Lockheed Martin
Missiles & Space- Huntsville

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Advanced Transportation System Studies Technical Area 2 (TA-2) Heavy Lift Launch Vehicle Development Contract

NAS8-39208
DR 4

Final Report

Prepared by
Lockheed Martin Missiles & Space
for the
Launch Systems Concepts Office
of the
George C. Marshall Space Flight Center

July 1995

Advanced Transportation System Studies Technical Area 2 (TA-2) Heavy Lift Launch Vehicle Development Contract

NAS8-39208
DR 4

Final Report
Volume I
Executive Summary

Approved:

James B. McCurry
TA-2 Study Manager

Lockheed Martin
Missiles & Space- Huntsville

Preface

The Advanced Transportation System Studies (ATSS) Technical Area 2 (TA-2) Heavy Lift Launch Vehicle Development contract, NAS8-39208, was led by the Missile Systems Division of Lockheed Martin Missiles & Space (LMMS), and supported by principal TA-2 team members Lockheed Martin Space Operations (LMSO), Aerojet, ECON, Inc., and Pratt & Whitney. Addition technical task support was provided by Lockheed Martin Skunk Works (LMSW).

The ATSS TA-2 contract was managed by James B. McCurry, Lockheed Martin Missiles & Space, and performed for Mr. Gary W. Johnson, Contracting Officer's Technical Representative (COTR), of the Launch Systems Concepts Office (Organization Code PT-51), National Aeronautics and Space Administration George C. Marshall Space Flight Center (MSFC).

The purpose of the TA-2 contract was to provide advanced launch vehicle concept definition and analysis to assist NASA in the identification of future launch vehicle requirements. Contracted analysis activities included vehicle sizing and performance analysis, subsystem concept definition, propulsion subsystem definition (foreign and domestic), ground operations and facilities analysis, and life cycle cost estimation. The basic period of performance of the TA-2 contract was from May 1992 through May 1993. No-cost extensions were exercised on the contract from June 1993 through July 1995.

This document is the final report for the TA-2 contract. The final report consists of three volumes:

Volume I	Executive Summary
Volume II	Technical Results
Volume III	Program Cost Estimates

Volume I provides a summary description of the technical activities that were performed over the entire contract duration, covering three distinct launch vehicle definition activities: heavy-lift (300,000 pounds injected mass to low Earth orbit) launch vehicles for the First Lunar Outpost (FLO), medium-lift (50,000-80,000 pounds injected mass to low Earth orbit) launch vehicles, and single-stage-to-orbit (SSTO) launch vehicles (25,000 pounds injected mass to a Space Station orbit).

Per direction from the TA-2 COTR, Volume II provides documentation of selected technical results from various TA-2 analysis activities, including a detailed narrative description of the SSTO concept assessment results, a user's guide for the associated SSTO sizing tools, an SSTO turnaround assessment report, an executive summary of the ground operations assessments performed during the first year of the contract, a configuration-independent vehicle health management system requirements report, a copy of all major TA-2 contract presentations, a copy of the FLO launch vehicle final report (NASA document with contributions from TA-2), and references to Pratt & Whitney's TA-2 sponsored final reports regarding the identification of Russian (NPO Energomash) main propulsion technologies.

Volume III provides a work breakdown structure dictionary, user's guide for the parametric life cycle cost estimation tool, and final report developed by ECON, Inc., under subcontract to Lockheed Martin on TA-2 for the analysis of heavy lift launch vehicle concepts.

Any inquiries regarding the TA-2 contract or its results and products may be directed at Mr. Gary W. Johnson, NASA Marshall Space Flight Center, (205) 544-0636.

Acknowledgments

The TA-2 study manager wishes to acknowledge the outstanding working relationships that developed during the TA-2 study contract, involving a true team effort between each of the TA-2 participants. The TA-2 COTR, Mr. Gary W. Johnson, was instrumental in fostering a team-play environment between the NASA and contractor participants that resulted in an extraordinary amount of engineering analysis results being produced. The TA-2 participants were immersed in a very dynamic environment in which the scope of the launch vehicle analysis activities constantly changed, reflecting the extraordinarily dynamic events that were unfolding at NASA Headquarters during the period of March 1992 through June 1994. The following personnel, listed by participating organization, are gratefully acknowledged for their outstanding contributions to the TA-2 contract. In addition, special recognition is due Messrs. Keith Holden and Kevin Sagis for their unique and innovative contributions during the entire course of the TA-2 contract in the development of vehicle sizing tools and the assessment of vehicle performance, respectively.

