AIAA 92-1550

Architecture Assessment of HLLV Candidates

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

ARCHITECTURE ASSESSMENT OF HLLV CANDIDATES

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ABSTRACT

An architecture study of four Heavy Lift Launch Vehicle (HLLV) families has been performed which looked at the total spectrum of ETO needs including civil, commercial, military, and the Space Exploration Initiative (SEI). The Synthesis Group's "Mars Exploration" architecture was used as the SEI model baseline (Cryogenic Lunar vehicle, Nuclear Thermal Mars vehicle). An architecture for each vehicle family was created and analyses conducted on ground processing, launch operations, on-orbit operations, mission performance, and cost. Emphasis was placed on the ability of each architecture to meet the needs of the SEI and non-SEI missions both near and far term. The results showed that a 70t ETO vehicle for lunar missions had definite cost advantages, with only small operational disadvantages, if the lunar program were small or medium in size. The commonality with non-SEI missions made the overall architecture seem appealing. As the size of the lunar program increased, the 150t Earth to Orbit (ETO) vehicle increased in appeal. For Mars, a comparison of 150t and 250t ETO vehicles showed that little operational advantage was gained by going to the 250t size. While the 250t vehicle required fewer flights, the segmentation of the Mars Transportation System (MTS) and complexity of assembly operations was almost identical. It was clear that the architecture elements were very sensitive to study assumptions, selection criteria, and mission model, and this is highlighted throughout the analysis.

Acronyms/Abbreviations

ASRM	- Advanced Solid Rocket Motor
CNDB	- Civil Needs DataBase
CRV	- Cargo Return Vehicle
DoD	- Department of Defense
ELV	- Expendable Launch Vehicle
ETO	- Earth To Orbit
HLLV	- Heavy Lift Launch Vehicle
JPO	- Joint Program Office
JSC	- Johnson Space Center
KSC	- Kennedy Space Center
LH2	- Liquid Hydrogen
LOI	- Lunar Orbit Insertion
LOX	- Liquid Oxygen
LRV	- Logistics Return Vehicle
LTS	- Lunar Transportation System
MSFC	- Marshall Space Flight Center
MTS	- Mars Transportation System
NLS	- National Launch System
PLS	- Personnel Launch System
R/D	- Rendezvous and Dock
RPSF	- Rotational Proc. & Surge Fac.
RP	- Rocket Propellant
SCRAM	- Station Crew Rescue Alt. Mod.
SEI	- Space Exploration Initiative
SSF	- Space Station Freedom
t	- metric tonnes
TEI	- Trans-Earth Injection
TLI	- Trans-Lunar Injection
VAB	- Vehicle Assembly Building
VIF	- Vertical Integration Facility

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INTRODUCTION

As the country looks toward the future and attempts to plot its space exploration course, it becomes evident that there are many elements to be considered. The plans to explore the solar system must be evaluated alongside terrestrial exploration needs. The civil, military, science, Space Station, Shuttle, and commercial sectors all have requirements and demands on a space launch future. The goal of this analysis[1] was to look at four architectures based on different Heavy Lift Launch Vehicle concepts and evaluate these architectures based on their SEI performance in particular and overall space capabilities in general.

Figure 1 shows the total U.S. space program mass to orbit requirements through the year 2020. The commercial level is based on projecting current requirements. The DoD mass requirements are based on the NLS JPO[2] published manifest and will change as world events and security needs dictate. The next category is the NASA Civil Needs Database (CNDB) requirements. The modified CNDB '90 is the 1990 CNDB updated to reflect the redesigned Space Station and other inputs received for inclusion in the CNDB 1991. This analysis was done before the release of the CNDB '91 so an update to the previous model was used. From Figure 1, it is apparent the Lunar and Mars mass peaks dominate the landscape, although they only account for about 10% of the ETO flights. This high mass-low flight rate character would indicate that the space infrastructure designed for the future should accommodate SEI, but not be solely designed around it.

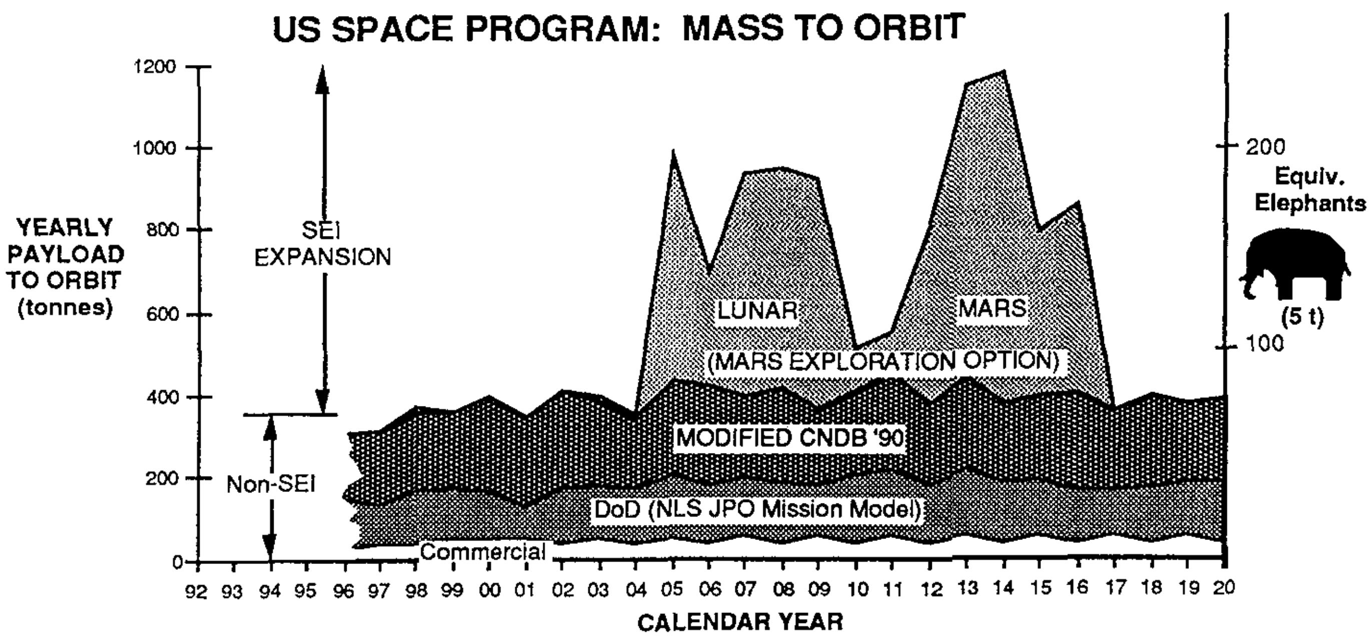


Figure 1. Total U.S. Space Program mass to orbit.

Thus, the non-SEI missions (Figure 1) must be the driver in planning for our space launch future. A pictorial showing the interaction of these different elements is shown in Figure 2. The Space Station, non-SSF missions, and SEI requirements would be met by a combination of the Shuttle, an NLS type cargo delivery system, and a set of larger vehicles to handle the SEI. Operationally, a launch system designed for each function would be ideal, but the cost would be prohibitive. There is an obvious need to minimize the number of new vehicle starts and find solutions with significant commonality. Future crew and cargo requirements have indicated that a Personnel Launch System (PLS) will be required to supplement and eventually replace the Shuttle for crew delivery; plus a cargo return vehicle (CRV) will be required to provide an up and down cargo capability. The decision on a mission model significantly affects the solution of what future systems will be required. The growth of Station, the amount of non-SSF science, the timing and scope of SEI all drive a particular solution. The next section will address the SEI mission requirements used to drive the ETO options for the remainder of this study.

MISSION REQUIREMENTS

The diversity of users and their requirements must be accounted for in defining a workable space launch solution. The SEI requirements, however, represent the most significant additions to what is currently needed for civil, commercial and military payloads. Figure 3 illustrates the SEI flight rates for the Stafford Synthesis Report[3] Case 1 "Mars Exploration" lunar and Mars scenario as well as the flight rates based on a fixed mass delivered to the lunar surface. The different ETO options (A-D) will be discussed later. The mission schedule, crew sizes, and payloads used in the analysis were obtained from the NASA JSC Lunar/Mars Exploration Program Office's "Mars Exploration Design Reference Mission"[4].

