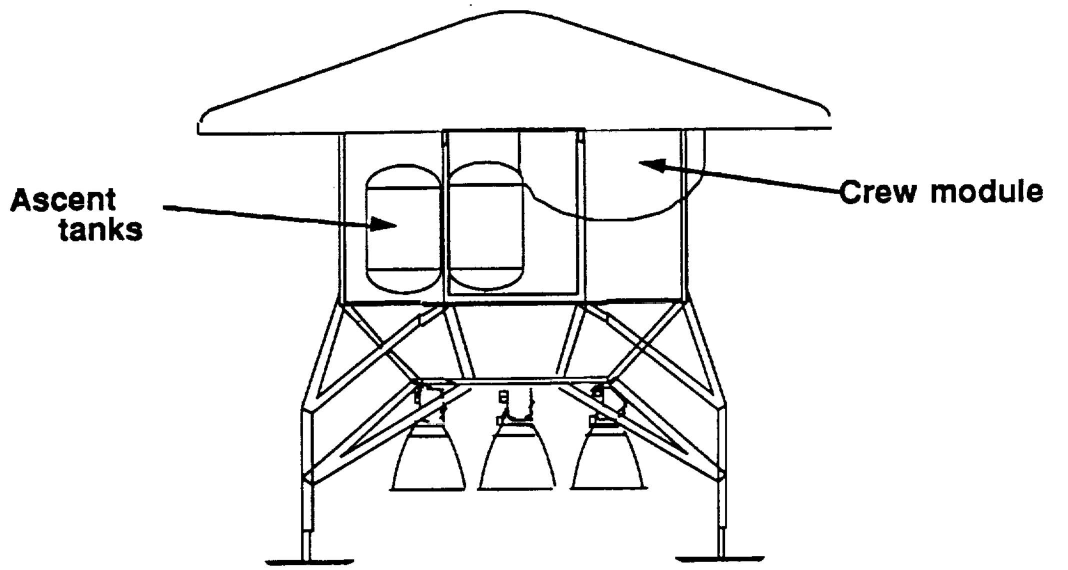
Lunar ascent

Vehicle core and aerobrake

Figure 3.3.2-2. Space-Based Vehicle Configuration Sequence

The crew module is offset from the vehicle centerline to provide lunar landing visibility and center-of-gravity offset for the aeromaneuver, as shown in Figure 3.3.2-3. The TLI tanksets, lunar descent tanksets, and cargo are launched in three to four launches and integrated with the core, and the core tanks are filled from a LEO tank farm prior to each mission.

During the mission, the TLI tanks are dropped after the TLI burn and the vehicle descends to the lunar surface following lunar injection. During descent, the core ascent tanks remain full, balancing the center of gravity to the centerline during the critical descent. For landing, the crew can view two landing pads and the horizon over the top of the cargo pallet. Upon arrival, the descent tanks are removed, the cargo is unloaded, the vehicle is hooked up to lunar surface support equipment, and the crew moves to the lunar habitat for the lunar stay. Because of the aerobrake overhang, cargo must be unloaded from the side of the core and moved to the base, either with built-in provisions or using a lunar flatbed trailer, as shown in Figure 3.3.2-4. At the end of the lunar stay, the crew loads return cargo, boards, and checks out the vehicle, then the core vehicle



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

Figure 3.3.2-3. Space-Based Crew Module Integration

ascends and returns to the LEO node, using an aeromaneuver, where it is inspected and refurbished for the next flight.

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

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

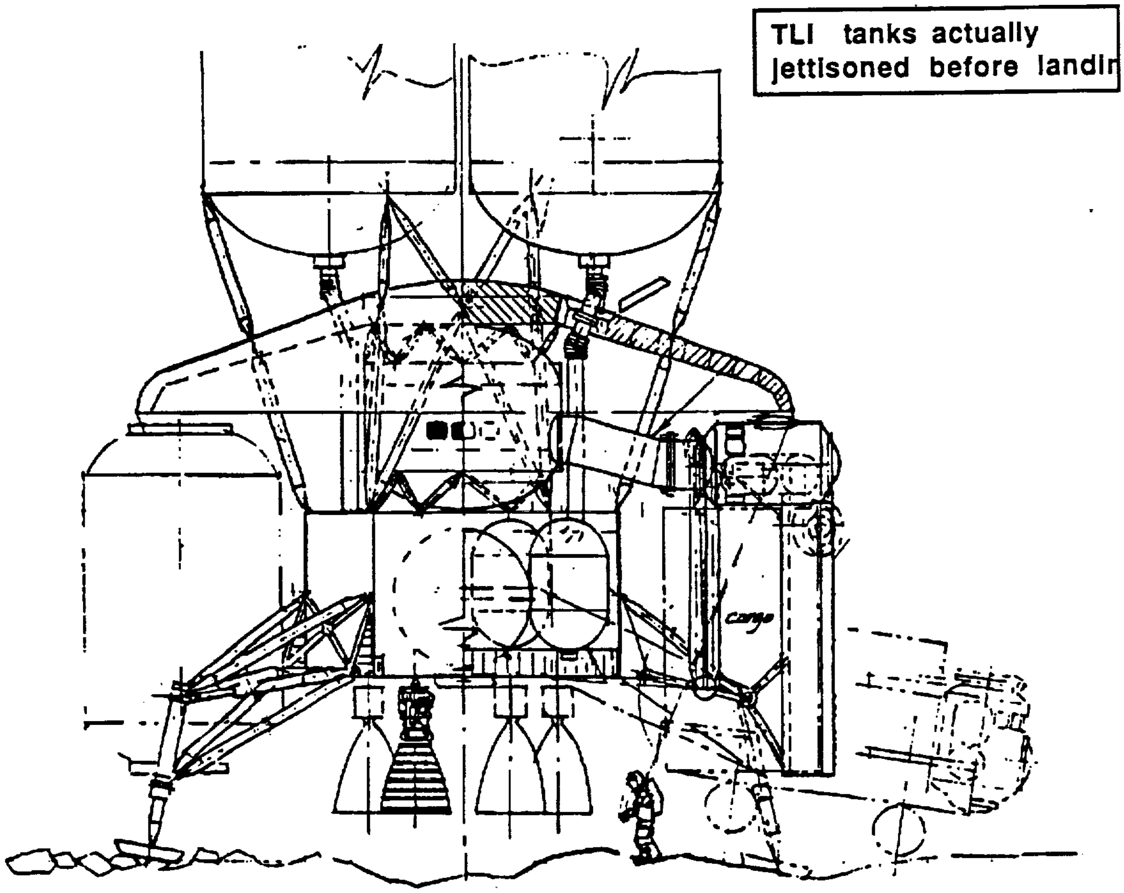
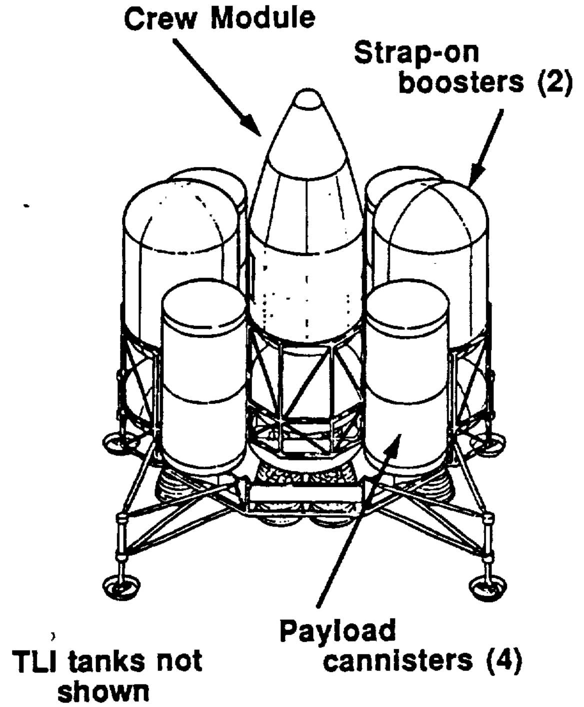


Figure 3.3.2-4. Space-Based Payload Unloading

Ground-Based Vehicle. The two selected ground-based concepts are cryogenic vehicles with a reusable crew module and avionics pallet, an expendable core stage made up of a propulsion module and tankset, a pair of expendable TLI drop tank sets, a pair of expendable delivery stages, and an expendable lunar lander platform. Figure 3.3.2-5 shows a single-launch concept in which all flight elements are launched full in a single HLLV launch. Figure 3.3.2-6 shows a concept in which most of the LO2 is launched in a separate launch and transferred to the main vehicle in LEO. In both cases, on-orbit assembly is minimized. The vehicles each have six main engines, allowing two engine-out capability during all mission phases.

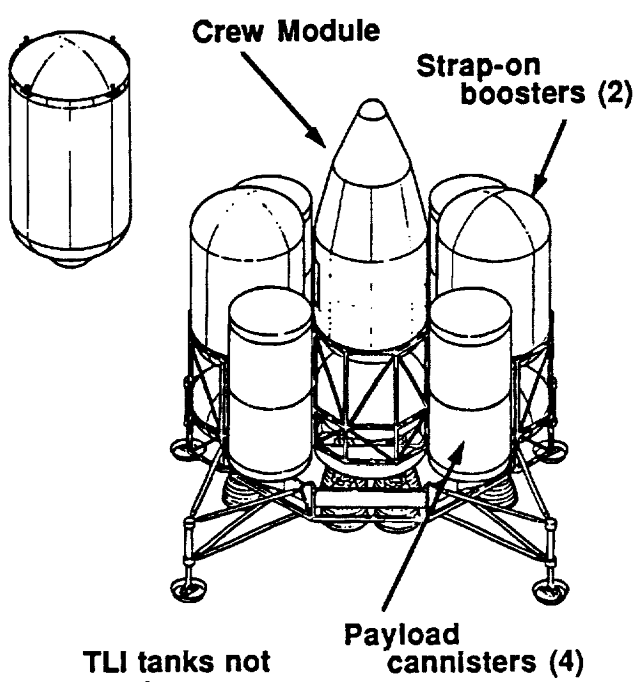
The on-orbit operations of the multiple-launch vehicle are depicted in Figure 3.3.2-7. The LO2 tanker is launched initially and remains on orbit until the core vehicle launch. The core vehicle is launched with a crew module escape