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Phil Sumrall	Headquarters sponsor

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Steve Creech	Cost assessment
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Joe Hammaker	Cost assessment
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Alan Chan	Structures analysis
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Steve Black	Vehicle health management
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Gary Letchworth	Ground operations assessment
Arlene Reese	CAD drawings
Pat Scott	Ground operations assessments

Lockheed Martin Skunk Works

Wally Eshelman	Reliability, maintainability, and supportability assessment

Pratt & Whitney

Paul Greasely	Russian propulsion assessment
Larry Tanner	Russian propulsion assessment

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1.0 Introduction

The Advanced Transportation System Studies (ATSS) Technical Area 2 (TA-2) contract, NAS8-39208, was one of four fixed-price NASA Research Announcements (NRAs) contracted by the Marshall Space Flight Center (MSFC) that assessed space transportation system requirements. The ATSS NRAs were conducted after the termination of the National Launch System (NLS) Phase B studies. The original charter of the TA-2 contract was to perform conceptual (Phase A) definition and analysis of crewed and un-crewed heavy lift launch vehicles (HLLVs) having injected payload masses of 150-300 Klbm to low Earth orbit, that would represent missions to either the Moon or Mars. During the course of the TA-2 contract, the Lockheed team was asked to look at a wide variety of additional launch vehicle configurations, including derivatives of the National Launch System (NLS) and Early Heavy Lift Launch Vehicle (EHLLV) for lunar missions, 50-80 Klbm payload-class expendable launch vehicles (ELVs) for the Access to Space Option 2 team, and Single Stage to Orbit (SSTO) concepts that augmented the Access to Space Option 3 team's candidate vehicle concept analyses. Each of the TA-2 contract's various vehicle assessments involved vehicle and propulsion sizing and ascent performance analyses, major vehicle component layout and load path definition, and ground operations and launch site facility analyses. Qualitative assessments of the associated launch system nonrecurring and recurring costs were made, and quantitative estimates of non-recurring and recurring launch system costs were performed by ECON, Incorporated, for the lunar/Mars mission scenarios. Figure 1-1 summarizes the significant events of the TA-2 contract.

Contract Major Events Chronology

Jan.-May 1992: First Lunar Outpost (FLO) Support (via NLS NRA)

- **NLS-derived parallel burn HLLV concept sizing and performance assessments**
- **FLO Technical Interchange Meeting support**

June-Sept. 1992: FLO Support (via TA-2 Contract)

- **FLO HLLV Team Preliminary Design Status report editing**
- **HLLV design goals identification and ranking via concurrent engineering process for min. DDT&E cost , min. recurring cost, and min. risk scenarios**
- **HLLV configuration identification and sizing for min. DDT&E scenario**
- **FLO HLLV tower drift requirements assessment (nominal and dispersed)**

Figure 1-1 TA-2 Significant Events Summary

Contract Major Events Chronology

June-Sept. 1992 : FLO Support

- First-order HLLV launch site evaluation
- Ground op.s assessments of mixed fleet architectures supporting Red/Blue teams and SSF-assembly "Super Red Team"
- HLLV propulsion requirements identification at Propulsion Synergy Group quality function deployment meeting
- FLO TIM support (monthly)

Oct.-Dec. 1992 : FLO Support

- Early HLLV derived parallel-burn and series-burn HLLV configuration and ground op.s assessments
- Alternative HLLV structural/manufacturing design concept assessments
- FLO TIM support

Jan-Mar. 1993: FLO Support

- Liquid and hybrid 50+K two-stage concept definition and assessment for evolution into FLO HLLV strap-on boosters
- Ground op.s assessments of mixed fleet architectures supporting SSF-assembly "Super Red Team" and multiple-booster FLO HLLV configurations (8 F-1A boosters vs. 7 RD-170 boosters)
- Integrated HLLV vehicle health management requirements identification supporting enhanced manufacturing and operability

April-May 1993: Access-to-Space Support

- Liquid, hybrid, solid two-stage concept definition and assessment
- Russian propulsion preliminary assessment
- Mixed fleet ground operations assessments

Figure 1-1 TA-2 Significant Events Summary (Continued)

Contract Major Events Chronology

June 1993- February 1994: Access-to-Space Support

- Lifting body and vertical-takeoff/vertical-landing Single Stage to Orbit (SSTO) vehicle design and assessment
- Concurrent engineering quality function deployment qualitative assessment of vertical-takeoff/vertical-landing versus vertical-takeoff/horizontal-landing SSTO pros and cons
- Concurrent engineering assessment of bipropellant versus tripropellant main propulsion impacts on SSTO operability
- Russian propulsion technology assessment final report (by Pratt & Whitney)

March 1994- June 1994: Access-to-Space Support

- SSTO turnaround processing assessment report (by Skunk Works)
- Russian tripropellant injector hot-fire test results report (by Pratt & Whitney)

Figure 1-1 TA-2 Significant Events Summary (Concluded)