Our trade study examined a number of sensitivities to the reference mission, including the previously mentioned fixed mass to the lunar surface as well as fixed LTS flight rate with variable surface mass delivered. Case 1 employed the reference mission and payloads. Case 2 used a fixed payload mass (114t) and variable mission schedule with a lunar mission scenario emphasizing reduced on-orbit operations (a single ETO launch for the 150t ETO options, and two launches for the 70t ETO option). The effect of doubling surface payload and/or doubling number of missions was also evaluated. The Mars payloads and missions were held constant for all of the cases analyzed.

Each case and its resultant lunar and Mars transfer vehicles were analyzed separately over a number of ETO vehicle family options as shown in Figure 4. The ETO options used in the trade were: Option A, NLS + NLS Derived 150 metric ton (t) vehicle; Option B, NLS + NLS Evolved 70t and 150t vehicles; Option C, NLS + NLS Derived Core w/ New LOX/RP Boosters for 150t and 250t vehicles; and Option D, Clean Sheet New LOX/LH2 Core w/ New LOX/RP Boosters for 150t and 250t vehicles, which was evolved down for NLS payload classes (23t & HLLV). Two approaches were used to evolve the families evaluated. For the Options A, B, and C families, the low end vehicles (NLS) were evolved upward to accommodate the lunar and Mars requirements. The Option D vehicle was a clean sheet approach designed to meet the SEI requirements, then evolved downward to handle the lower mass end of the mission model.

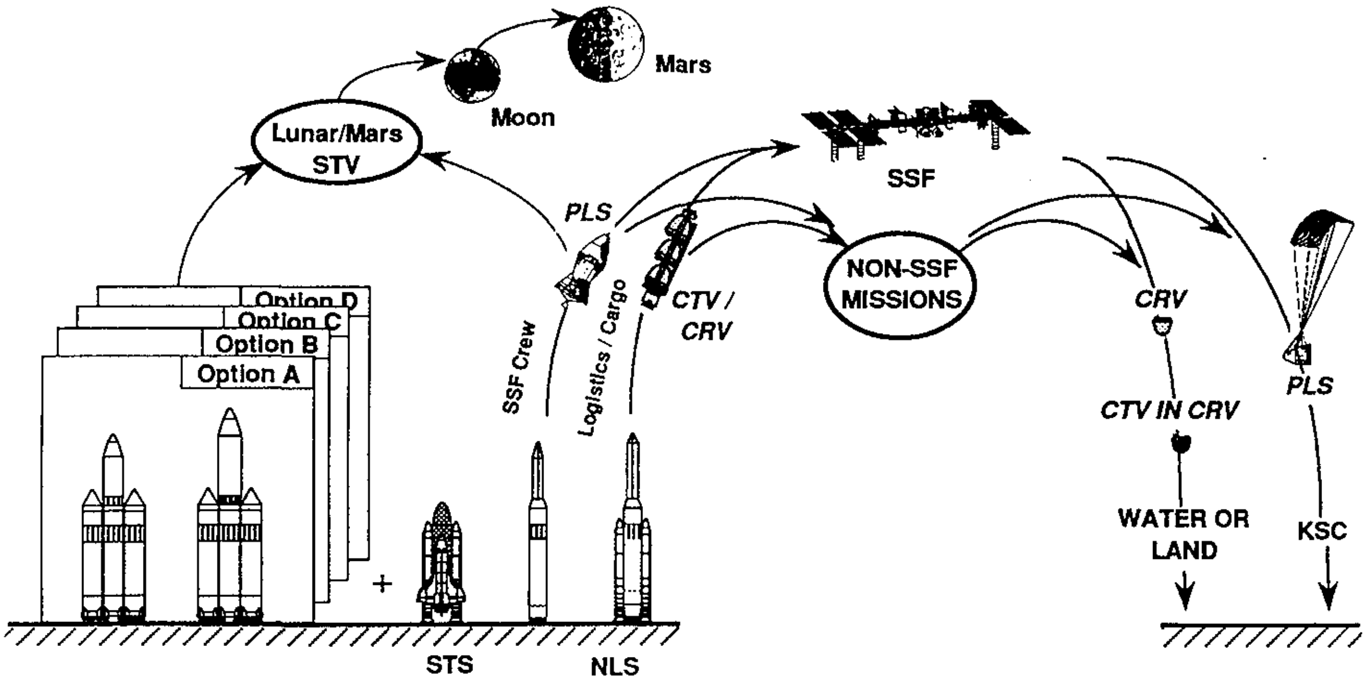


Figure 2. The diversity of space transportation missions will require vehicle families to meet the needs.

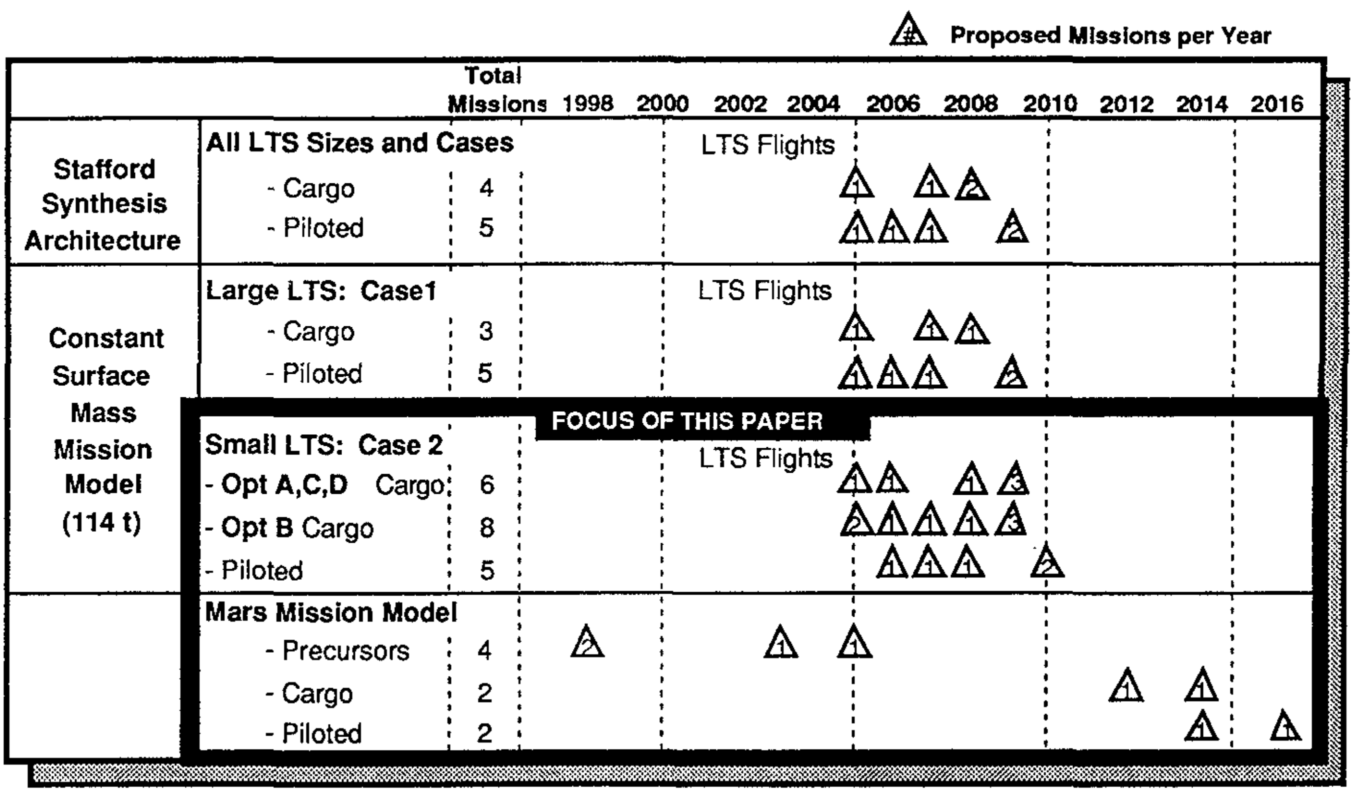
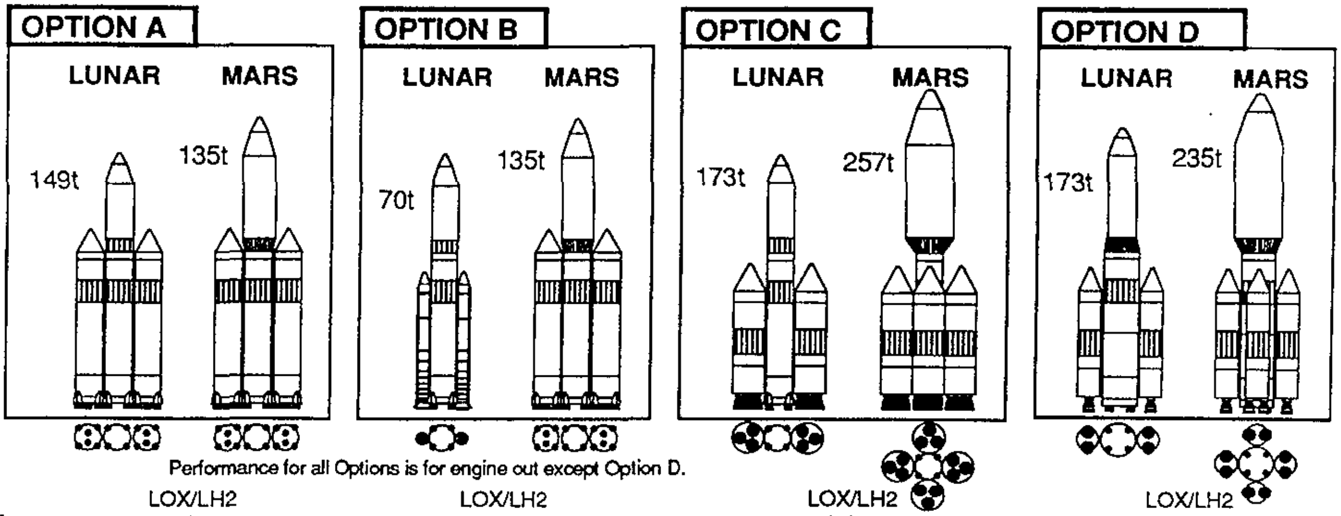