Features

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

Figure 3.3.2-5. Ground-Based, Single-Launch Vehicle



Features

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

Figure 3.3.2-6. Ground-Based, Multi-Launch Vehicle

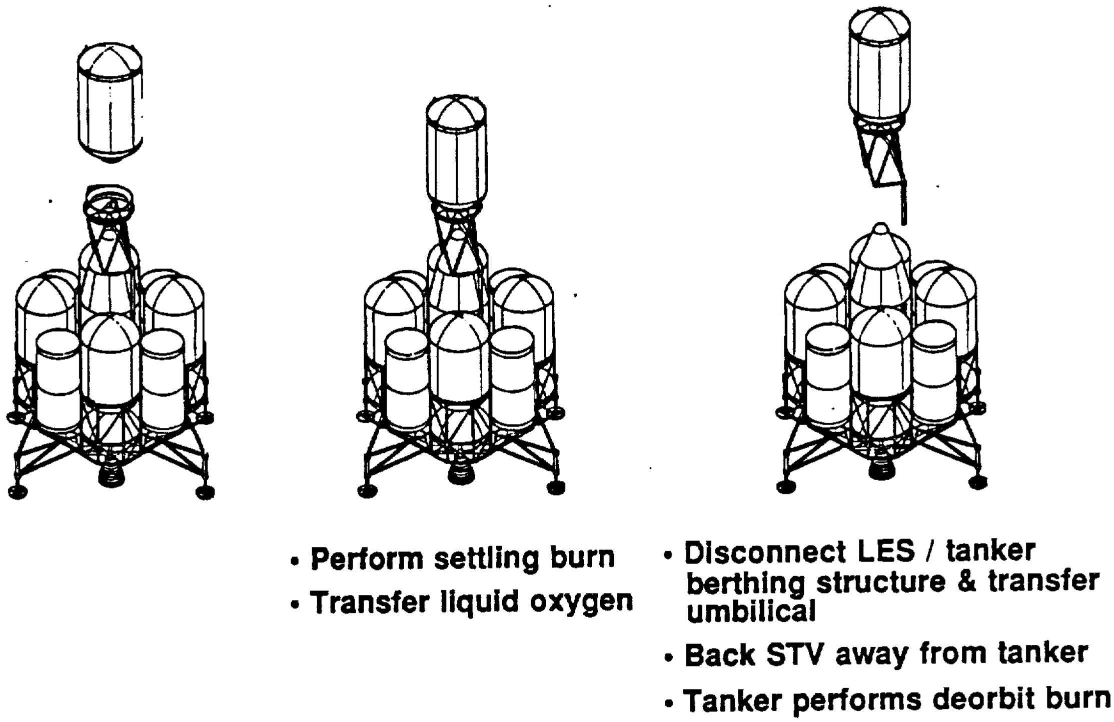


Figure 3.3.2-7. Ground-Based Orbital Refueling Operation

structure that includes a docking mechanism and tank fill provisions. It docks with the tanker, fills its LO2 tanks, and then jettisons the tanker, escape structure, and fill plumbing. From that point, both ground-based concepts are similar in mission configuration.

The common configuration sequence of the ground-based STV is shown in Figure 3.3.2-8. The TLI tanks are dropped after the TLI burn and the vehicle descends to the lunar surface following lunar injection with the lander, core stage, delivery stages, and cargo. During landing, the crew can view two landing pads and the horizon over the top of the cargo pallets. Upon arrival, the cargo is unloaded and the delivery stages, with one engine each, are either removed or tilted aside. The vehicle is hooked up to lunar surface support equipment and the crew moves to the lunar habitat for the lunar stay. Cargo can be unloaded from the side of the core, as shown in Figure 3.3.2-9, and moved to the base, either with built-in provisions or using a lunar flatbed trailer. At the end of the lunar stay, the crew loads return cargo, boards using a hoist, and checks out the vehicle. Then the core vehicle ascends, with the

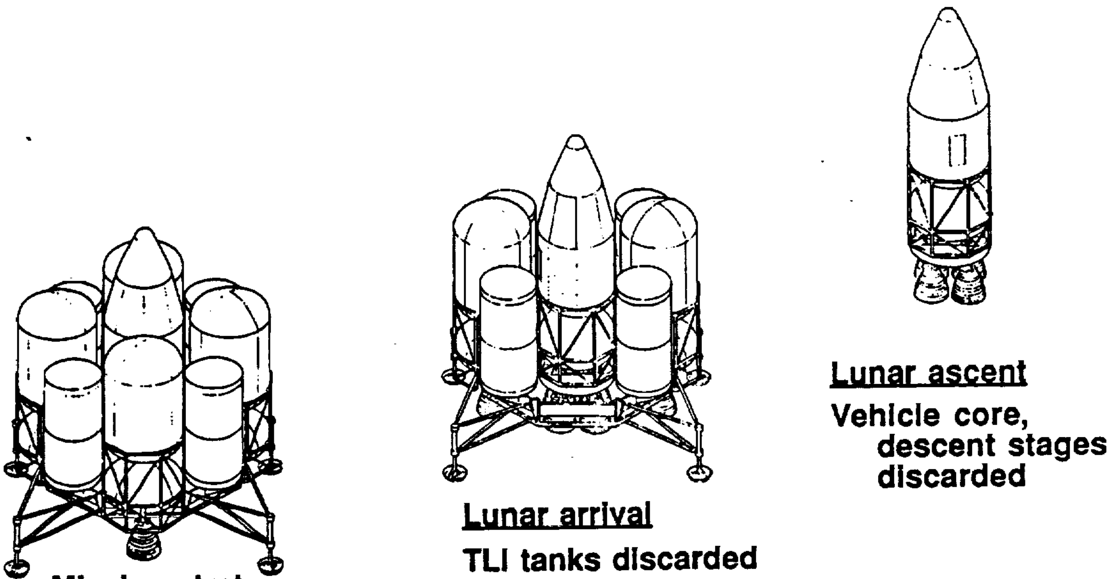
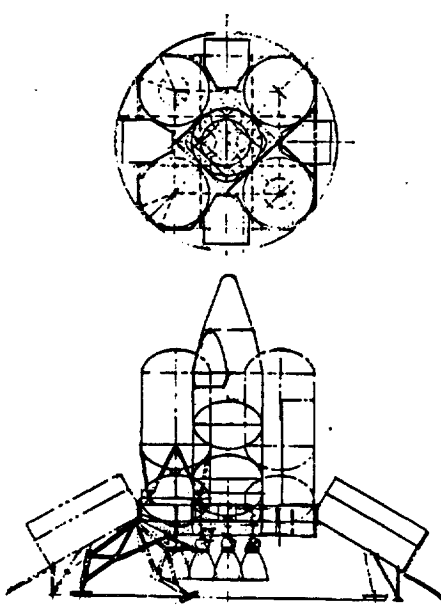


Figure 3.3.2-8 Ground-Based Vehicle Configuration Sequence

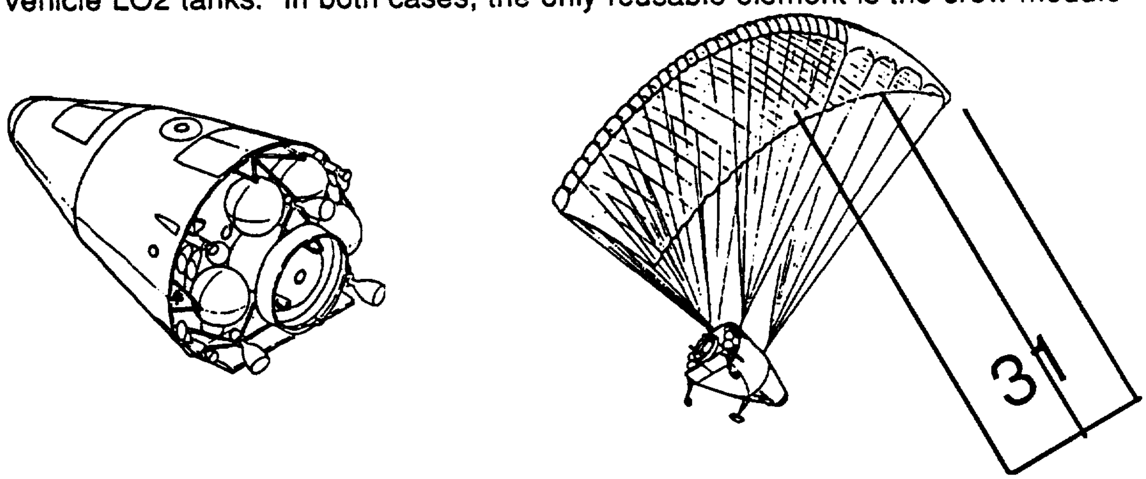


- Self unloading
- Cargo split into four pieces
- Ramps integral to cargo pallet
- Relatively low c.g. for landing stability

Figure 3.3.2-9. Ground-Based Payload Unloading expendable lander acting as a launch platform. The core stage is expended prior to reentry, and the crew module with avionics pallet reenters and lands near the launch site (Figure 3.3.2-10), where it is inspected and refurbished for the next flight.

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

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



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

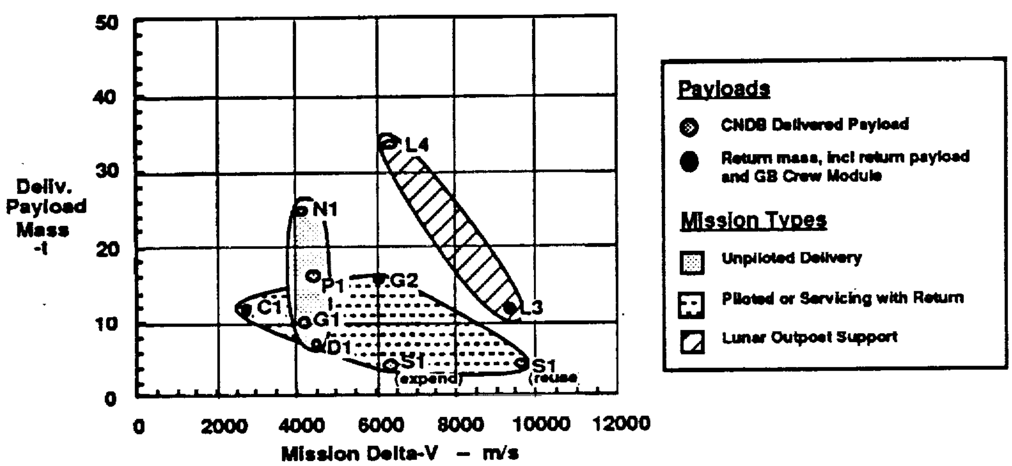
Independent RCS
Capable of surviving water ditch

Figure 3.3.2-10. Ground Recovered Crew Module with the avionics pallet, which reenters the Earth's atmosphere and returns to the ground where it is refurbished and reused.

3.3.3 Mission Capture

Evolutionary Mission Capture. The selected vehicle concepts are designed primarily to satisfy both piloted and unpiloted lunar missions, but other non-lunar missions may also be captured by elements of the lunar-designed vehicles. Specific mission requirements based on the 1989 civil needs database (CNDB) are given in Figure 3.3.3-1 in terms of mission type, delta-V requirements, and design payloads.

For the lunar missions, the delivered cargo assumed in the design is optimized for each vehicle, based on an optimum split between piloted and unpiloted missions, and is different from the CNDB amount on a per-mission basis.