2.0 First Lunar Outpost

In response to President Bush's Space Exploration Initiative (SEI) declaration of returning astronauts to the Moon by 2020, the NASA Headquarters Office of Exploration (Code X), under the leadership of Dr. Michael D. Griffin, devised the concept of First Lunar Outpost (FLO) during the fall of 1992. The central premise of FLO was to place, via a single launch with no on-orbit assembly, a lunar habitat module onto the lunar surface, followed by an additional launch of a crewed four-person lunar lander. The FLO mission was to accomplish a significant period of lunar exploration for 45-60 days; i.e., more than simply putting "flags and footprints" on the lunar surface. Griffin sough to identify the requirements for accomplishing the FLO single-launch mission by 1999, as a fast-track way of returning to the Moon, followed by a crewed mission to Mars two to four years later. The single-launch FLO mission resulted in the requirement for an HLLV capable of injecting a minimum of 93 metric tons, after burnout of the translunar injection (TLI) stage. While the single-launch scenario required a larger HLLV than that required for on-orbit assembly of a TLI stage and its lunar habitat or crewed lander, Griffin maintained that the programmatic risk and associated life cycle cost would be greatly reduced over a multiple-launch scenario requiring on-orbit assembly. The shear mass of the requisite payloads for the single-launch scenario, equivalent to that of a 4-8-4 locomotive, required launch vehicles with over twice the lift capacity of the venerable Saturn V, vehicle lengths of approximately 400 feet, and diameters from 27.6 feet (Shuttle External Tank diameter) to 38 feet (for a no-hammerhead condition with the 38 feet diameter lunar mission payload shroud).

The original charter of the TA-2 contract, as defined in the NRA request for proposals, was to define and assess candidate HLLV configurations that were based upon existing hardware (Shuttle-derived) or designs (Saturn V-derived), as well as clean-sheet approaches. Two basic families of HLLV concepts were assessed against the FLO program requirements: those derived from varying degrees of commonalty with National Launch System (NLS) launch system hardware, and those derived from varying degrees of commonalty with Early Heavy Lift Launch Vehicle (EHLLV) system hardware. The original NLS "heavy lift" concepts had payload capabilities of 100-150 Klbm to low Earth Orbit (LEO) and utilized Space Shuttle solid rocket booster strap-ons and a core vehicle derived from Space Shuttle External Tank (ET) hardware and powered by three to six Space Transportation System Main Engines (STMEs) . The EHLLV concept was similar to that of the NLS HLLV design but utilized Space Shuttle Main Engines. The politics of seeking to maximize the leveraging of existing or "sunk" launch vehicle hardware development costs caused the NASA customer to de-emphasize the assessment of clean-sheet approaches. Some "clean-sheet" HLLV concepts were defined that consisted of constant-diameter series-burn stages (i.e., each stage had the same outer diameter); with the diameter chosen to be the same as the FLO 38 foot diameter payload shroud. The manufacturability issues associated with the stage diameter resulted in less attention being paid to the further analysis of these particular concepts.

2.1 FLO Design Requirements and HLLV Design Drivers

A two-day concurrent engineering brainstorming session was held on August 5-6, 1992 to identify desirable HLLV system attributes that were first-order HLLV design drivers, subject to the requirements and desirements contained in the First Lunar Outpost Requirements and Guidelines (FLORG) document produced by the inter-NASA FLO requirements team. The affinity process was utilized to distill the attributes into sixteen idealized design characteristics. The design characteristics were further categorized into three programmatic approaches to defining the HLLV launch system: minimizing design, development, test, and engineering (DDT&E) funding requirements; minimizing recurring funding requirements; and minimizing programmatic risk. Figure 2.1-1 summarizes the sixteen ideal HLLV system characteristics.

HLLV LAUNCH SYSTEM "IDEAL" DESIGN CHARACTERISTICS

1) Minimize Environmental Impacts
2) Maximize Crew Safety
3) Minimize Aerodynamic Impacts
4) Maximize Launch Availability
5) Maximize Operability
6) Minimize Plume Impacts
7) Maximize Vehicle Reliability
8) Maximize Vehicle Performance
9) Maximize Existing Infrastructure
10) Minimize Design Complexity
11) Minimize Vehicle Volume
12) Minimize Number of Flight Elements
13) Minimize Number of Engines
14) Maximize Flight Element Commonality
15) Maximize Evolution Capability
16) Minimize Safety Hazards

Figure 2.1-1 Principal HLLV System Design Characteristics

Within each of the three programmatic categories, the sixteen design characteristics were ranked numerically according to their relative influence, or impact, on the three design categories. Figure 2.1-2 summarizes the results of the characteristic rankings, with a high ranking being a strong design influence. The correlation "direction" of the influence, either a positive (beneficial), negative (detrimental), or neutral (shown as no influence) was also identified, as shown in Figure 2.1-3.

DESIGN CHARACTERISTIC

DESIGN CATEGORY	Min Environmental Impacts	Max Crew Safety	Min Aero Impacts	Max Launch Availability	Max Operability	Min Plume Impacts	Max Vehicle Reliability	Max Vehicle Performance	Max Existing Infrastructure	Min Design Complexity	Min Vehicle Volume	Min Number Flight Elements	Min Number Engines	Max Flight Element Common	Max Evolution Capability	Min Safety Hazards
Min DDT&E	8	1	8	4	0	0	8	0	8	9	7	5	4	8	7	2
Min Recurring	5	7	0	9	9	0	9	3	5	9	6	9	8	9	0	9
Min Risk	4	9	0	8	9	0	9	4	9	9	0	7	8	9	2	9