Figure 3. Stafford Mars Exploration Theme missions and SEI mission model used in this paper.



	Option A	Option B		Option C	Option D
Core	LOX/LH2	LOX/LH2		LOX/LH2	LOX/LH2
Core Dia.	8.4 m	8.4 m		8.4 m	10.5 m
Booster	LOX/LH2	ASRM	LOX/LH2	LOX/RP	LOX/RP
Bstr Dia	8.4 m	3.8 m	8.4 m	10.1 m	6.7 m
Core Material	Al-Li	Al or Al-Li	Al-Li	Al-Li	Al-Li

Lunar Shroud 7.6 x 20 m Mars Shroud 14 x 30 m and 10 x 30 m

Figure 4. Four ETO vehicle family options were analyzed.

The Lunar transfer vehicles used in the study are shown in Figure 5. The lunar transportation systems (LTSs) were designed in three categories: direct (ground integrated LTS on one launch); rendezvous and dock (two LTS segments delivered on two launches); and assembly (multiple LTS segments from multiple launches assembled on-orbit). Figure 6 illustrates the direct and rendezvous and dock options. The type of LTS associated with Mission Case 1 or 2 and a specific ETO Option are shown in Figure 5. All LTSs were cryogenic expendable with Apollo style crew return to Earth. The Mars transportation systems (MTSs) are shown in Figure 6 and were nuclear thermal with assembly required on-orbit. The Mars missions were the same for each of the cases and only two ETO sizes (150t and 250t) were evaluated. The MTSs had the same performance capability and their designs were optimized for the ETO vehicle that would carry them. A free-flying node was used for assembly of the MTSs.

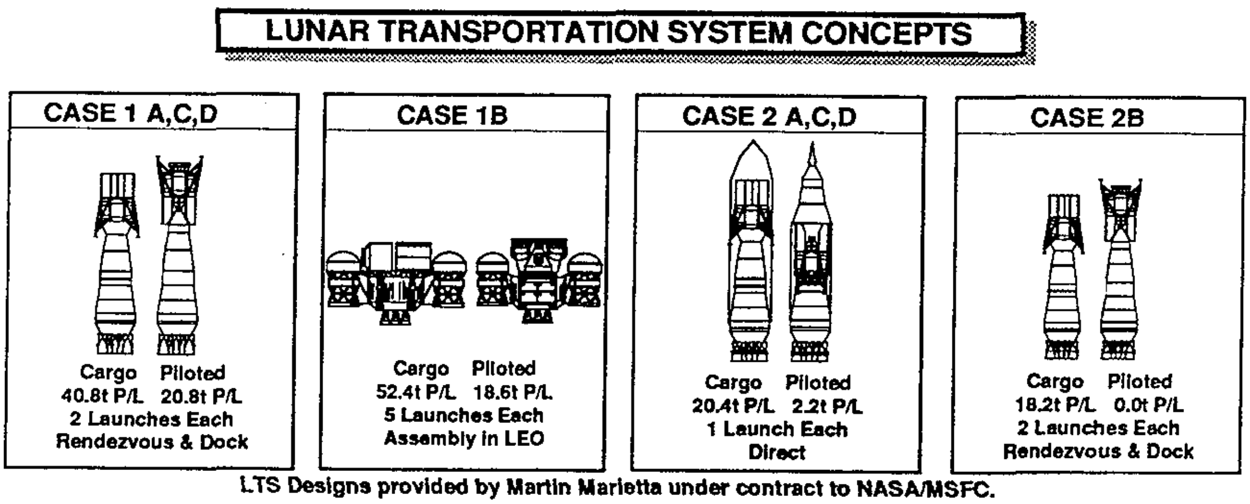


Figure 5. Lunar Transportation Systems used in the analysis for Case 1 and 2 Mission models.

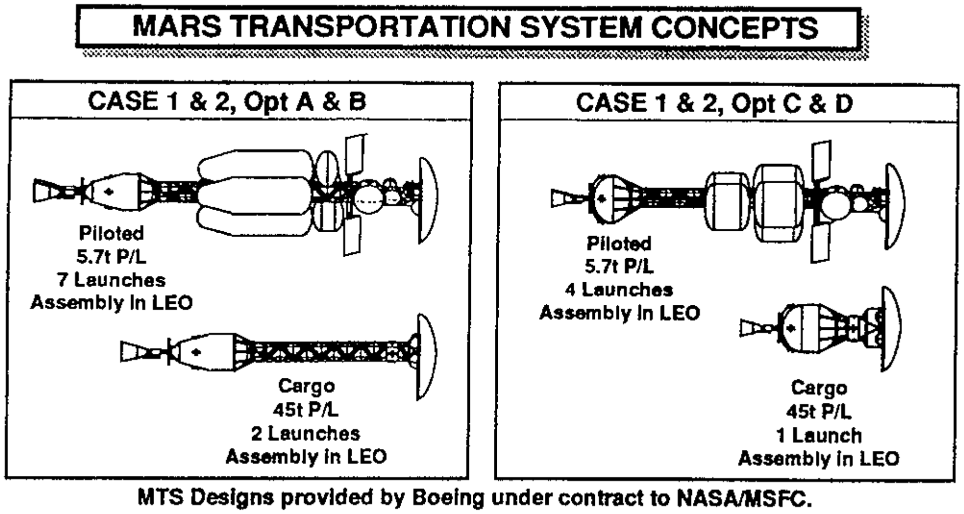


Figure 6. Mars Transportation Systems designed for the 150t and 250t ETO vehicles.

The study analyzed Cases 1 and 2 for ETO Options A, B, C, and D. Figure 7 is a mission model layout of the the Option A architecture for Cases 1 and 2 which is representative of the architectures which were created for each of the Options. The chart includes non-SEI missions (both civil and DoD) although ELV flights on existing vehicles were excluded for clarity since these were common across each of the Options. Although both Cases 1 and 2 were assessed, the remainder of this paper will focus on the Case 2 mission model for brevity (fixed mass to the lunar surface (114t) with variable LTS flight rate).