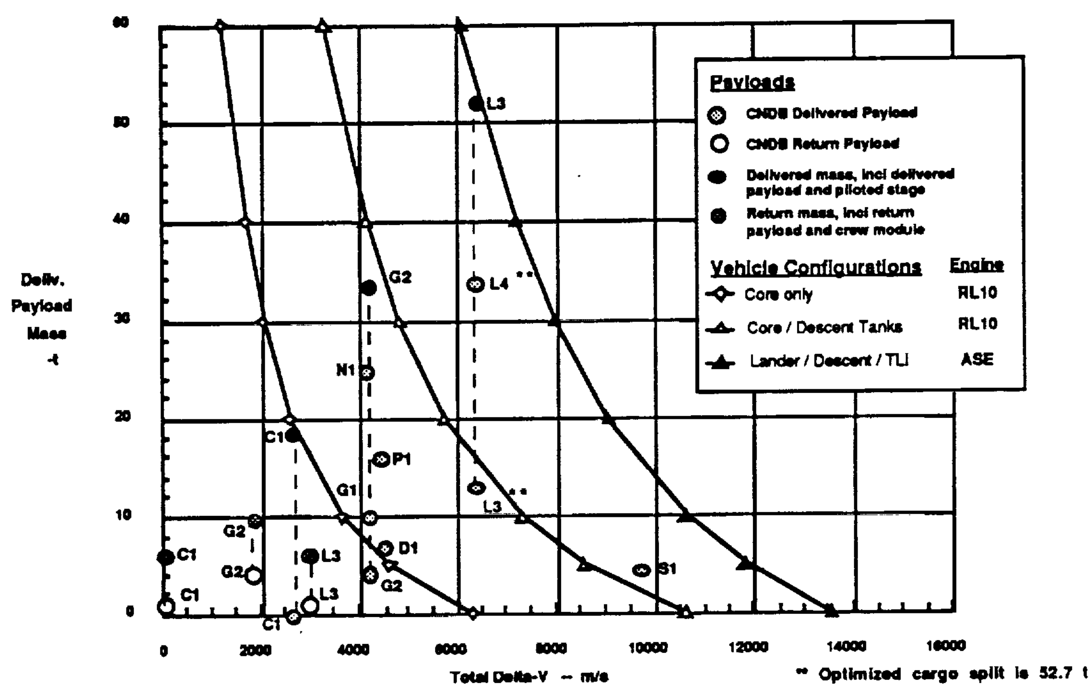
	Mission	Delta V (m/s) Delivery	Delta V (m/s) Return	Deliv. Cargo (kg)	Return Cargo (kg)
N1	Nuclear Debris Disposal	4175	0	25000	0
G1	GEO Delivery	4207	0	10000	0
P1	Planetary Delivery	4451	0	16000	0
D1	Molniya Delivery	4499	0	6800	0
S1	Polar Platform Servicing (GB)	6356	0	4500	0
S1	Polar Platform Servicing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

** Excluding Mass of Crew Module

Figure 3.3.3-1. CNDB Mission Requirements

Space-Based Vehicle. For capture of unpiloted non-lunar missions, the core stage of the space-based lunar-designed vehicle can be used as a delivery stage, without landing gear and with fewer, less-advanced engines, such as RL10's. For more demanding missions, descent and TLI tanksets can be added for larger propellant loads. Cargo delivery capabilities of various flight element configurations of the space-based vehicle are shown in Figure 3.3.3-2. Also shown are the design mission payloads and delta-V's.

Unpiloted delivery missions, shown as single points on the chart, are captured by the core stage with RL10's and descent tanksets, except for the lunar cargo delivery mission (L4) and recoverable polar platform servicing mission (S1), which require additional tanksets. Piloted missions with both delivery and return legs, are shown as groupings of delivered mass (including return stage, crew module, and delivered payload) to delivered payload and return mass



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

Figure 3.3.3-2. Space-Based Vehicle Element Capabilities (including crew module and return payload) to return payload. The core stage captures the sample return mission (C1) delivery and return legs, the lunar (L3) return, and the GEO servicing (G2) return. Additional delivery tanksets increase the capability to capture the G2 mission delivery, and additional TLI tanksets and landing gear increase the capability to capture the lunar piloted and cargo mission deliveries.

Ground-Based Vehicle. For capture of non-lunar unpiloted missions the delivery stage portion of the ground-based vehicle can be used as an independent vehicle with an avionics/power pallet and RL10 engine. For non-lunar piloted missions, the ascent stage with crew module and avionics/power pallet can be used. For greater capability, an ascent stage with avionics/power pallet and two delivery stages can be integrated onto a lander platform. Cargo delivery capabilities of various configurations of the ground-based STV concept, as well as CNDB mission payloads and delta-V's are given in Figure 3.3.3-3.

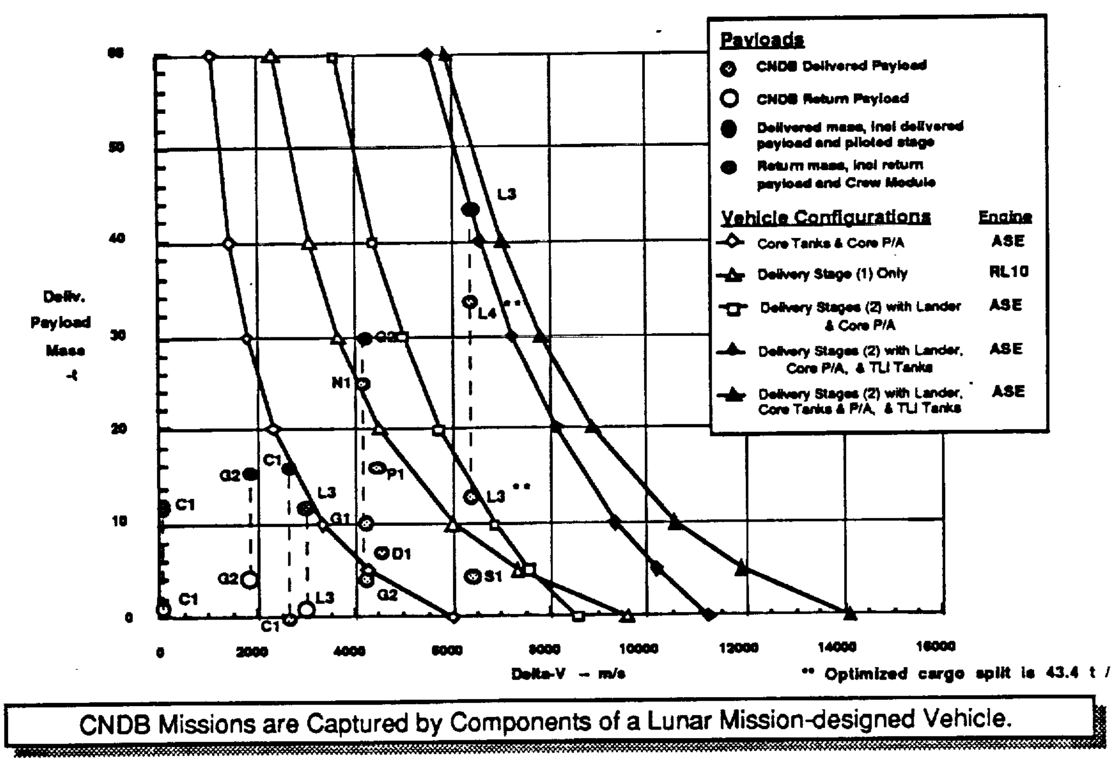


Figure 3.3.3-3. Ground-Based Vehicle Capabilities

Unpiloted delivery missions, shown as single points on the chart, are captured by a single delivery stage with RL10 except for the lunar cargo delivery mission, which requires the lunar vehicle with advanced engines. Piloted missions for both delivery and return legs are shown as groupings of delivered mass (including return stage, crew module, and delivered payload) to delivered payload and return mass (including crew module and return payload) to return payload. The sample return mission (C1) delivery and return is captured by the ascent stage only, as are the piloted lunar (L3) return and GEO servicing (G2) return. For the delivery leg of the G2 mission, a combination of descent stages and lander platform is required, and for delivery of the lunar piloted cargo mission, the full lunar vehicle is required.

3.3.4 Ground and Space Interface Requirements

The interface requirements for the three configurations studied are strongly affected by the assumptions that went into the subsystems, operations concepts, and external systems concepts of the three vehicles. Remembering that the three vehicles were developed with an eye toward the difference in ETO launch capabilities, this variation in capability has dramatic impacts on required interfaces. It is also important to remember that since the fundamental mission of the three vehicle concepts is identical, most of the interface requirements are common to the three vehicles. In other words, although the space-based vehicles may have an additional set of interfaces (i.e., with the Space Station), the vehicles all have a common set of interface requirements to payloads, lunar surface systems, and launch vehicle. An overview of these external interfaces is most clearly provided by considering location during different mission phases. These interfaces can be subdivided into ground processing, launch vehicle, low Earth orbit interfaces, and the interfaces between the vehicle and the lunar base.

Ground Interfaces. All three vehicle will use the facilities of and launch from the Cape Kennedy Space Center. Any differences between the concepts result from the size of the elements launched. In all cases propellant tanks, crew modules, lunar payloads, and vehicle stages (including avionics & propulsion) must be integrated, checked out and launched. Even the space "assembled" vehicle was anticipated to be preintegrated on the ground for fit and function, then broken into the 5 elements for launch to the station. New integration facilities were conceptualized and priced for each vehicle concept - details are contained in volumes II Book 3. The only significant concept unique facility requirement is for the (space based) aerobrake deployment and restow activities. The other major unique difference between the concepts from a processing standpoint is the space based concept will launch the crew module empty and unpiloted, where the other two concepts will launch the crew inside the module, and require special pad escape contingencies.

Launch Vehicle Interfaces. All three vehicles will require some type of heavy lift launch vehicle (HLLV). The size of this launch vehicle varies from a 71 metric ton booster for the space-based vehicle to a 125 metric ton booster for the ground-based, multiple-launch vehicle and finally to 250 metric tons for the ground-based, single-launch vehicle. In all of these HLLV configurations, the STV will require pad based propellant fill, drain and vent (cryogenic hydrogen and oxygen) with ground-based power and thermal conditioning, and potentially a "bent pipe" or RF "shroud window" for telemetry and command feedthrough while still on the ground. Last minute crew access will be required, but access for vehicle LRU changeout is still undefined. Current designs for future launch vehicles seek to minimize/eliminate on pad LRU changeout to enable lower cost & higher dependability. This could be accomodated by STV by having extra layers of redundancy and to "fly with failures" - however, that would require a major change in manned spaceflight practice.

The STV may also require power, thermal conditioning, and communications porting during launch vehicle flight (although power/thermal independance could be designed into the STV). For unpiloted launches a flight termination system (FTS) to provide for safe disposal of the vehicle in the case of launch vehicle failures will be provided. For the piloted flights a launch escape system (LES) is provided to separate the crew module from the rest of the STV/launch vehicle stack in case of failures. A flight termination system with a safety interlock system (to ensure no FTS initiation until after sufficient LES separation) would then destroy the stack. Carrying an FTS on piloted flights requires careful design considerations, but the current STS external tank and SRB range safety systems provide an example of currently flown systems.