Note: Top 9 scores selected as top-level design drivers for each category

Figure 2.1-2 HLLV System Design Characteristic Rankings

DESIGN CATEGORY	Min Environmental Impacts	Max Crew Safety	Min Aero Impacts	Max Launch Availability	Max Operability	Min Plume Impacts	Max Vehicle Reliability	Max Vehicle Performance	Max Existing Infrastructure	Min Design Complexity	Min Vehicle Volume	Min Number Flight Elements	Min Number Engines	Max Flight Element Common	Max Evolution Capability	Min Safety Hazards
Min DDT&E	+		-				-		+	+	+	+		+	-	
Min Recurring		-		+	+		+			+		+	+	+		+
Min Risk		+		+	+		+		+	+			+	+		+

Note: Correlation only evaluated for top-level design drivers

Figure 2.1-3 HLLV System Design Characteristic Correlations

The top nine design characteristics were then baselined for the remainder of the HLLV design studies as the focusing design drivers, and are summarized by programmatic design category in Figure 2.1-4.

Min DDT&E	Min Recurring	Min Risk
Min Environmental Impacts	Max Crew Safety	Max Crew Safety
Min Aero Impacts	Max Launch Availability	Max Launch Availability
Max Vehicle Reliability	Max Operability	Max Operability
Max Existing Infrastructure	Max Vehicle Reliability	Max Vehicle Reliability
Min Design Complexity	Min Design Complexity	Max Existing Infrastructure
Min Vehicle Volume	Min Number Flight Elements	Min Design Complexity
Min Number Flight Elements	Min Number Engines	Min Number Engines
Max Flight Element Common	Max Flight Element Common	Max Flight Element Common
Max Evolution Capability	Min Safety Hazards	Min Safety Hazards

Figure 2.1-4 Baselined HLLV Design Drivers

The effect of the HLLV system design on recurring ground and flight operations was recognized from the very beginning of the TA-2 contract as having the most fundamental influence on the affordability and viability of the system. As a result, the LMSO TA-2 teammates identified which of the primary design characteristics had a correspondingly profound influence on the recurring operations cost of the system. Figure 2.1-5 summarizes that assessment.

Design Category

Design Characteristics

Minimal DDT&E	Minimal Recurring	Minimal Risk
☑ Min Environmental Impacts	☐ Max Crew Safety	☐ Max Crew Safety
☐ Min Aero Impacts	☑ Max Launch Availability	☑ Max Launch Availability
☐ Max Vehicle Reliability	☑ Max Operability	☑ Max Operability
☑ Max Existing Infrastructure	☐ Max Vehicle Reliability	☐ Max Vehicle Reliability
☑ Min Design Complexity	☑ Min Design Complexity	☑ Max Existing Infrastructure
☑ Min Vehicle Volume	☑ Min Number Flight Elements	☑ Min Design Complexity
☑ Min Number Flight Elements	☑ Min Number Engines	☑ Min Number Engines
☑ Max Flight Element Common	☑ Max Flight Element Common	☑ Max Flight Element Common
☐ Max Evolution Capability	☑ Min Safety Hazards	☑ Min Safety Hazards

☑ Operations Impact

Figure 2.1-5 HLLV Design Characteristics that Influence Operations

LMSO also identified which of the SEI requirements and guidelines (either specified in the FLORG or by the TA-2 team) had an influence on operations, as summarized in Figure 2.1-6.

REQUIREMENT	GUIDELINE	IMPACTS OPERATIONS	
☑			The Earth to Moon transportation system shall provide the capability to emplace 27.5 t (including 10% manager's reserve) on the Lunar surface in a single flight. Current assessment is 34t of cargo with margin resulting in 93t to TLI
☑			A single HLLV shall be utilized for each flight to the Moon
	☑		The HLLV shall provide the capability for designed growth to 250t to 220NM
☑			Flight elements shall provide the capability to access any Lunar latitude or longitude
	☑		The HLLV shall provide the capability for launch as early as 1999
	☑	☑	The capability shall be provided to support 4 flights per year
☑			The usable shroud size for Lunar flights shall be 38 x 60 ft (goal)
☑			The HLLV shall be designed for no engine out on the core, boosters or upper stage(s)
☑			The HLLV shall be sized to provide launch capability any day during the lunar cycle
☑			The HLLV shall provide the vehicle health monitoring capability to provide notification that an abort condition exists. Launch escape system (LES) jettisoned at shroud separation (400Kft)

Reference: *First Lunar Outpost Requirements and Guidelines (FLORG)*, EXPO-T1-920001EXPO, 6/10/92

Figure 2.1-6 SEI Requirements and Guidelines that Influence Operations

Requirement	Guideline	Impacts Operations	
	✓	✓	Time between cargo and piloted flights: 60 days (goal)
	✓	✓	No pad services except fueling, checkout and launch
	✓		Maximum acceleration during boost phase: 4g (goal)
✓		✓	Capability to launch from 72 – 108 deg. azimuth
	✓	✓	Maximum dynamic pressure during ascent: 900 psf
	✓		Minimum liftoff thrust-to-weight: 1.2
✓			Dry weight contingency: 10%
✓			Ascent flight performance reserve: 1% delta-V
	✓		Jettison shroud / nosecone at 400Kft
	✓		Lunar direct ascent profile thru 100NM circular Earth orbit
	✓	✓	Method of on-pad holddown during engine start
	✓	✓	Primary avionics located on TLI stage