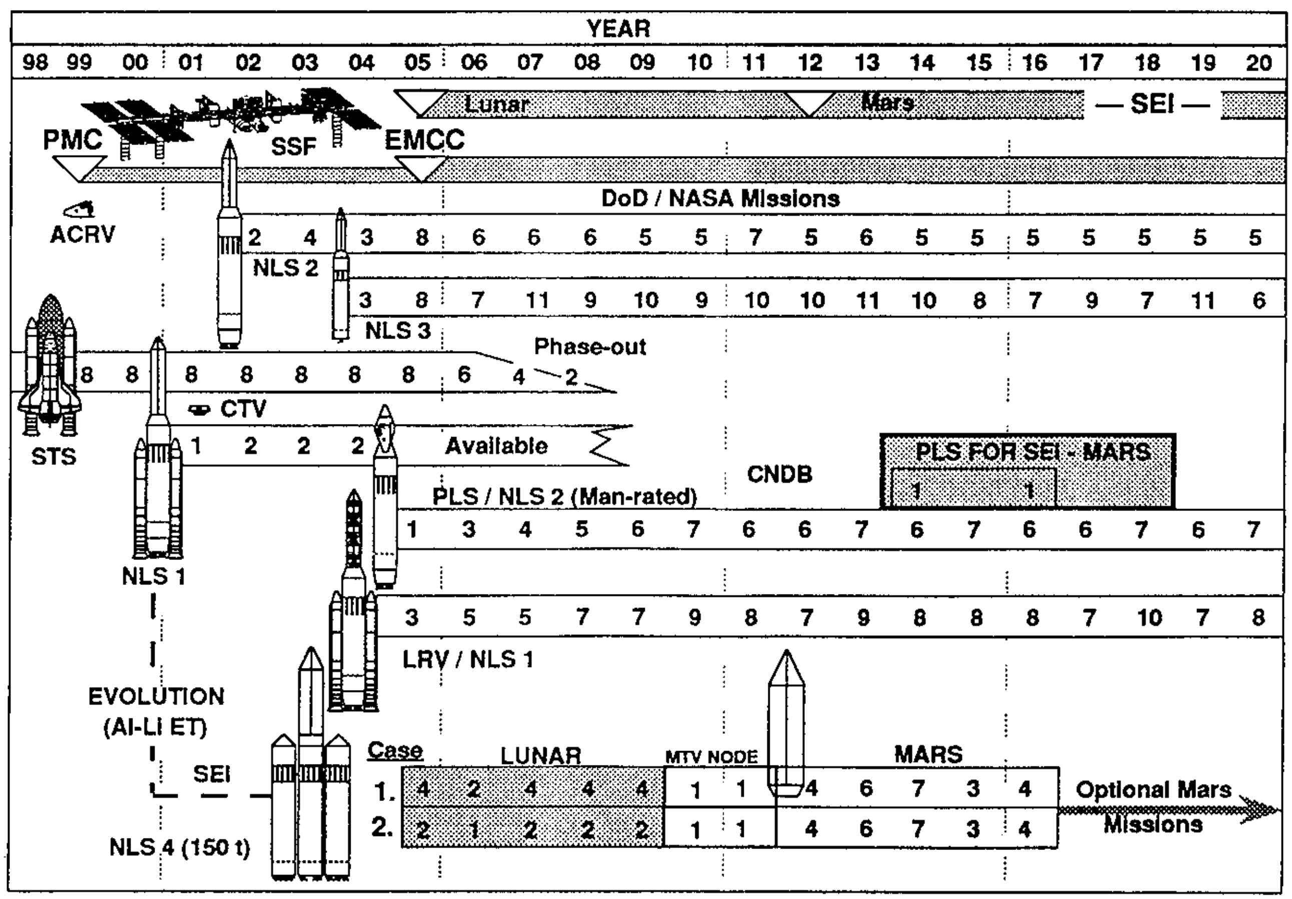


Figure 7. Representative Mission model layout for Option A.

FACILITIES AND GROUND OPERATIONS

The ground facilities required for each of the architectures and the ground operations to support them over the whole mission model were investigated. The architectures had some common facilities requirements to support the SEI payloads and missions. Additionally, each had equipment and facilities peculiar to its configuration. Option B required an expanded ASRM stacking and processing capability since the lunar missions are performed with the 70t (2 ASRM) NLS vehicle. ETO Options C and D required a new Vertical Integration Facility (VIF) and launch complex to accommodate the 250t vehicle. Option D also required production facilities to manufacture its larger vehicle cores (10.5m).

The analysis looked at the processing of the SEI vehicles along with the nominal vehicle ground processing flows and timelines. The facility usage requirements were developed so that bottlenecks and shortfalls could be discovered. Three areas of particular interest were launch complex 39A/B, the Vertical Assembly Building high bay usage, and the ASRM processing capabilities. The launch complex 39 A/B capabilities were sufficient for each of the Options for Case 2, however, the Option B Case 1 scenario was over by 15%. The Vertical Assembly Building (VAB) utilization required using the high bay 2 (currently used as Orbiter safe haven) into a bay to handle the 150t vehicles. As such, the flow through the VAB could be accommodated for each of the Options. The ability to handle the ASRMs was also a concern, especially for Option B . The Rotational Processing and Surge Facility (RPSF) can handle 12 ASRM shipsets per year. The flight rate for Option B exceeds this even for Case 2. As shown in Figure 8, the other Options

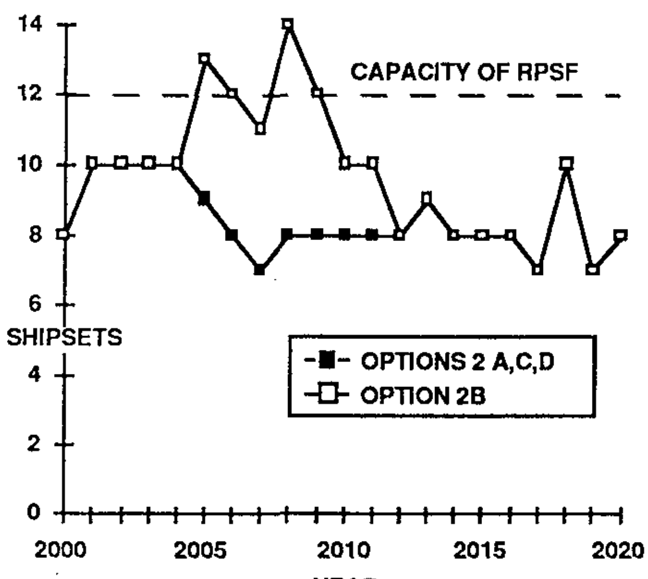


Figure 8. ETO option impact on ASRM processing capability.

do not present a concern since only Option B uses the ASRMs for lunar SEI. The additional facilities required to handle each of the architectures' requirements were included in the architecture as well as the costing.

A related concern is that of environmental and siting issues. The real estate around KSC is very environmentally sensitive making the addition of facilities and launch pads quite complicated. Additionally, the blast radius for the 250t vehicle could infringe upon industrial areas restricting launch flexibility. The exhaust products of the vehicles are also a concern. From an SEI only perspective, Option A (all LOX/LH2) has the cleanest exhaust, Options C and D next (LOX/LH2 and LOX/RP-1) and Option B (LOX/LH2 with ASRMs) the least favorable. Since all of these options use a similar set of vehicles for the non-SEI missions (>90% of launches), the exhaust consideration differences would only be a concern for a large SEI program.

From an overall launch processing viewpoint, Option A is the preferred architecture because it has the smallest additional facilities requirements, the cleanest exhaust, and has no pad siting issues. Option C would be next, followed by D, and then B. For Case 2, where a modest lunar program is being pursued, a similar order of preference can be made. However, none of the options has ground processing or support requirements that would keep it from being a viable alternative.

SPACE OPERATIONS

Space operations is a key area for consideration. Little historical experience combined with the inherent complexity of performing operations at a remote location makes minimizing the amount of operations required for an option attractive. Experience with the Space Station and other programs could increase our confidence and reduce the anxiety of assembling or refurbishing elements in space. For this study, on-orbit operations for each LTS/ETO and MTS/ETO match was examined to determine the on-orbit requirements. Topics such as complexity of operations, number of ETO flights per mission, Space Station interference, debris protection, supportability and node requirements were examined. The resulting findings for each case were then compared to determine relative merits of one option over another.

The LTSs, as shown in Figure 5, had three different design configurations. Only the Direct and the Rendezvous/Dock (R/D) Options were evaluated for the Case 2 missions which had as groundrules minimum space operations and a sub-orbital burn of the TLI stage. The TLI stage performing a suborbital burn on its delivery increases the initial mass in LEO of the overall LTS and correspondingly, increases the mass to the lunar surface. As shown in Figure 9, the Direct

LTS would be launched as a fully integrated LTS and only minimal on-orbit checkout would be required. The design of the Direct LTS, however, would require that the piloted mission perform a docking operation on-orbit. This is because the crew module must have a method of escape, which requires an on-orbit docking with the LEV similar to the sequence used for Apollo missions. This approach also precludes performing a sub-orbital TLI burn. The Rendezvous/Dock approach requires more operations and coordination. The R/D cargo vehicle would be delivered in two fully integrated sections. The first flight would carry up the TLI stage and the second would carry the LOI stage and cargo lander. These two sections would rendezvous and dock in LEO and an integrated vehicle checkout would then be done.