LEO Interfaces. The requirements for LEO interfaces are the most significant difference between the three configurations studied. On the ground-based, single-launch vehicle, the vehicle has no requirements for any LEO interfaces. This vehicle is launched intact on board the 250 metric ton booster and hence may only to stay in LEO long enough to perform systems checkout before departing for the Moon, if it needs to establish an earth orbit at all.

The ground-based, multiple-launch vehicle is launched on board a booster that is not large enough to launch the entire fully fueled vehicle into orbit. Consequently, this vehicle will be launched fully integrated but without approximately 80% of the liquid oxygen required to go to the Moon. The remaining oxygen will be launched on board an oxygen tanker. This provides for essentially equal mass elements (when the launch vehicle shroud is taken into account - the tanker requires a much smaller shroud).

This oxygen tanker, as currently configured, will be an independent space vehicle that will be launched first, stationkeep in LEO, then dock with the STV and transfer fuel. However, the tanker could also be attached to some type of space tug (cargo transfer vehicle) that would provide the vehicle with any reaction control subsystem, propulsion, power, and avionics capability that the tanks require. The docking interface (shown in figure 2.1-5) uses the launch escape system structure to transmit the structural loads to the vehicle structure. (The launch escape rocket detaches after the launch vehicle passes through 300,000 feet altitude.) On this structure is mounted the fill lines to the main oxygen manifold. All the vehicle oxygen tanks are filled using this manifold, partial gravity can be supplied by propellant settling burns of the reaction control system. After emptying the oxygen, the tank is deorbited into the earth's atmosphere where it burns up.

The space-based vehicle has the most extensive LEO interface requirements of the three vehicles. This vehicle design requires a node that will have the capability to assemble, store, refuel, and refurbish the vehicle. As such, the node will have to be able to provide element manipulation, debris shielding, propellant handling and conditioning, subsystem LRU changeout, and consumable remanifest capability.

Lunar Surface Interfaces. The current level II systems requirements document (SRD) requires the lunar outpost provide simple beacons as landing aids for unpiloted STVs and lights and markings for piloted flights. Initial unpiloted STVs must land a certain number of times without these aids; and current navigational accuracy capabilities show the STVs may not require the beacons to meet current landing precision requirements (land within 5m of a designated point).

Of the other requirements levied by the STV on the lunar outpost systems, the most significant is that the STV will require subsystem support for stays longer than 30 days. (Current plans provide for up to 6-12 month stays on the lunar surface.) The chief subsystems needing support are communications, power, cryogenic fluid management, and meteroid protection. After the crew leaves the STV vehicle for an extended stay at the lunar outpost, the vehicle will be powered down (to a maintenance level) and covered with an enclosure to provide additional meteroid protection. Bringing this protective mass to the moon once, and using it during all subsequent vehicle stays is much more efficient than designing each vehicle for 6 months of unprotected exposure. Likewise, the systems to maintain the cryogenic hydrogen for the 6 month stay can be more efficiently be brought once than penalize each flight. These support systems can be thought of as a lunar "ground cart" equivalent to similar launch pad ground carts that provide temporary support better left off the vehicle.

As a design goal, the lunar outpost will not be required to provide planned maintenance beyond placing the STV into storage mode. However, expended STV elements will be cannibalized for use by the lunar outpost. Interfaces and elements will be common between STV and the lunar outpost systems whenever possible.

The lunar base will provide for a relay to Earth of STV telemetry whenever the STV is on the surface. The lunar base to STV link will be a safe system, such as low-power RF. Otherwise the STV would be required to broadcast directly to Earth which may present a hazard to an EVA crew.

Lunar Mission Payloads. The STV will transport essentially all of the equipment required for the buildup and maintenance of the lunar outpost. The outpost element design and the STV cargo capabilities are therefore ineluctably tied. In fact the one constant between the three STV concepts has been the cargo carrying capability: approximatately 50 metric tons for the cargo expendable mode, and 13 tons for the piloted reusable mode. Cargo offloading is illustrated in figures 1.1-1 and 1.1-2, with additional detail in Volume II, Book 1.

The lunar outpost equipment description was taken from Option 5 of the Lunar Initiative and were based on an informational data book written by NASA-JSC (Initial Study Period Results Summary - Planet Surface Systems - Conceptual Design and Development Requirements) defining the 27 STV flight manifests and planetary surface systems to be taken as cargo by the STV. The primary interface with the cargo will be structural and status data - which will be transmitted to earth interleaved in normal STV vehicle telemetry. While no active thermal cooling interfaces are required, some pointing or thermal roll attitudes may be required for the cargo to properly cool itself. Power requirements can be met either with a cargo pallet kit, or additional STV fuel cell reactants and associated tankage - which reduce payload capacity accordingly.

3.4 PROGRAMMATICS

3.4.1 Development Schedule

STV Phase C/D Program Schedule.

The Space Transfer Vehicle Phase C/D Program Schedule was developed by conducting a detailed assessment of all major tasks required to accomplish Full Scale development and launch of the first three STV test vehicles. Major milestones which drive the overall schedule include Preliminary Design Review (PDR), Critical Design Review (CDR), Subsystem Qual complete, Small Vehicle Pathfinder complete, 1st and 2nd STV Test Flights, LTS Qual and Pathfinder complete and the LTS Test Flight. A key requirement is successful integration of critical technology tests and demonstrations into the system design process (ref. Vol. II Book 4). which has been a major planning focus for this study. Interrelationships between specific activities are identified by the STV Phase C/D Logic Network which also highlights two separate critical paths. The primary critical path goes through the Advanced Propulsion activity while the Software Development schedule establishes the secondary critical path. All major tasks as identified in the Work Breakdown Structure (WBS) are included and grouped together in the following vehicle process elements:

LTV Core Stage- Includes design, fab assembly, and test of all subsystem elements required to support integration of the first two STV test vehicles and the LTS test vehicle. Two separate Core Propulsion schedules are identified. The modified RL 10-A4 Engine development results in a qualified engine available for the first STV test vehicle. The advanced engine schedule identifies what is required to provide new 480 Isp engines for the LTS test vehicle. The primary critical path flows through the propulsion schedule.
design and development of the STV flight software as well as software test facilities to support each STV test vehicle. The mission control software schedule is required to support the first STV vehicle pathfinder activity.

LTV Drop Tanks- Includes design, fab, assembly, integration and test of all elements of the fuel tank sets required for each STV test vehicle. The physical tanks must be integrated with the avionics, attitude control, and fluid supply subsystems prior to final assembly and integration with the core vehicle.

LTV Tanker- Includes design, fab, assembly, and test of each component of the Tanker subsystem. As with the Drop Tanks, the Tanker hardware is integrated with The attitude control, avionics, and fluid supply elements to support two separate Tanker set launches 20 days prior to the 2nd STV and 1st LTS test flights.

Crew Module- Identifies the schedule requirements to procure, design, fab, assemble, and test each element of the Crew module in support of the 2nd STV and 1st LTS test vehicles. Key elements of the Crew module include Structures & Mechanisms, Thermal Protection, Avionics, Flight Software, Environmental Control & Life Support System (ECLSS), and Launch Escape System. The initial crew module must go through a qualification process following assembly, integration, and test.

LTV Final Assembly- Identifies the schedule required to accomplish final assembly of all elements previously discussed for each STV test vehicle. This includes System Qualification and Pathfinder activities for the first STV and LTS vehicles. A Flight Processing activity is also scheduled prior to each flight to allow for final integration testing, checkout and launch Pad preparation.

3.4.2 Technology and Advanced Development Program

The Space Transfer Vehicle program provides both an opportunity and a requirement to increase our upper stage capabilities with the development and application of new technologies. Issues such as man rating, space basing, reusability, and long lunar surface storage times drive the need for new technology developments and applications. In addition, satisfaction of mission requirements such as lunar cargo delivery capability and lunar landing either require new technology development or can be achieved in a more cost-effective manner with judicious applications of advanced technology.

During the STV study, advanced technology development requirements and plans have been addressed by the Technology/Advanced Development Working Group composed of NASA and contractor representatives. Figure 3.4.2-1 provides a list of the technology categories and Figure 3.4.2-2 shows the priority placed on those technology categories for either the space-based or ground-based options. Sections 4-1.0 and 4-2.0, Volume II cover the technology assessments and schedules for the STV program.

Main Engine. The objectives of main engine technology development are to develop deep throttling required for lunar landing, increase Isp and thus reduce IMLEO requirements, support reusability and low maintenance requirements, ensure compatibility with long-duration space exposure, provide engine instrumentation that supports vehicle health monitoring (VHM) requirements, and maintain or improve the reliability of current systems.

1. Aerobrake (Aeroassist)
2. Cryogenic Primary & Auxilliary Propulsion
3. Vehicle Cryogenic Fluid Systems
4. Vehicle Avionics, Electrical Power & Software
5. Vehicle Structures, Cryogenic Tankage, & Auxiliary Equip.
6. Crew Modules & Systems (Incl. Biconic Crew Module)

Current STV Study Analysis

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7. Evironmental Control & Life Support Systems (ECLSS)
8. Vehicle Fabrication, Assy., Servicing & Processing
9. Space Vehicle Orbit Launch & Mission Control
10. Vehicle Flight Operations (Incl. Launch Escape Systems)
11. Artificial Gravity
12..-13. Advanced Propulsion Systems (Not adressed in STV Contract)

Figure 3.4.2-1. Technology Categories Listing

1. Aerobrake (Aeroassist)
2. Cryogenic Primary & Auxilliary Propulsion
3. Vehicle Cryogenic Fluid Systems
4. Vehicle Avionics, Electrical Power & Software
5. Vehicle Structures, Cryogenic Tankage, & Auxiliary Equip.
6. Crew Modules & Systems (incl. Biconic Crew Module)

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Figure 3.4.2-1. Technology Categories Listing

The current state of the art in upper stage class LOX/LH2 main engines is represented by the latest upgrade in the RL10 engine family produced by Pratt & Whitney. Upgrades to the current RL10 capabilities are both required to satisfy STV mission requirements and reduce costs. Medium to deep throttling capability is required to initially provide (at 100% thrust) enough thrust to reduce gravity losses when leaving low Earth orbit (LEO) and then to hover, reduce thrust, and accomplish the lunar landing (requiring throttling down to 20% (5:1) of thrust). Deep throttling has been demonstrated but is not currently available in off-the-shelf engines.

For the space-based STV, maintenance considerations must be addressed because of the high costs of space-based maintenance. The preferred approach to accomplishing main engine maintenance is to remove and replace the engine as a line replaceable unit (LRU). This requires design and verification of a simple interface/attachment method. Conceptual designs of a carrier plate concept containing all propellant and electrical connections in one interface have been identified. Advanced turbomachinery and seal technologies will improve system launch readiness and availability for both STV and ETO systems.