Reference: *First Lunar Outpost Requirements and Guidelines (FLORG)*, EXPO-T1-920001EXPO, 6/10/92

Figure 2.1-6 SEI Requirements and Guidelines that Influence Operations (Continued)

Requirement	Guideline	Impacts Operations	
✓		✓	Maximum HLLV DDT&E less than $4 billion
✓			Maximum annual HLLV budget of $2 billion for all life cycle
✓		✓	Maximum 5 years Phase C/D new start to ILC
✓			Utilize existing/planned engines only.
✓			Minimum HLLV reliability .985
✓		✓	Utilize KSC launch site and JSC MOD infrastructure
✓		✓	All vehicle stages require safe disposal (deorbit or safe orbit) *
✓		✓	TLI stage will have an attitude control system *
	✓		Minimize dependance on other development programs *
	✓	✓	Mission control will be capable of monitor & control of all flight elements *
	✓	✓	Payload integration will include hazardous processing *
✓			Crew abort capability (water landing) during all piloted ascent phases *

* Reference: *First Lunar Outpost Requirements and Guidelines (FLORG)*, EXPO-T1-920001EXPO, 6/10/92

Figure 2.1-6 SEI Requirements and Guidelines that Influence Operations (Concluded)

2.2 Stage Propulsion Options

The HLLV concepts presented unique requirements for stage main propulsion. Because of the launch vehicle hardware being expendable, as specified in the FLO requirements from NASA, any strap-on boosters or core vehicle first stages naturally wanted to have high density-impulse main propulsion with very large-thrust engines, whereas the upper stages wanted to have high specific impulse and moderate-thrust engines. High density-impulse dictated use of hydrocarbon-based fuels, and high specific impulse usually dictated hydrogen as the fuel. Performance specification and technology data on domestic propulsion concepts were obtained directly from Aerojet and Rocketdyne. Data on candidate Russian propulsion concepts were obtained from Aerojet and Pratt & Whitney. Figure 2.2-1 summarizes the domestic main propulsion concepts that were assessed.

				Thrust, Klbf*		Specific Impulse, Sec.*	
Engine	Country	Fuel	Oxidizer	Sea Level	Vacuum	Sea Level	Vacuum
F-1	US	RP-1	LOX	1522	1746	265	304
MA-5A**	US	RP-1	LOX	423.5/50.5	473.4/85.0	264/220	295/309
RS-27A	US	RP-1	LOX	200	237	255	302
H-1	US	RP-1	LOX	205	230	264	296
LR-87	US	RP-1	LOX	300	344.4	252	289
XLR-109	US	RP-1	LOX	500	---	265	---
J-2	US	LH2	LOX	161.4	230	293.7	422.7
J-2S	US	LH2	LOX	201	265	330	435
M-1	US	LH2	LOX	---	1500	---	428
M-1A	US	LH2	LOX	1300	---	344.5	---
SSME	US	LH2	LOX	373.5	468.4	362	454

NOTES:
* 100 percent rated power level
** MA-5A Data: Booster(2 Thrust Chambers)/Sustainer (1 Thrust Chamber)

Figure 2.2-1 Domestic HLLV Main Propulsion Candidates

Figure 2.2-2 summarizes the candidate Russian propulsion concepts that were assessed.

				Thrust, Klbf*		Specific Impulse, Sec.*	
Engine	Country	Fuel	Oxidizer	Sea Level	Vacuum	Sea Level	Vacuum
RD-107	CIS	Kerosene	LOX	184.6	224.8	257	314
RD-108	CIS	Kerosene	LOX	167.5	211.5	248	315
RD-170	CIS	Kerosene	LOX	1631	1777	309	337
RD-120	CIS	Kerosene	LOX	---	181.5	---	350
NK-33	CIS	Kerosene	LOX	339	378	297	331
NK-43	CIS	Kerosene	LOX	---	395	---	346
RD-0120	CIS	LH2	LOX	326	441	354	452.5