The first ETO for the piloted version of the R/D LTS would also be the TLI stage (with similar sub-orbital burn performance benefits). The second flight would carry the lander (LEV) and LOI stage but not in an integrated configuration. Once on-orbit, the lander would rendezvous and dock with the crew transfer module/LOI stage. This segment would then rendezvous and dock with the TLI stage, undergo systems checks, and perform the TLI burn. Operationally, the Direct is less complex than the Rendezvous/Dock, but either option would be acceptable.

The Man-Tended Free Flyer shown in Figure 9 would be used for assembly of the Mars vehicles. This facility would use elements common to the Space Station and be delivered by two of the 150t ETO vehicles.

Separate Mars Transportation Systems (MTS) were designed for the 150t and 250t ETO vehicles. The 150t version was manifested on two ETO flights for the cargo mission and seven ETO flights for the piloted mission as shown in Figure 10. Both cargo and piloted missions require one of the ETO flights to employ a 14m x 30m shroud. There is some question as to the technical feasibility of this larger diameter shroud on a 8.4m core ETO vehicle as would be the case for ETO Options A, B, and C.

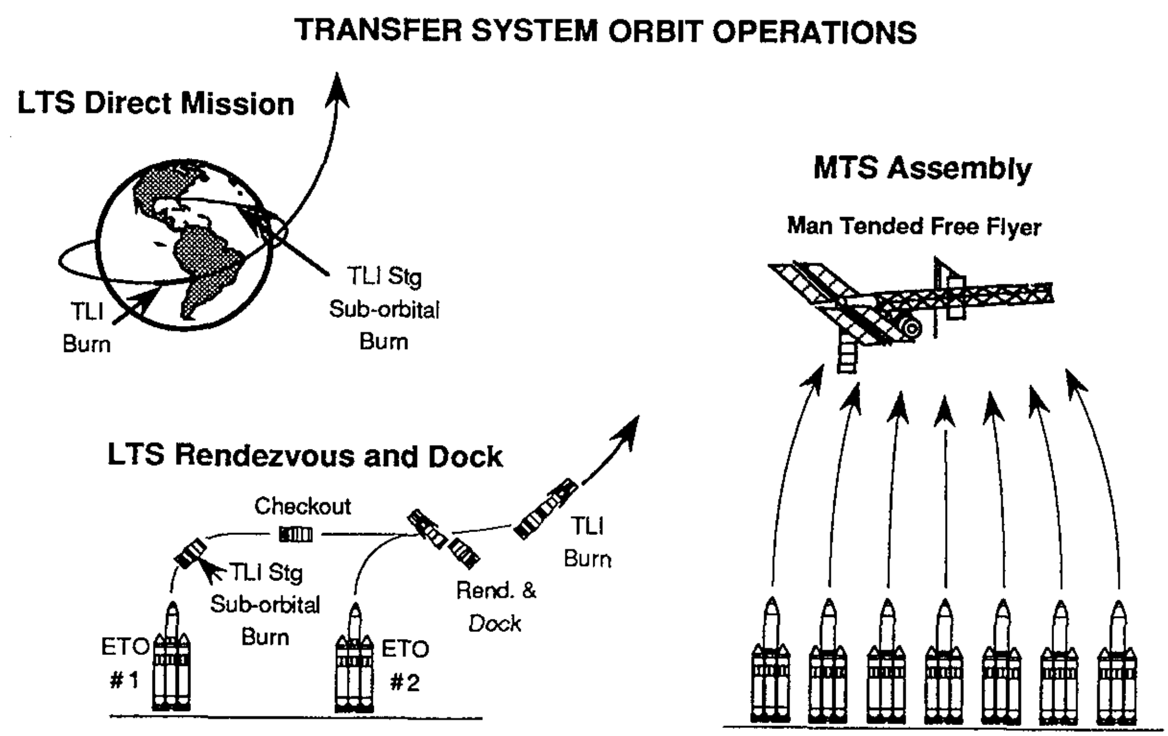


Figure 9. On-Orbit Operations: Lunar Direct and Rendezvous/Dock options and Mars MTFF.

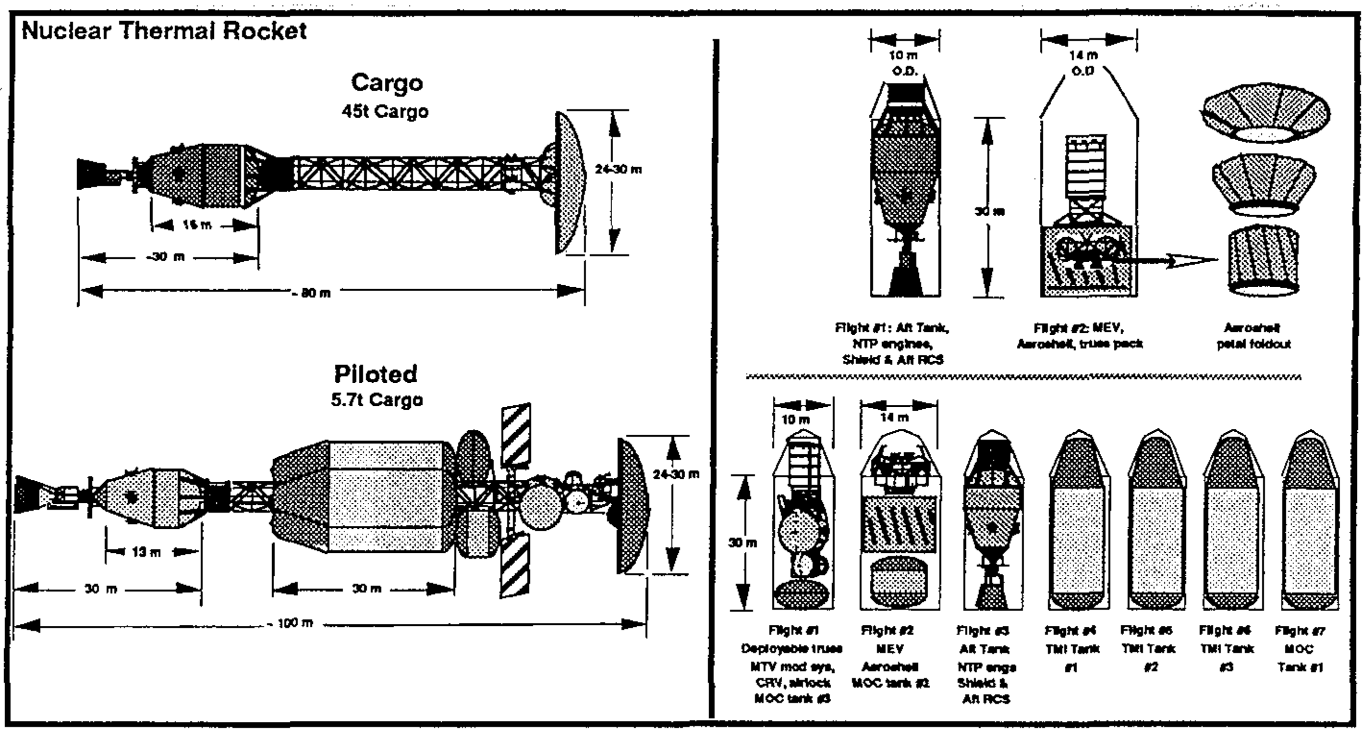


Figure 10. 150t ETO delivery of MTS vehicle.

The 150t ETO MTS cargo mission requires some on-orbit processing to integrate the vehicle. The first delivery flight requires no processing while the second flight requires that the truss and aeroshell be deployed and the aft tank structure mated to the forward truss structure. Once fully configured, the debris shielding would be removed and a complete vehicle checkout performed prior to departure. This could be accomplished at either a 296 or 408 km orbit since processing at the Man Tended Free Flyer (MTFF) node would optional.

The piloted mission requires major assembly at the MTFF node. There are six propellant tanks to be mated to the vehicle as well as mating the lander, aeroshell and aft tank/engine structure. With seven ETO flights required, the driving factor in determining LEO stay time would be the ETO launch capability. Assuming 40 day centers, the MTS would be a minimum of 280 days at the MTFF. Also, any problems encountered during assembly could cause schedule delays resulting in reduced mission performance or even the missing of a Mars opportunity. This could require resizing of propellant tanks to account for contingencies.