Requirements for reusability, autonomy, and lunar surface stay times prior to engine restart for Earth return drive requirements for engine reliability and health monitoring capability. Additionally, the space-based concept requires compatibility with long-term exposure to space environments. Under an STV subcontract, Pratt & Whitney has identified RL10 health monitoring instrumentation concepts that will need to be further refined and demonstrated with highly reliable sensors and health monitoring architectures (e.g., dual redundancy). Reusability also drives requirements for multiple restart capability.

Isp increases can significantly reduce propellant requirements. Figure 3.4.2-3 shows ETO mass per mission as a function of Isp. A savings of 50 metric tons per mission is provided by an Isp increase from 440 seconds to 481 seconds. Increases in Isp at a given thrust can be accomplished by increasing the

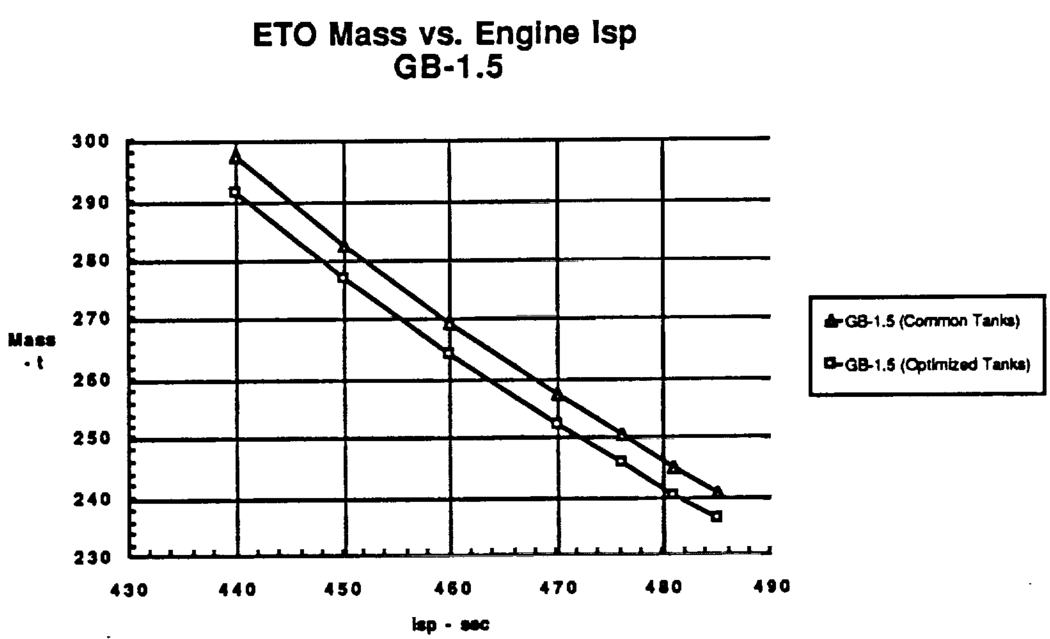

Figure 3.4.2-3. ETO Mass Versus Engine Isp

combustion chamber pressure (Pc) to between 400 and 1,500 psi and adjusting the area ratio accordingly. Increases in Pc will require advances in turbomachinery seals and bearings and chamber materials.

Two options for meeting these requirements were identified. The first option is an RL10 upgrade (the RL10B-2) with a maximum Isp of 470 seconds. The second option is an all new engine development. The RL10 upgrade is limited in terms of Pc increases and corresponding adjustments of the area ratio both by turbomachinery limitations and engine envelope considerations. Alternatively, development of a new engine is being investigated that will potentially provide Isp's up to 481 seconds at Pc's up to 1,500 psi. The difference in ETO mass required is approximately 12 metric tons more mass required per mission for the lower (470 seconds) Isp engine. Throttling capability, reliability, thrust level, engine restart, and other requirements for a new engine are all being currently defined through the STV study contracts and the engine workshop activity being conducted by LeRC.

Attitude Control. The objective in attitude control is to develop the technology required for high-performance gaseous O2/H2 thrusters in the 25- and 80-lb or in the 75-lb range that integrate well with the rest of the vehicle and are instrumented to support VHM. Propellants for the attitude control system (ACS) were not baselined, so there was first the question of propellant selection and, secondly, the selection of the required hardware. O2/H2 was the propellant selected for the STV ACS to be used with gaseous O2/H2 thrusters.

No gaseous O2/H2 thrusters are currently available. Work was initially started on this type of thruster for the Space Station. Thrusters from Rocketdyne, Aerojet, and Bell underwent substantial testing. Aerojet then had a follow-on contract to work on a thruster designed specifically for an 8:1 mixture ratio. The 8:1 mixture ratio was desired so that the electrolysis products of water at the Space Station could be used for propellants. Rocketdyne also had a follow-on contract for further thruster work.

In summary, sufficient capability to develop the thrusters is available. No major technical hurdles have been identified while the primary technology has been demonstrated. The thrusters will need to be developed for STV but there is a substantial database on which to draw.

Development will also be required for the attitude control system propellant accumulator. The accumulator will draw low-pressure propellant from common main propulsion/ACS tankage for accumulation as a supercritical fluid, high pressure in the ACS accumulator.

Cryogenic Fluid Systems. The objectives for cryogenic fluid systems technology development is to support STV mission requirements for long-duration storage of cryogenic fluids (e.g., on the lunar surface), in-space cryogenic fluid transfer, and make and break of cryogenic fluid connections (multiple times for space-based concepts). Tank venting and propellant-level gaging is also required. Additionally, a common fuel, common supply equipment, redundant, integrated propulsion/reaction control system has great potential to reduce operational complexity. Integration of the power system with the integrated propulsion/reaction control system is a possible growth candidate, but it was not baselined for the STV.

The state of the art in insulation systems is double aluminized Kapton blankets with Dacron net spacers between the blankets, also known as multilayer insulation (MLI). Use of foam and MLI insulation has been baselined for the STV. In this concept, an isocyanurate foam is either sprayed on (sprayed on foam insulation (SOFI)) or machined and attached to the cryogenic tanks with MLI attached over the foam. Foam and MLI provides better insulation than helium-purged MLI alone. Use of foam also allows a (less expensive and less operationally complex) GN2 ground purge instead of a helium purge. This combined concept provides good insulation on both the ground (Earth) and in space.

The state of the art in propellant gaging used to determine the amount of propellant in a tank is represented by either point sensors or capacitance probes. Both of these require the vehicle to perform a settling burn to move the propellant to one end of the tank. Several concepts for zero gravity propellant gaging have been proposed and investigated to some level. Acoustic methods have been examined but proved to be too sensitive to tank geometry.

Nucleonic methods are where radioactive sources are mounted on one side of the tank and sensors on the other side. The method proposed for the STV is the pressure-volume-temperature (PVT) method. With this approach, a small piston is used to rapidly perturb the tank internal volume. With the change in volume, a corresponding change in tank internal pressure and temperature is measured. These changes can be related to the internal density in the tank and the remaining propellant mass can be determined. PVT systems are being developed by Ball Aerospace for JSC.

The state of the art in cryogenic fluid connectors is represented by the STS external tank umbilical connectors. These connectors are not designed for multiple mate and demate operations required by the STV. New development is required in this area.

Similarly, there is no state of the art in zero gravity cryogenic fluid transfer. The current approach for the STV is to perform all required fluid transfers either during main propulsion burns or during ACS burns, which may include settling burns conducted for the primary purpose of aiding propellant transfer. Experiments have been planned (e.g., Coldsat) to support development in this area.

The final major area in cryogenic fluid systems of interest for STV is tank pressure control. The state of the art in this area is represented by the use of settled venting to control pressure buildup. The approach baselined for the STV is the thermodynamic vent system (TVS). With this concept, a quantity of gas and liquid mixture is expanded to the point where primarily gas remains, which is then vented. This expansion draws heat from the propellant remaining in the tank and thus primarily gas is vented and at the same time heat is removed from the tank. The TVS also provides for propellant mixing to reduce temperature stratification. A TVS was developed for the Shuttle/Centaur program and tested extensively on the ground. However, with the cancellation of the program, the TVS was not tested in space.

In summary, several advances are required for cryogenic fluid systems. MLI and foam combination insulation systems; PVT propellant gaging; reusable, reliable cryogenic connectors; and TVS pressure control are all baselined for the STV and need development. In addition, zero gravity cryogenic fluid transfer development would be applicable to the STV.

Aeroassist and Aeromaneuver. The ground-based STV concept requires a reentry crew module and the space-based STV concept uses an aerobrake for LEO capture. The objectives for technology development in the aeroassist and aeromaneuver category is to provide advances in thermal protection systems (TPS) both for heat protection requirements and for simplification of operational requirements. Development is required in tile and supporting structure materials, seals, and tile attachment methods. Thermal and stability modeling will also be required to verify and validate both biconic and aerobrake design parameters.

The shuttle represents the state of the art in TPS. Fibrous refractory composite insulation (FRCI) tiles are used over aluminum structure for protection from reentry heating, except for the wing leading edges, which are made from advanced carbon-carbon. AFRSI (or Q felt), a flexible quilted quartz cloth, is used in more benign regions.

A distinction must be made between different TPS approaches. Options include both hot and cold structure approaches. In hot structures, the thermal and aerodynamic loads are accommodated with the same structure. In the cold structure approach, the structure that carries the aerodynamic loads is different from the structure that carries the thermal loads. The shuttle uses a cold structure approach (FRCI tiles for thermal loads over aluminum structure that carries aerodynamic loads) except at the wing leading edges, which consist of the hot structure using advanced carbon-carbon.

Figure 3.4.2-4 contains a summary list of the materials with possible applicability for the STV. In general, ablators tend to be heavy, used in an expendable mode that requires additional refurbishment, and many contain organics that can outgas in space and pollute the local environment. Ablator thickness and material can be selected to support higher maximum temperature and heating loads than reusable tile materials but, based on the above considerations, were considered only if tile materials with the required capabilities were not available.