NOTES:
* 100 percent rated power level

Figure 2.2-2 Russian HLLV Main Propulsion Candidates

2.3 NLS-Derived HLLVs

The TA-2 contract was awarded to LMMS in October of 1991, but was not executed until May of 1992. From January of 1992 until May of 1992 partial TA-2 funding was provided through LMMS' existing NLS contract to allow LMMS to define NLS-derived HLLV concepts in support of the FLO Interim Report that was scheduled for delivery to NASA Headquarters in May of 1992. During that period of time, it was believed that an NLS family of medium-lift launch vehicles was going to be developed as a joint NASA/Air Force activity, and could provide building blocks for a lunar/Mars HLLV. LMMS' approach was to identify vehicle concepts that minimized non-recurring development costs. The FLO launch vehicle definition manager, Gene Austin, provided the additional design constraints of: parallel-burn (i.e., core vehicle with strap-on boosters), vehicle element diameters common with the Shuttle's External Tank (ET), and utilization of Rocketdyne's F-1A main engine for the strap-on boosters. The F-1A was an improved version of the venerable Saturn V first stage engine with up to 2 million pounds of sea level thrust and deep throttling down to 65% percent of full rated power. A significant study finding was that the NLS-sponsored Space Transportation Main Engine (STME) was not a viable option from a performance basis as an upper stage or TLI stage. Booster configurations that leveraged high density-impulse provided minimum dry mass solutions, which in turn provided the lowest development and non-recurring unit costs. The TA-2 team also supported the monthly FLO technical interchange meetings during the January-May period of 1992.

Figure 2.3-1 summarizes some of the primary parallel-burn HLLV design options that were assessed. A very specific set of vehicle and operations groundrules and constraints were imposed, as summarized in Figures 2.3-2 through 6. Vehicle sizing and mass properties estimation tools were developed under the TA-2 contract to perform first-order vehicle definition, down to the subsystem level. The sizing tools were principally derived from the works of I. O. MacConochie and P. J. Klich (NASA TM 78661), with substantial modifications from a variety of technical sources on advanced materials, empirical structural weight estimation, and efficient design of extremely large space transportation systems. These sizing tools were to form the basis of the vehicle sizing tools that were used for the course of the TA-2 contact. An assessment of launch pad tower clearance requirements for worst-case vehicle drift concluded that a minimum lift-off thrust-to-weight of 1.25 was required, which resulted in the conclusion that use of F-1A powered strap-on boosters required a minimum of three or more F-1As per booster.

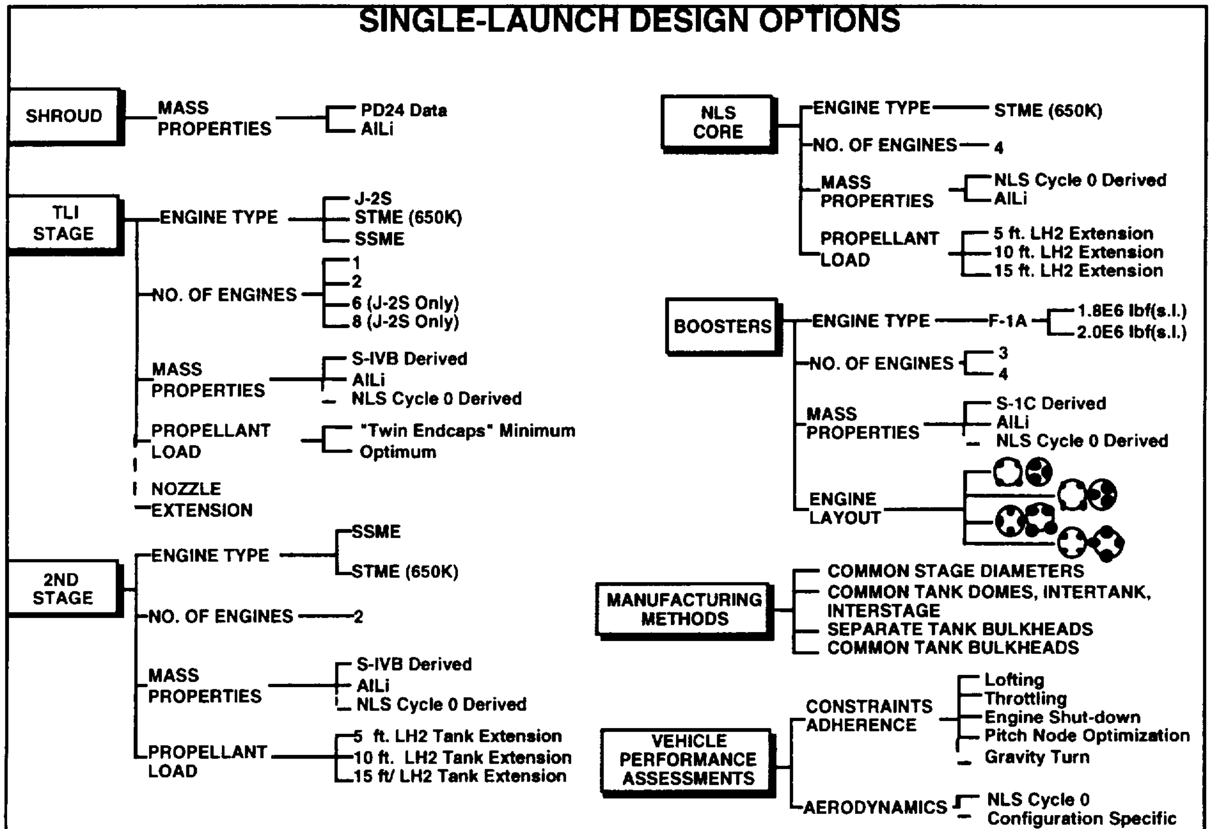


Figure 2.3-1 NLS-Derived Parallel-Burn HLLV Design Trade Tree

GROUNDRULES AND CONSTRAINTS

CORE VEHICLE

- Four STMEs
- 5 ft. LH2 tank stretch (1.69E06 lbm propellant load)
- Expendable hardware
- TBD delta weight addition to intertank to carry booster thrust loads
- No air start of core STMEs (a manned mission requirement that carries over to unmanned)
- TBD throttle profile; 70% (new STME groundrule) or 100% RPL step throttle