The MTS design for the 250t launch vehicle (shown in Figure 11) has some, but limited, advantages over the 150t ETO MTS. The cargo version of this design is configured for a single ETO flight. Even so, there is more complex on-orbit assembly required than for the 150t MTS design. As shown in the figure, the aeroshell is not attached to the vehicle. Thus, the aeroshell could be deployed anywhere, but the attachment would have to be manually performed. The cargo vehicle would have to be delivered to the MTFF rather than conduct the less complex rendezvous and dock operation in LEO. One ETO flight is saved with this configuration but increased on-orbit operations are required. The compact design of the cargo vehicle raises concerns about the radiation from the nuclear thermal engines. An immediate observation is that the nuclear reactor and avionics/MEV are very close together. The hydrogen in the tank is a good shield for the avionics and MEV during the early stages of flight. As the hydrogen is consumed, the protection will decrease. The longer the engines operate, the more radiation they emit, which means when the radiation is at its worst the shielding is at its lowest. The only alternative is to increase the mass of the shadow shield to reduce the radiation levels. This increase in mass could be substantial, and since it is carried for the entire mission, could increase the mass of the vehicle beyond the 250t vehicle capability. If the avionics and MEV must be moved farther away from the reactor, the additional structure volume would require either a longer shroud or two launches. The size of the shroud is already an issue for the cargo vehicle delivery. The standard shroud is 30 m long, whereas the unitary MTS vehicle shroud would have to be at least 40 m, decreasing further the performance margin of the 250t vehicle.

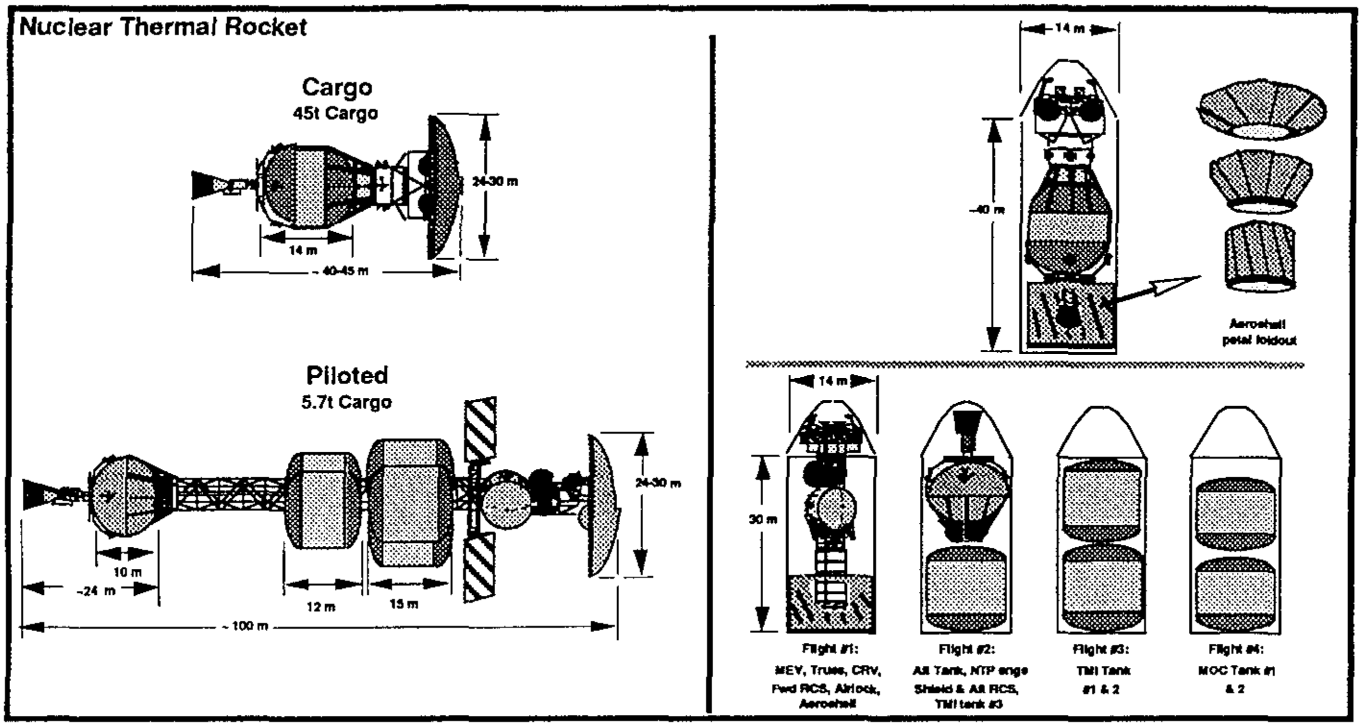


Figure 11. 250t ETO delivery of MTS vehicle

The piloted MTS saves three ETO flights over the 150t configuration (four ETO flights rather than seven). The actual on-orbit operations, however, are only minimally impacted. The 250t ETO MTS configuration requires five propellant tank installations instead of six for the 150t ETO MTS. Also, the lander is already attached to the core for a small savings in the operations necessary to attach it on-orbit. All other operations performed for the 150t ETO piloted MTS will also need to be performed for the 250t ETO piloted MTS.

An issue that is unresolved is the performance and dynamics impact of the 14m payload envelope shroud on the 8.4m ET derived core (shroud-core ratio of 1.75). The shroud is currently required for all Mars missions whether on the 150t or the 250t vehicles. Only option D has a large diameter core (10.5m) and is thus unaffected by a 14 m shroud size (shroud-core ratio of 1.39). For the 150t vehicle, if the shroud-core ratio is an issue, the Option A, B and C vehicles would be constrained to a 10 m shroud. The impact of this would be a redesign of the MEV and the aerobrake, probably resulting in one additional flight for each of the piloted and cargo missions (7 total over model). The addition of 7 extra flights would need to be weighed against the impacts of the larger shroud, and impact of the core diameter on other missions such as the CNDB and DoD. For the 250t vehicle, only the C and D options have vehicles in this class. If the C family was constrained to the 10 m shroud because of concerns, their would be no benefit to a 250t vehicle since the MTS is more volume constrained than mass constrained. Therefore, the only option would be D with the 10.5m core. The conclusion is that if the shroud-core ratio is limiting, the need for a 14 m shroud would require the Option D ETO family and if the 10 m shroud were chosen, additional flights would be required due to the volume limitations.

Overall, there are some savings in time on-orbit for the 250t ETO due to the lesser number of flights (4 vs 7 and 1 vs 2) required for hardware delivery. The launch centers alone will save about 120 days for the piloted case and 40 days for the cargo case. The on-orbit operations for the 250t ETO MTS vehicles, however, are similar in duration and equal to or greater in complexity to the 150t ETO versions with only two assembly operations saved.

COST ANALYSIS

The total transportation investment costs for Case 2 scenario, ETO Options A and B, are shown in Figure 12. Analysis on all four options was performed but only A and B are shown for comparison. Total transportation cost turned out to not be a major discriminator among the various options with the total investment estimates for all options coming within 5% of each other. Within the individual categories, the only significant differences occurred in MTS DDT&E and non-SEI facilities. The 250t ETO MTS (Options C and D) had a DDT&E 12% lower than that of Options A and B. Option D incorporates a new ETO production facility thus increasing its ETO Facilities estimate over that of the other options.

Option B's LTS is slightly cheaper than the other options. This, along with Option B's lower SEI ETO DDT&E funding in the lunar phase, combine to create an 11% lunar peak-year funding advantage for this option compared with Option A. If compared against Option C the advantage increases to 19%. Options C and D have slight (8% of Option A) Mars peak-year advantages over Options A and B.

If we choose to examine the lunar phase in isolation from Mars (since it is probably too early to identify and quantify the Mars elements) then important cost and programmatic differences are revealed. Figure 13 shows the lunar only comparison for Options A and B.