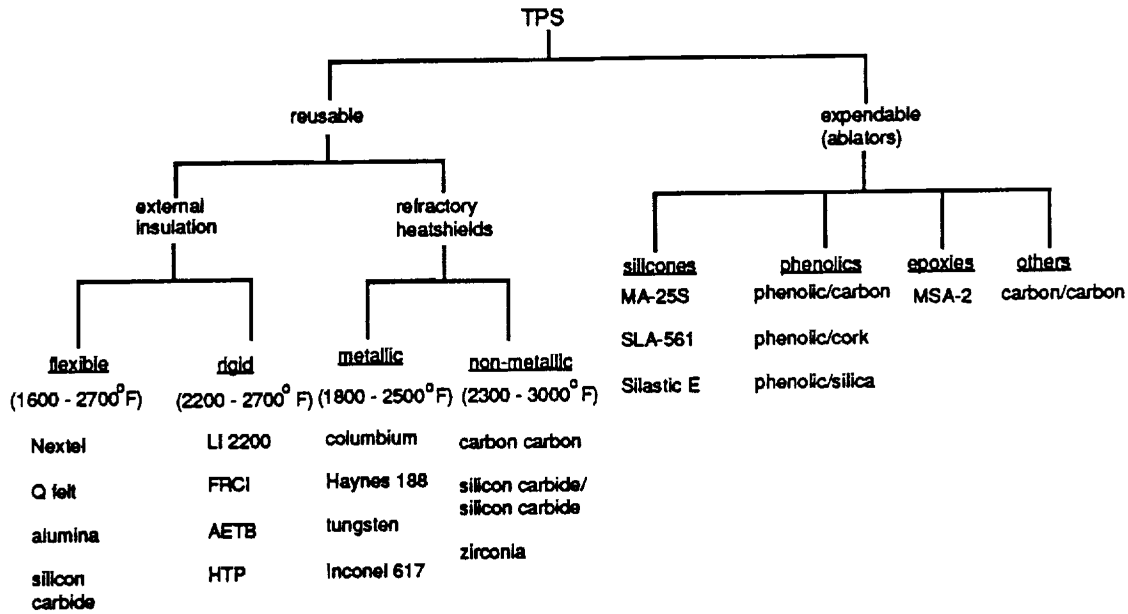

Figure 3.4.2-4. TPS Materials Options Tree

Zirconia was selected because it will satisfy the requirements for STV except for in the areas of the biconic nosecap. Options for use in this area include either an easily replaced sacrificial nose ablator or active cooling of the nosecap. One method of active cooling is to use Zirconia ceramic with a higher porosity than might be used elsewhere on the body and pump water through it to transpiration cool the vehicle nose.

Again, there are many material options for the underlying structure. The most commonly used material is aluminum as used on the shuttle. Aluminum presents a thermal expansion mismatch with the available tile materials requiring strain isolation pads between the structure and the tiles. Composite materials offer an alternative.

The STV baselined a thermoset graphite polyimide (Gr/Pi) structure with a Gr/Pi honeycomb core and Gr/Pi face sheets because of (1) surface temperature properties, (2) the capability to tailor the thermal expansion characteristics, and (3) the lightest weight.

Mechanical attachment of the Zirconia TPS to the underlying STV structure is desired to facilitate maintenance and refurbishment. Zirconia is a ceramic that is processed and manufactured at a relatively low temperature (approximately 800°F.) A high-temperature metallic honeycomb (e.g., titanium) can be cast into the back surface of the ceramic. Metallic face sheets with integral mechanical fasteners can then be attached to the metallic honeycomb and with these fasteners, connected to the underlying structure.

Conceptual designs of high-temperature seals have been developed for the NASP program. More work needs to be done in the areas of high-temperature seals and sealants for spaces between tiles.

Avionics. Avionics is probably the area of most rapid technology advancement. The objectives for avionics technology development are to develop highly reliable, low maintenance avionics capable of safe, autonomous operations and compatible with long duration space exposure. Reductions in avionics power requirements and weight are also desired.

Avionics Components. In application specific integrated circuits, the state of the art is very large integrated circuits. The baselined integrated circuits for the STV are radiation-hardened and single-event upset (SEU) tolerant chips and wafers. Radiation hardness requires special design, layout features, and materials. SEU results in a memory state change from passage of galactic cosmic ray particles (e.g., electrons, protons, and nuclei of all elements) or solar flare (lower energy protons and alpha particles) through the memory cell. Submicron CMOS/Silicon on sapphire (SoS) provides more radiation resistance than silicon alone. With the long durations in space and on the lunar surface, and subsequent extended exposure to radiation, SoS is seen as providing a substantial benefit to the STV.

Traditional integrated circuits are produced by slicing a wafer of material off of a cylinder of the material. The wafer is then usually cut into rectangles, circuits are etched, and then the leads are attached. With the use of SoS wafer scale integration, the full wafer itself is used and these wafers can then be stacked like pancakes. This will allow what once were several boards to be combined on one wafer, again with the resulting increases in reliability due to the reduction in outside connections. Processing density is increased and packaging is made much simpler, reducing volume requirements and enhancing maintenance, not to mention performance ranges of mini-supercomputers in a "tuna can."

Fiber-optic sensors in conjunction with sensor networks show promise in reducing cost and weight and increasing reliability. Fiber-optic sensors are impervious to electrostatic discharge problems and also enhance safety by reducing electrical components in the vicinity of hazardous fluids.

A promising area is the development of neural networks that, with input data from multiple sensors, can generate an output signature for specific flight elements or for the entire vehicle. For example, multiple temperature measurements across a tank can be combined within a neural network with the output consisting of a tank signature, such as OK or not OK, localized hot spot, or localized cold spot. Neural networks also show promise in the area of fault tolerance with the large number of processing sites and interconnectivity making a failure in any individual neural site relatively unimportant. Neural networks show promise as a component in the VHMS and in support of vehicle autonomy.

Navigation instrument advances have been baselined for the STV. The state of the art in gyros is represented by ring laser gyros (RLG). RLGs have flown on Boeing 757 and 767 commercial airplanes and have been used in space applications for the Ariane 4. RLGs use a single laser light source that sends laser beams in opposite directions around either a triad or rectangle of mirrors. A rotation of the RLG in the measured axis respectively lengthens and shortens the distance that the beams sent in opposite directions travel. This then creates a phase difference between the two beams of light that is measured and related to the rotation.

Fiber-optic gyros (FOG) work on somewhat the same principle as RLGs; however, a winding (≈ 400m) of fiber-optic cable is used for the light path instead of the mirror system. Dithering mechanisms, mirrors, and path length controllers are not required, as on the RLGs, thus reducing weight and, with fewer parts, increasing reliability. Also with the reduction in electrical parts, susceptibility to radiation induced SEUs is greatly reduced. The STV has baselined RLGs in a six RLG, six accelerometer skewed axis hexad inertial measurement unit; however, FOGs will be maintained as an option.

The state of the art in radar used for rendezvous and docking and potentially landing is the range/range rate Doppler radar used by the shuttle. STV will also use Doppler radar. Space-based Doppler radar as used in the shuttle is still fairly crude and will require further development. Advanced radar systems use a phased array of sensors place about the vehicle. By sensing the phase difference among the multiple sensors, angle, range, and rates are resolved. This eliminates the need for a bulky pedestal-mounted motor-driven dish. The STV may also include side-looking aperture radar (SLAR). SLAR is used on the Magellan spacecraft to map the surface of Venus. With multiple lunar orbit passes the SLAR can build a map of the lunar surface terrain height features to support landing decisions.

Avionics Networks. Sensor networks include fiber-optic sensors, smart sensors, analog multiplexing techniques, digital multiplexing techniques, network components (wire/fiber media and connectors), and network interface units. Benefits of using a sensor network of fiber-optic sensors or multisensors includes the reduction of weight and increases in reliability associated with reductions in the number of point-to-point wirings commonly used in current connection approaches. In addition, fiber-optic sensors can enhance safety when used to sense propellant levels without using electrical components within the tank.

Also of interest for STV are higher speed digital data buses. Data buses again reduce point-to-point wiring and the associated reliability and weight concerns. To support the concept of modular avionics, which allows technology changes, upgrades, and growth add-ons, the embedded networks must have high channel capacity and be very robust and damage tolerant. A zero-downtime network will support long-duration missions with multiple sorties without maintenance. Fiber-optic digital data distribution networks (100 to 1,000 MIPs, multiple wavelength, active redundant) with separation of flight critical data from non-flight critical data are baselined.

A third area of importance is the standardization of digital interfaces. Standardized interfaces support common test interface equipment and generally lower costs. In-space assembly or mate/demate/remate of flight elements is facilitated by standard interconnection systems.

Avionics Subsystem Areas. STV autonomy requirements and long durations spent in space will require a vehicle health management system (VHMS). VHM provides better vehicle availability and lower costs through automated launch processing. The operational status of the vehicle can be determined through all mission phases based on previous operating performance and built-in tests (BIT). VHM also supports maintenance with fault isolation.

The VHMS is essentially distributed throughout the entire vehicle. VHMS consists primarily of sensing and monitoring functions and management functions. Sensing is primarily the domain of the avionics subsystem with monitoring being either a hardware or software area. The management functions can be software-based mission management rules, as well as hardware based such as hardware-driven reconfiguration. VHMS is a concern of software, the entire avionics architecture, and individual avionics components. LeRC has conducted some workshop activity with the intention of supporting an initiative in this area.

The state of the art in fault tolerant systems is represented by the shuttle. The shuttle uses a combination of hardware and software for fault tolerance in the flight critical subsystems. For example, four computers with software voting are all active. Failed computers are identified to the flight crew deck and may be taken offline. The STV has baselined a primarily photonic avionics system using common avionics modules providing hardware voting. Application programmers will not need to provide software for the voting.

Finally in the area of communication and tracking, the STV has baselined a radio/laser communication systems. Laser provides radiation-tolerant, high-speed secure communications and will be used for vehicle-to-vehicle, vehicle-to-ground communications. Laser will also be used for docking. The state of the art in laser communications is Navy ship-to-ship antijam, secure links. Radio communication data rates have been steadily increasing due to electronics technology and data encoding schemes that increase the link margin between terminals. Adaptive communications provide a robust link in the presence of solar activity.

The state of the art in communications support satellite systems is represented by the tracking and data relay satellite system (TDRSS). TDRSS has the capability to handle communications in the 300-Mbps range; however, subtracting the communications overhead leaves an approximate 180-Mbps data rate available for sending information. Availability of TDRSS is also a concern. Advanced TDRSS (ATDRSS) will be able to support a much higher data rate, which STV will use to support color video, high amounts of scientific data transmittal, and so forth.

Software. Software is characterized by the avionics. The primary new technology objective in software development is to develop adaptive guidance, navigation, and control (GN&C) algorithms with appropriate validation and verification methods. Adaptive GN&C also needs to be integrated with the appropriate sensor networks.

The state of the art in software is that used in the shuttle software and avionics design. The shuttle used the HAL/S software language, a language developed for, and used only on, the shuttle. Some manned Earth aircraft have been built for the DoD agencies that incorporate new GN&C software, but these systems are classified or not man rated for space.

STV will be using the Ada language. Ada provides an Ada task scheduler, which is essentially a part of the executive program. The individual user must provide the additional executive functions required for the specific application. Ada is expected to reduce software maintenance requirements and provide commonality across development programs. Selected development and prototyping of the software is planned to reduce program risk.

Power. Objectives for power technology development are to support lightweight, reliable, power equipment development that integrates well with the vehicle, supports the total and peak loading requirements, and is instrumented to support VHM. The STV power subsystem baseline consists of fuel cells with lithium thionyl chloride (LiSOCl2) batteries for handling peak loads required by the thrust vector control (TVC) system. With this approach, the fuel cells do not need to be sized for the high TVC load levels required over an extremely small portion of the mission.

The state of the art in space-qualified batteries are the silver zinc (AgZn) batteries used on the IUS (and the shuttle). Currently in development by SAFT of France for Centaur is a 250 amp-hour LiSOCl2 battery. Planned for qualification in December 1991, the LiSOCl2 battery features an energy density of approximately 242 watt-hour/kg, which is almost twice that of the AgZn batteries with the resulting reduction in weight for the required power.