Figure 2.3-2 NLS-Derived Parallel-Burn HLLV Core Vehicle Groundrules and Constraints

GROUNDRULES AND CONSTRAINTS (Continued)

TLI AND SECOND STAGE

- One or more STMEs, SSMEs, or J-2Ss
- Minimum propellant load limited to dome-to-dome LOX tank design (590,000 lbm total propellant)
- Post-insertion burn or pre-insertion burn (which ever is optimal)
- Mass fraction curves (function of propellant load) for S-IVB derived stage with J-2S, STME, or SSME engines
- 30 day boil-off MPS margin (from STV trade study groundrules)
- Expendable hardware

Figure 2.3-3 NLS-Derived Parallel-Burn HLLV TLI and Second Stage Groundrules and Constraints

GROUNDRULES AND CONSTRAINTS (Continued)

BOOSTER

- Three or four F-1As per booster
- 75% RPL step throttle (unknown throttle altitude constraint)
- Mass fraction curve (function of propellant load) for S-1C derived booster
- Expendable hardware
- 27.6 ft (NLS-derived) tank diameter
- Boosters support the core vehicle on the pad
- Booster thrust taken out at forward attach

Figure 2.3-4 NLS-Derived Parallel-Burn HLLV Booster Groundrules and Constraints

GROUNDRULES AND CONSTRAINTS (Continued)

GENERAL

- Flight Performance Reserve (FPR) for each stage is 1% of stage delta V
- 10% inert mass margin for growth
- Unusable propellant is 0.05 x Wprop (from NASA TM 78661, "Techniques for the Determination of Mass Properties of Earth-to-Orbit Transportation Systems", NASA LaRC, June, 1978)
- Lift-off thrust/weight ratio minimum is 1.25:1
- No engine-out protection for making mission
- 4.5 Gs maximum thrust acceleration constraint
- Optimal pitch-rate steering during ascent
- +/– 5000 psf-degree Qbar-alpha constraint during atmospheric flight
- Shroud pre-defined by MSFC
- Shroud jettison at 400,000 feet (geodetic altitude)

Figure 2.3-5 NLS-Derived Parallel-Burn HLLV General Groundrules and Constraints

GROUNDRULES AND CONSTRAINTS (Concluded)

OPERATIONS

- 7 days between launches for the Dual Launch scenario
- 60 days (minimum) between launches for the Single Launch scenario
- VAB high bay door height constrains the total vehicle length; limit of 390-400 ft
- VAB high bay crane hook height limit, including height of lifting equipment, imposes a similar vehicle length limit of 390-400 ft
- VAB high bay side door height modifications are determined to be minor in cost impact; current height of 111 ft
- MLP width constrained to current pad support post spacing
- TBD MLP length growth allowed; limited by crawler overhang

Figure 2.3-6 NLS-Derived Parallel-Burn HLLV Operations Groundrules and Constraints that vehicle configurations with a minimum of two F-1A engines per strap-on booster did not provide sufficient thrust for viable configurations. Figure 2.3-7 illustrates the resulting set of candidate HLLV configurations and Figure 2.3-8 summarizes the significant conclusions.