TOTAL TRANSPORTATION INVESTMENT COST (CASE 2)

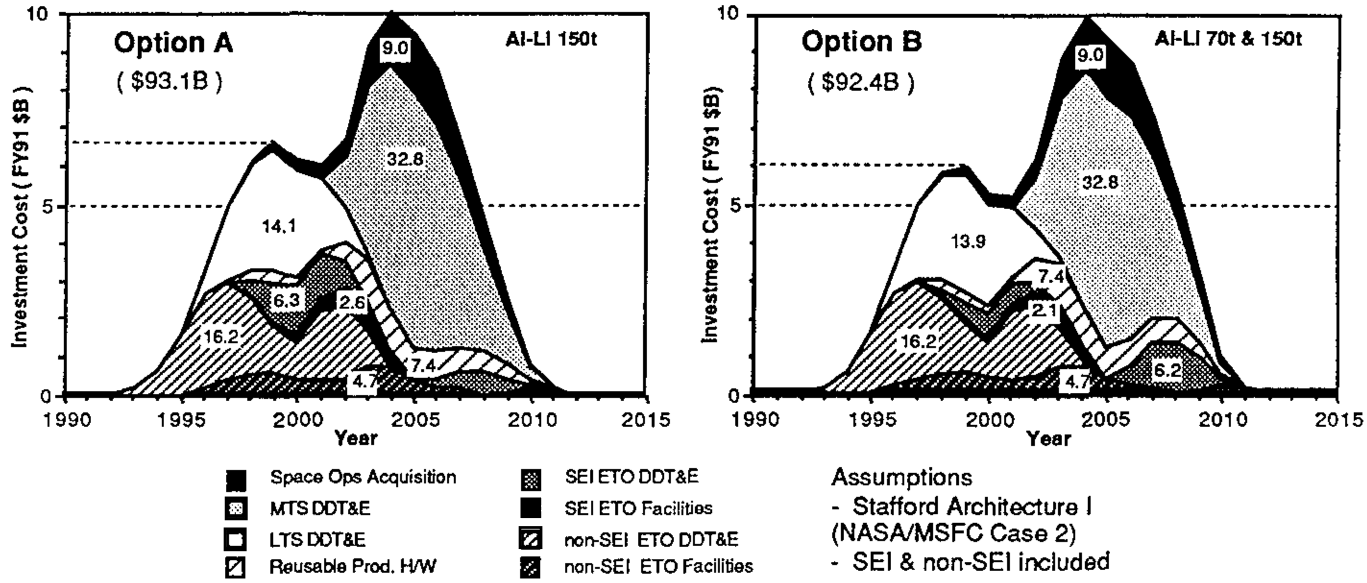
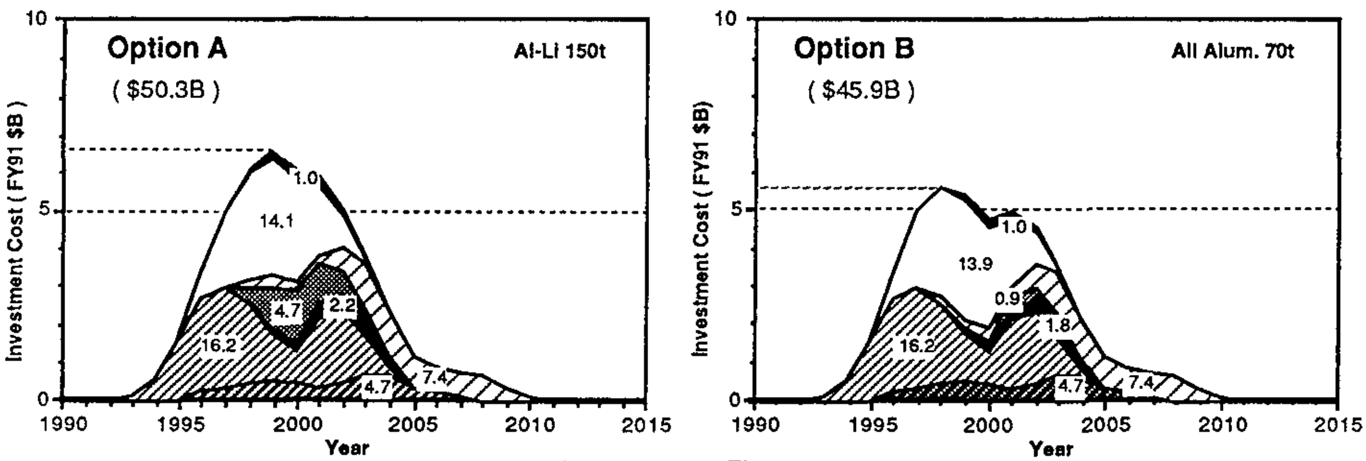


Figure 12. Total Mars and Lunar Investment Cost (A&B only) for Case 2.

TOTAL TRANSPORTATION INVESTMENT COST THROUGH LUNAR (CASE 2)



KEY: Same as Figure 14.

$6.9 B	**LUNAR ETO**	**$2.2 B**
$22.0 B	**TOTAL LUNAR INVESTMENT**	**$17.6 B**
$45.6 B	**LIFE CYCLE COST**	**$40.9 B**

Figure 13. Total Lunar Investment Cost comparison for Case 2.

The upfront investment cost is a significant parameter due to the constant pressure of budget concerns. Also, the lunar 70t vehicle is very similar to the vehicles used to fly the non-SEI missions and as such has a per vehicle cost of about 33% of a 150t vehicle. Thus, while it may have to fly twice as many times, its equivalent cost is less. This translates into a lunar life cycle cost advantage, but including Mars, life cycle costs were not sufficiently different to be a discriminator. For a small lunar program, Option B will have definite cost advantages.

OBSERVATIONS

After the ETO options were assessed from an SEI perspective, a look at the performance of each of the HLLV families in flying the CNDB and low end payloads was made. The vehicles evaluated were the ETO Option family vehicles which were replacement options for the Titan IV and Shuttle. Options A, B, & C all had a common ET derived core, so the current definition NLS 50k and NLS HLLV were the low end family members for these. Option D had a new clean sheet 10.5m core which evolved into the low end family members. The only real difference which emerged between the vehicles is in their delivery capability. The larger core for Option D gives each of its vehicles a 40-60% greater mass to LEO. For the Option D 1.5 stage vehicle, the extra performance (9t) is not required to support the current mission model. The Option D vehicle with 2 ASRMs would have a 27t greater payload capability. This excess could allow a reduction in flights required to replace the Shuttle if flying cargo return vehicles with payloads. It could also allow greater flexibility for STS upgrade, offload or replacement since it would have the ability to fly an engine-less Orbiter, larger payloads, or new concepts of crew and cargo. Unless, however, use of this extra performance were a part of the future space launch plan, the extra cost per flight could not be justified. Bottomline, no significant differences between the options exist with regards to the current mission model payloads, although Option D allows for future growth.

CASE 2 ETO TRADE STUDY EVALUATION
Lunar Only - (114t Model)

CRITERIA		ETO OPTION B (70 t all Alum.)	A (150 t Al-Li)	C (150 t Al-Li)	D (150 t Al-Li)
Lunar ETO Flights		26	11		
Cap. to Deliver Crew Habitat?		2 Flights	Habitat Redesign		No Engine Out
Trans. MTS crew Mod to LLO?		No	No		
Reliability: ETO Engines Used		52 ASRMs 104 STMEs	176 STMEs	66 F-1As 44 STMEs	44 F-1As 44 STMEs
Environ-mental Impact	Relative QD Impact	1 (Reference)	1.35		
	Overall Impact	3	1	2	2
Space Operations	On-Orbit Prep.	Rend./Dock	Direct/Dock		
	TLI Stg Sub-Orbital Burn Possible?	Piloted & Cargo	Cargo Only		
SEI Cost (FY91 $B)	ETO Avg. Flt Cost $M	164	522	Not Avail.	Not Avail.
	ETO Investment	2.7	6.9	8.9	6.8
	Total Investment	17.6	22.0	24.0	21.9
	Life-Cycle Cost	40.9	45.6	Not Avail.	Not Avail.

KEY: LESS DESIRABLE

Figure 14. Comparison of Case 2 Lunar ETO Options.

The analysis of the lunar Options considered two ETO vehicles, the 70t (Option B) and the 150t (Options A,C, and D). A summary of some of the results are shown in Figure 14. The criteria listed along the left side of the figure are factors which allow quantitative discrimination between the options. As has been shown from the previous discussions, the size of the mission model makes a big difference on how significant each of these differences are. From an operational standpoint, Option B is the most cumbersome and complex. From an upfront and life cycle cost perspective, it has clear advantages. The results suggest that for a small or medium manifest, the operational difficulties are acceptable and the cost advantages great enough to make Option B very attractive. As the mission model grows, however, the operational complexities of Option B would no longer be desirable, making the 150t a more prudent choice.

The analysis of the Mars Options also fell into two ETO categories, the 150t (Options A and B) and the 250t (Options C and D). Due to the mass and size of the MTS vehicles, a 70t delivery vehicle was not considered viable. Figure 15 summarizes some of the results using similar criteria to those used for the lunar assessment. The mission analysis revealed that a 150t booster requires over 70% more flights and 25% more payload elements, but similar on-orbit assembly operations. As a result, the 150t booster options have substantially higher recurring and life-cycle costs. Options C and D have similar life-cycle costs, with option C having lower nonrecurring and option D lower recurring ETO costs.