Fuel cells use an electrochemical reaction to produce power. Gaseous O2 and H2 come into contact with their respective electrodes and combine, producing power and, as a byproduct, potable water. The state of the art in fuel cells is represented by the shuttle fuel cells made by International Fuel Cells, a subsidiary of United Technologies. No technology development in the area of fuel cells is required for the power system baselined for the STV.

Crew Module Systems. Crew module designs need to be directed at special crew safety and comfort requirements of a deep space transportation vehicle. Crew modules must comply with NASA STD 3000 requirements. Additionally, crew module reuse will impose operational refurbishment requirements such as easy access and replacement. Effective radiation protection, high-reliability environmental control, fault tolerant backup systems, easily maintainable life support system elements, appropriate abort and emergency equipment, and sealed redundant control capability are required.

The state of the art in environmental control and life support systems (ECLSS) is represented by the shuttle. A cabin atmosphere of 79% nitrogen and 21% oxygen is maintained at 14.7 psi. The atmosphere is supplied by a redundant O2/N2 supply system and an additional O2 emergency supply system is provided. Atmospheric revitalization is accomplished by using fans to circulate the air through lithium hydroxide and activated charcoal filters. Trace contaminant control is accomplished with an ambient temperature catalytic oxidizer that primarily serves to remove any CO in the air.

Thermal control of the cabin is accomplished with a series of pipes and heat exchangers using water as the active fluid. Byproduct water (75 kg) from the fuel cells is stored in a tank and pumped through the supply/heat exchanger system. This water is then passed through a mid-fuselage water/Freon 21 heat exchanger and the excess heat is radiated to space through the Freon 21 radiators located in the cargo bay doors.

For fire detection and suppression, the shuttle uses ionization detectors located in each of the three avionics bays to sense combustion byproducts. Fire suppression is accomplished by means of both manual and switch activated fire extinguishers.

The state of the art in crew controls and displays is represented by reconfigurable liquid crystal displays (LCD). The LCDs can display graphical or numerical data and are driven by separate controllers for redundancy. The displays and pushbuttons are reconfigurable and would assist in reducing information overload by presenting only data applicable to the current flight phase. This technology requires low power and is state of the art in current military and commercial systems.

Primary areas requiring development for the STV are in the crew controls and displays and in development of a new, lightweight, reliable commode.

Structures, Tankage, and Auxiliary Equipment. Current structural technologies are based on the shuttle and advanced military airplane structures that are proprietary or classified. Development of structural materials and methods will be required for the STV. The STV will use aluminum-lithium (Al-Li) tankage with composite interstages. Al-Li provides a lighter, stronger tank structure. Development and verification of Al-Li welding processes is required and is being conducted for the HLLV program.

Auxiliary equipment such as disconnects and advanced materials fasteners are required. The aerobrake requires large hinges for packaging in the HLLV payload shroud and will need reliable hinges and deploy mechanisms. Cryogenic, electrical, and structural disconnects are required for tank staging and GB/GO concept vehicle staging of landing legs and engines.

A launch escape system (LES) is required for the GB/GO option. This system will act to pull (or push) the crew and crew module away from the launch vehicle in the event of a booster failure. LES development will be required for a system that integrates with both the STV and the booster and provides escape for the crew in time and within acceleration limits. Sensing of imminent booster failure will be required for activation of the LES.

3.5 COST (1989 DOLLARS)

Executive Summary of Cost Estimates. The STV program is developed to capture both lunar transportation system (LTS) and other CNDB missions. The program life cycle costs are estimated separately for the LTS missions (lunar trips with additional space tug missions for SEI).

The other CNDB missions are proposed to be captured using a small stage derivative of the LTS flight hardware. A summary of the three LTS configuration life cycle cost estimates is presented in Figure 3.5-1. Further details of the summary life cycle cost data are available for review in Volume III, Book 1 of the Boeing final report.

The final three Boeing LTS candidate configurations are presented as a comparison set. Each final 1.5-stage configuration candidate has merits for the final selection. The final configuration choice will be dependent on the outcomes of national space program decisions concerning the Space Station Freedom (SSF) and Heavy Lift Launch System (sometimes referred to as the National Launch System) projects.

Each LTS configuration requires a unique space or ground infrastructure systems support to operate. The life cycle cost estimates in Figure 3.5-1 assume that the Space Station and/or launch system elements already exist by the first LTS mission.

The additional costs to upgrade the Space Station (low earth orbit node) and an advanced launch vehicle system (ETO) are included in each of the the LCC estimates. The low Earth orbit (LEO) node and ETO modification estimates are based on current Advanced Launch System LCC model data and cost

WBS Number	WBS Title	Space Based	Ground-Orbital	Ground Based
7.1	NASA Mgmt.	1.405	1.584	1.451
7.2.1	Prog. Mgmt.	9.556	11.541	10.157
7.2.2	System Engr.	0.310	0.203	0.179
7.2.3	Flight Hardware	19.163	23.454	20.319
7.2.4	Support Equip.	0.403	0.348	0.308
7.2.5	Payload Accom.	(TBD)	(TBD)	(TBD)
7.2.6	Software (Flt.)	1.500	1.875	1.775
7.2.7	System Test Op.	.882	.508	.473
7.2.8	Ground Ops. N/R	.586	.358	.358
7.2.9	Mission Ops. N/R	.296	.296	.296
7.3	LTS O&S (Veh.)	5.171	8.600	8.348
3.0	ETO (HLLV)	52.434	28.053	23.447
4.0	LEO Node (SSF)	6.674	0	0
Total LCC -	Project Total	$ 98.380 B	$ 76.820 B	$ 67.111 B

(1991 dollars in billions)

Figure 3.5-1. Lunar Transportation System LCC Estimates

estimates from the Boeing Advanced Civil Space Systems group and General Dynamics Infrastructure studies (being conducted for NASA-MSFC).

Final ETO cost estimates were developed using dollars per pound factors provided by the NASA program office. For smaller HLLV launch booster system flights delivering approximately 71 metric tons costs are estimated at $2,500/lb of LTS payload. For large HLLVs, with 125 to 250 metric ton payload capability, ETO cost is estimated using an average recurring flight cost factor of $1,300/lb of LTS payload

C.2

LTS Development Estimates Summary. A development estimate for each Boeing LTS candidate configuration (including small stage derivative development) was created using a Boeing proprietary parametric cost model. Each estimate is time spread to the program development schedule shown in Figure 3.5-2. The DDT&E estimates include phase B and phase C/D program costs.

The time spread expenditures profiles for each Boeing LTS configuration development program is contained in Figures 3.5-3 through 3.5-5. In each case, the estimated peak expenditure for development funding does not exceed $4.5 billion (in constant-year, 1991 dollars).

LTS Development Estimates Comparison With Apollo. As a reasonableness check, a comparison profile of Apollo program development costs (in 1991 dollars) with the Boeing LTS candidates is presented in Figure 3.5-6. The comparison excludes prior Apollo engines and advanced space engines development for the new project cost estimate distributions. Thecomparison shows that the Apollo command and service module, lunar module, and Saturn IV-B stage development costs, in 1991 dollars using NASA source data and escalation tables, were much higher. The higher Apollo costs are a result of a shorter development schedule (7 years) and a smaller technology experience base (first time development with less in-space and lunar surface environment experience).

LTS Development Cost Risk Analysis Summary. A cost uncertainty analysis is also required to provide a cost risk evaluation of the Boeing parametric cost estimates for LTS program development. Figure 3.5-7 data show that in all cases the estimated development costs still fall below the Apollo historical cost experience total and peak funding in equivalent year (1991 dollars).

All estimates are developed with top-level NASA planning information and contractor interpretation of the 90-day study requirements. The mission need requirement for a future STV lunar transportation system with a large 34 metric ton payload capability also drives the Boeing development and life cycle cost estimates.