Two major issues emerged from the trade study as discussed earlier. The fairing-to-core ratio and radiation hazard issues must be resolved before any ETO option can be recommended for the Mars mission. The fairing-to-core ratio for the A, B, and C booster Families are greater than the maximum ratio flown to date (1.67 on Titan) and may cause substantial aerodynamic and performance problems. The second issue is the unitary launch of the MTS cargo vehicle on a 250t booster which requires that the avionics be packaged in proximity to the nuclear reactor. This would require a significant increase in shielding and vehicle mass which could exceed the 250t capability.

ETO TRADE STUDY EVALUATION
Delta Mars Only - Case 2

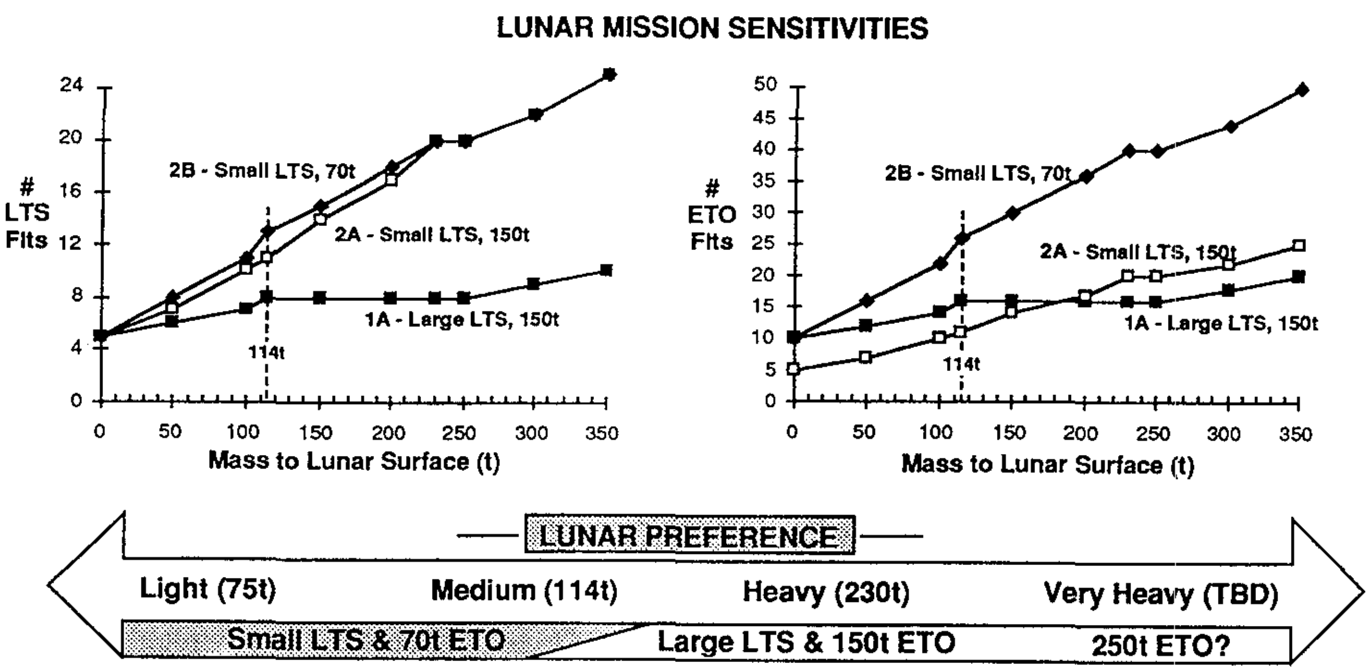
CRITERIA		ETO OPTION A (150 t)	B (150 t)	C (250 t)	D (250 t)
Mars Ground Operations	ETO Flights	24		13	
	Elements Proc.	72		65	
Intensity (1st Year ETO Flt Rate)		4		2	
Reliability	Total P/L Elements	35		26	
	ETO Engines Used	384 STMEs	384 STMEs	156 F-1As 52 STMEs	104 F-1As 52 STMEs (No Engine Out)
Environmental Impact	Relative QD Impact	1 (Reference)		1.3	
	Overall Impact	1	1	2	
Space Operations	On-Orbit Oper.	Assembly		Assembly	
	MTTF Node Req'ts	2 Assembly Flights		2 Assembly Flights	
ΔMars Total Up-Front Invest.		42.9	45.2	37.7	39.1
Fairing / Core Ratio		1.75		1.75	1.39
Radiation Hazard (Cargo Only)		Lower		High	

KEY: LESS DESIRABLE

Figure 15. Comparison of Mars only ETO Options.

If both of these issues can be resolved, option C may emerge as the recommended approach for Mars. However, if the fairing-to-core ratio issue cannot be satisfactorily resolved, option D prevails. If the radiation hazard issue cannot be resolved and a unitary mission is not feasible, the flight rates and recurring costs of options C and D would increase dramatically and option A or B would be more favorable. If neither issue can be resolved, a new ETO approach must be developed, perhaps a larger diameter core for options A, B, or C; or use of the 150t and 250t vehicles with options C and D.

As was mentioned earlier, the solution is very sensitive to the mission model flown. Figure 16 shows the sensitivity of the LTS/ETO options to mission scenario based on number of LTS and ETO flights required. Since Options A, C, and D are very similar for the Lunar case, they are combined in the figure. For comparison, the Case 1 Option A,C, and D rates are also shown. For the lower mass requirements, the options are relatively close in the number of LTS flights required. As can be seen by the flat curve, the large LTS of Case 1 is actually very inefficient at the low mission scenarios. Similarly, the ETO flight rate versus mass graph shows that up to about 180t, the Case 2 small LTS options actually have a lower flight rate than the Case 1 large LTS options (mostly due to timing requirements on cargo and piloted flights and individual element masses). As the total surface mass increases and the number of LTS and ETO flights increase, the small ETO and small LTS options suffer from increasing concern over reliability and flight rate. These graphs assume crew delivery is constant with five missions while they would likely increase as mass to surface increases.



MARS PREFERENCE

ISSUE	PREFERENCE
• FAIRING TO CORE RATIO < 1.7 reqd ≥ 1.7 okay	Opt D All Opts Equal
• NUCLEAR RADIATION Unitary Cargo okay 2 cargo flights reqd	Opt C or D All Opts Equal
• GENERAL EVAL CRITERIA	TBD

Figure 16. Lunar mission sensitivity and Lunar / Mars ETO preferences.

For a small program, upfront cost is probably the most important factor since the number of missions is low and any operational complexity could be handled for a few missions. As the mission model grows, however, the initial costs begins to be overcome by recurring costs and operational capability. Therefore, the small ETO and small LTS (2B) are well suited for a light or medium lunar program. As the model increases, the operational complexity and mission performance considerations drive a larger LTS and ETO (eg. 1A, C, or D).

While ideally one HLLV family concept would emerge from a study like this a clear winner, regardless of scenario, the preceding work shows this is not necessarily the case. For a small lunar program, a 70t HLLV would work quite well. If the program were to grow, a larger vehicle is required. For Mars, technical issues and considerations must be assessed before a conclusion can be drawn. The 250t vehicle reduces the number of delivery flights, but due to the packaging requirements of the MTS vehicles, little operational advantages can be realized.

Thus, for the reference (Case 2 - 114t) lunar scenario, Option B emerges as our preference. For Mars, many factors cloud the selection preference. No recommendation is offered at this time. However, it is likely that a hybrid ETO family (combination of ETO options A, B, C, and D, or other) could be found to best satisfy both lunar and Mars SEI options as well as CNDB/DoD/Commercial requirements. Selection criteria, SEI/CNDB models, and assumptions drive the solution.

ACKNOWLEDGEMENTS

Much of the work presented in this paper has been performed under the Space Transportation Infrastructure Study (IS) at General Dynamics Space Systems Division for NASA Marshall Space Flight Center (Contract NAS 8-37588). The author wishes to acknowledge Bob Armstrong, Uwe Hueter, and Gene Austin of MSFC as well as Jack Duffey, Dan Eimers, Karen Kakazu, Greg Farmer, Don Deming, Todd Mosher, Sam Wagner, Dom D'Annibale, Jim Risch, Walt Boost, and Jon Barr of the GDSS Infrastructure Study team for their contributions to this work.

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