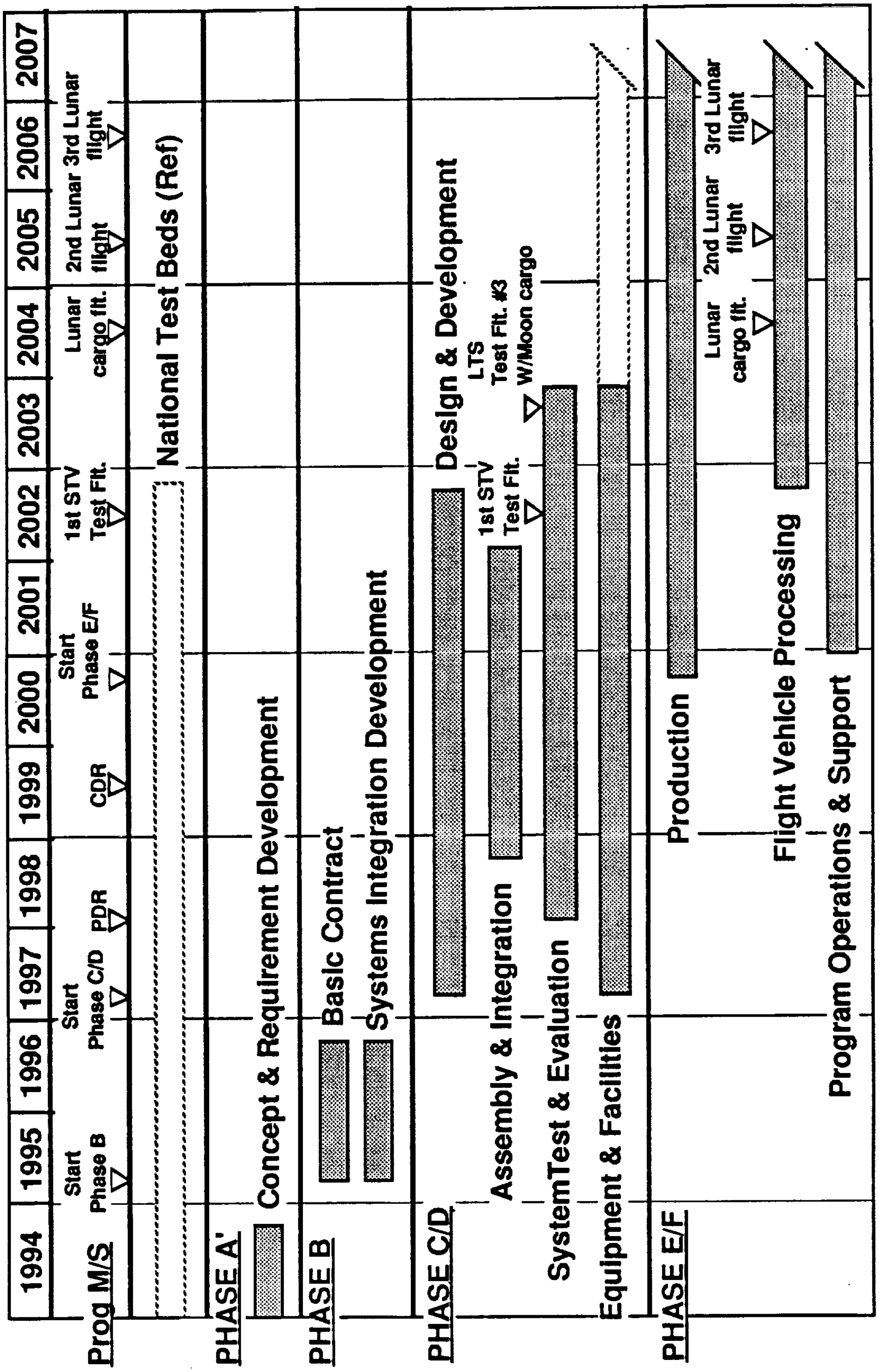

Figure 3.5-2. Program Master Schedule

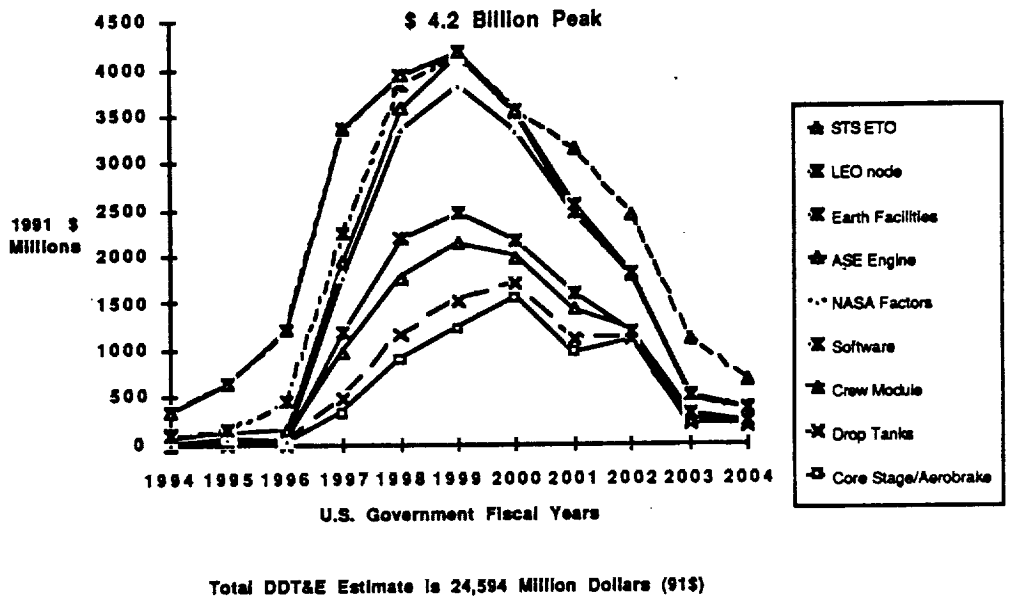

Figure 3.5-3. Space-Based LTV (at SSF) DDT&E Fiscal Year Funding Profile

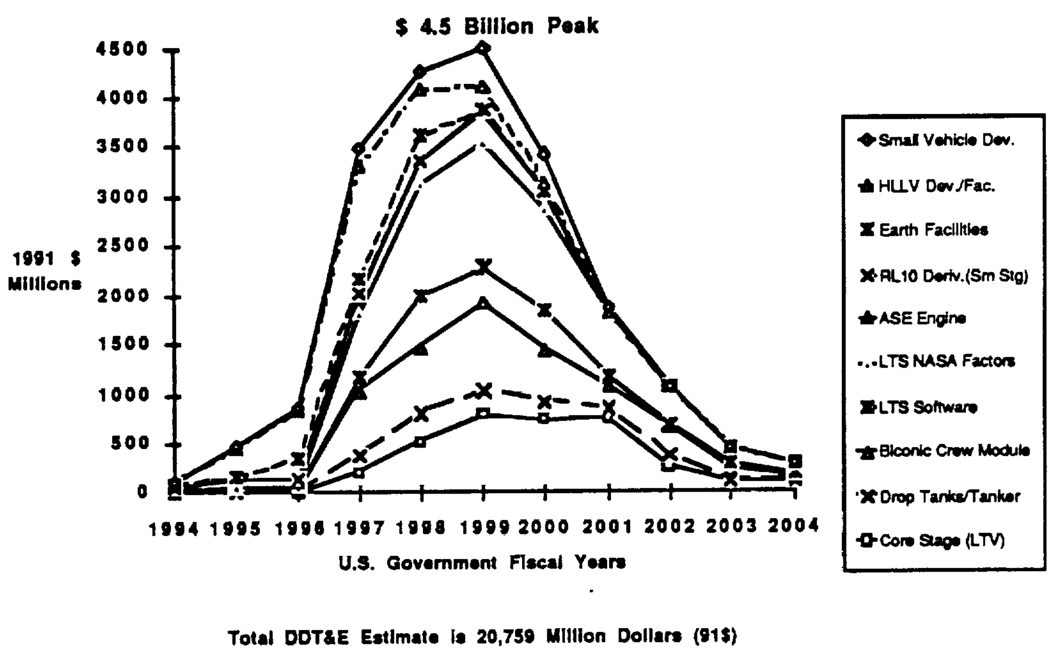

Figure 3.5-4. Ground-Orbital LTV DDT&E Fiscal Year Funding Profile

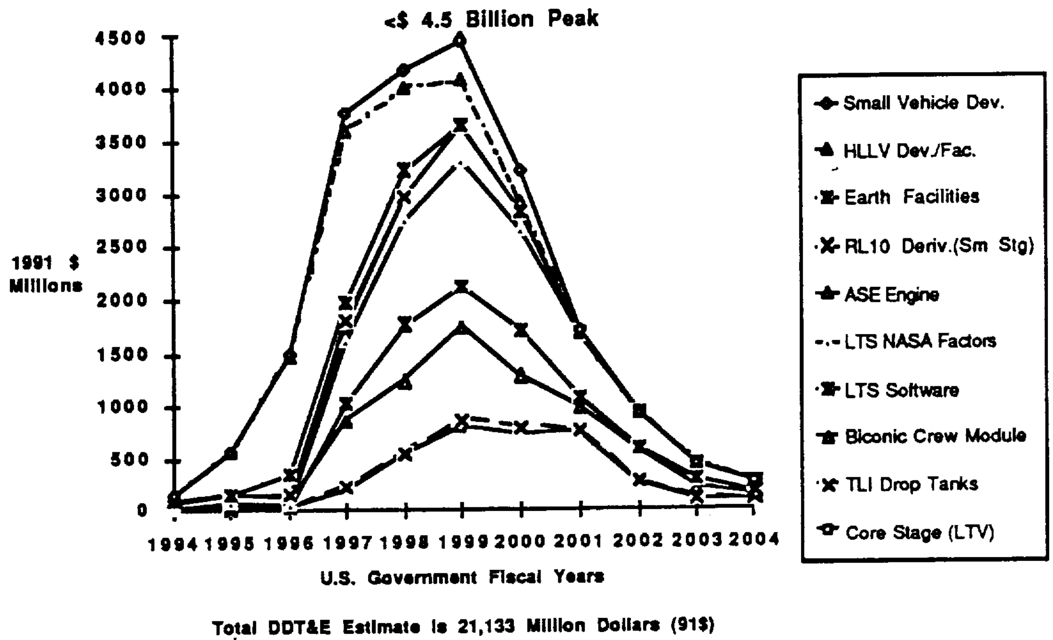

Figure 3.5-5. Ground-Based LTV (1 Launch) DDT&E Fiscal Year Funding Profile

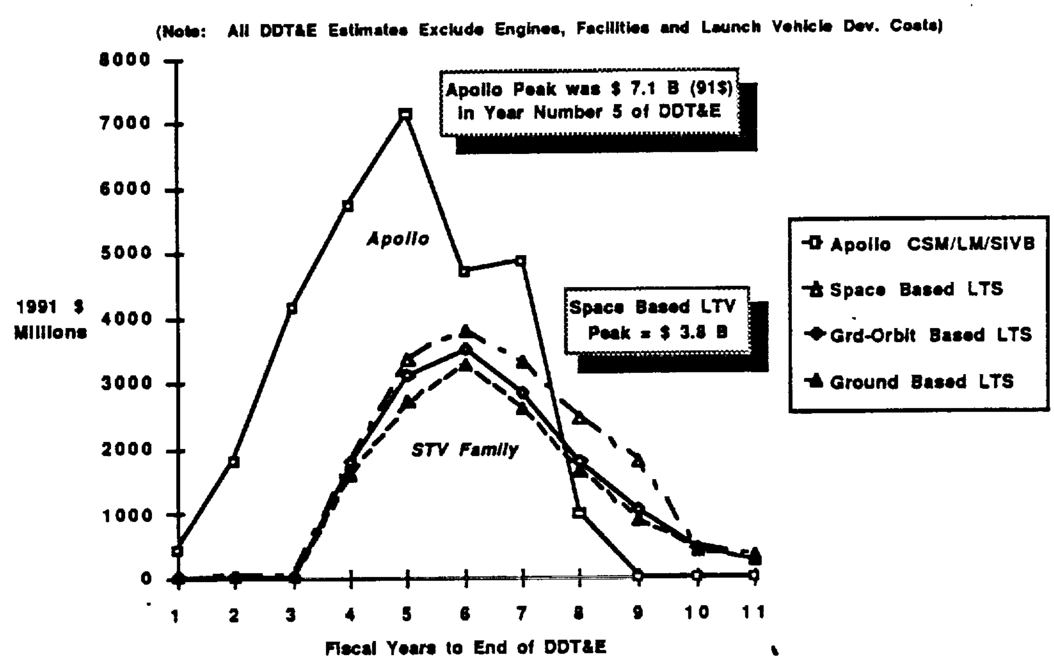

Figure 3.5-6. Comparison of Apollo Hardware DDT&E to Boeing LTS Vehicle DDT&E Estimates

(1991 Dollars in Millions)

	DDT&E Hardware Estimate*	Low	50/50	High
GROUND-ORBITAL	($20,759 Total)			
CORE	3896.59	3221.73	4027.92	4846.45
CREW MODULE	2579.61	2143.31	2662.98	3196.46
TLI TANKS	390.08	323.28	402.95	484.23
TANKER	921.32	756.91	952.23	1148.88
SPACED BASED	($24,594 Total)			
AEROBRAKE	1987.65	1655.08	2051.49	2459.81
CREW MODULE	4303.70	3558.59	4514.20	5366.02
TLI TANKS	390.08	323.28	402.95	484.23
LUNAR DESCENT TANKS	680.19	564.89	702.86	843.93

NOTE:

* PROGRAM ESTIMATE EXCLUDES SCHEDULE PENALTY
* PARAMETRIC COST MODEL OUTPUT EXCLUDES ADVANCED SPACE ENGINE AND NASA PROGRAM LEVEL FACTORS (REQUIREMENTS CONTINGENCY, FEE, NASA PROGRAM SUPPORT)

Figure 3.5-7. Ranger Cost Risk Analysis by LTS Flight Element (Before Factors Application)