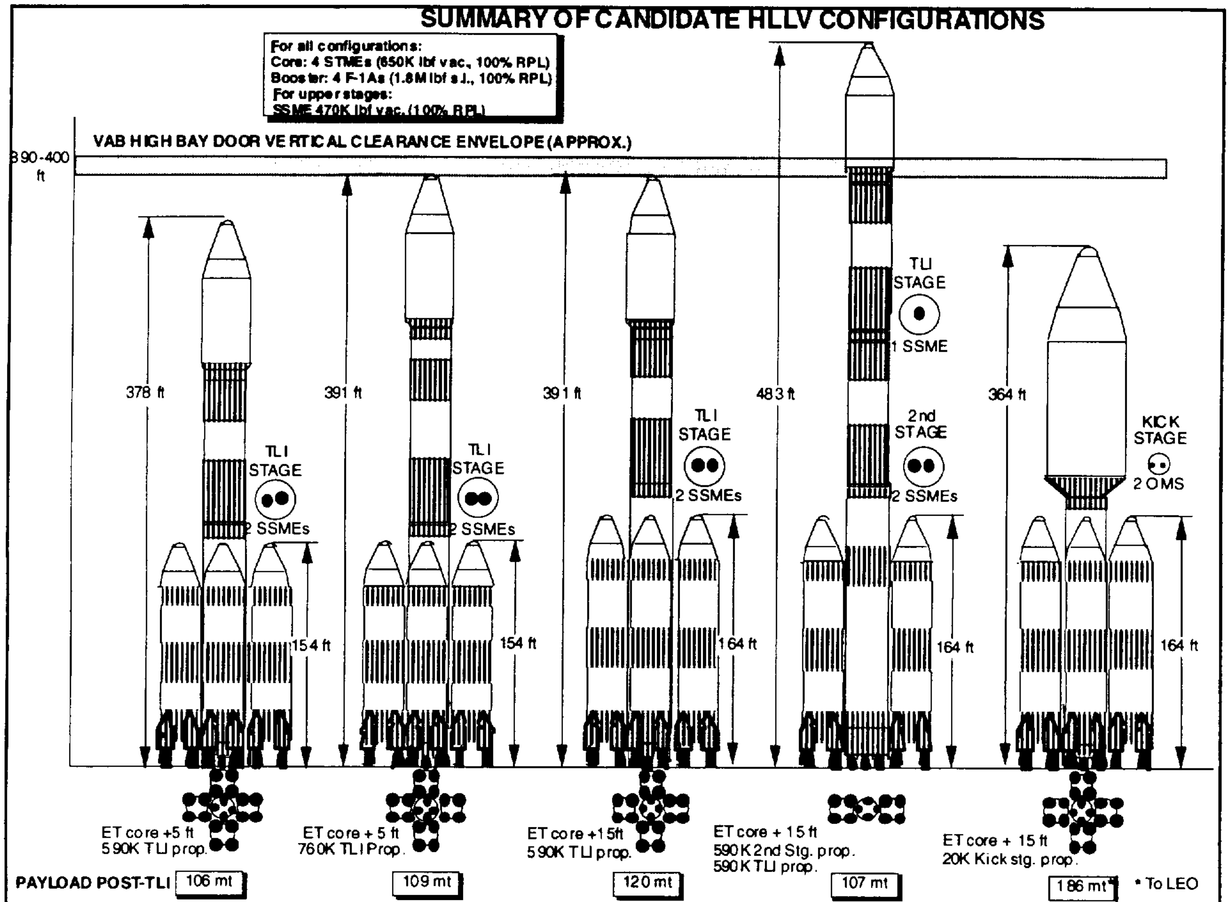


Figure 2.3-7 NLS-Derived Parallel-Burn HLLV Candidate Configurations

CONCLUSIONS

- To meet a lunar mission requirement of >90 mt post-TLI and not violate the VAB high bay door clearance, must use four F-1A powered boosters with the NLS-derived core
- Use of four F-1As on four NLS-derived boosters with 2 SSMEs on the TLI stage provide approximately 115 mt of payload post-TLI
- Use of F-1A throttling for Qbar control gains approximately 30,000 lbm of payload post-TLI
- Use of STMEs for upper stage applications cannot compete with SSMEs from vehicle sizing and performance standpoints
- Use of 2 boosters with 3 F-1As (1.8E6 lbf s.l. thrust) will not allow greater than 90 mt post-TLI, without using both a second stage and a TLI stage
- Use of common propellant tank, intertank, aft skirt, and forward skirt/interstage piece-parts still allows reasonable vehicle configurations to be designed, without incurring inordinate performance losses
  --Associated cost savings could offset non-optimality of the design
- The VAB high bay doors can be economically modified, up to a point, to accommodate booster height, but not economically for core vehicle height

Figure 2.3-8 NLS-Derived Parallel-Burn HLLV Significant Conclusions

LMMS was also requested to perform an initial assessment of two families of "clean-sheet", series-burn, or monolithic, HLLV configurations: one family that utilized a 38-foot common diameter for all stages, common with the lunar mission payload shroud diameter; and one family that utilized 50-foot diameter stages, common with the Mars mission payload shroud diameter. A high-level intent was to maximize utilization of NLS-derived components, although the propellant tankage was not constrained to be derived from ET hardware. The following first-order design parameters were assessed for the series-burn configurations: number of stages, stage propellant combination, type of stage engine, number of stage engines, and stage tankage configuration. Various means were utilized to minimize total vehicle length, in order to keep from violating the Vehicle Assembly Building (VAB) 400-foot hook height limit, and to maximize usable propellant tank volume given the extremely large stage diameters, such as use of toroidal and cluster propellant tanks. Figure 2.3-9 summarizes the series-burn design groundrules and assumptions. Seven different vehicle configurations were preliminarily assessed. The fourth design case was found to be most promising, and was assessed in further detail, as summarized in Figure 2.3-10. The performance of the Case 4 configurations was confirmed through 3-degrees-of-freedom (3-DOF) trajectory simulations using a heavily modified advanced vehicle design version of the standard Space Shuttle flight design tool Simulation and Optimization of Rocket Trajectories (SORT). Figure 2.3-11 illustrates the Case 4 candidate configurations, with the associated payload mass (post-TLI) shown in parentheses.

Groundrules:

- Figure of merit was number of engines
- Two, three and four stage vehicles were considered
- First stage engines were F-1As
- Second stage engines were F-1As, SSMEs or STMEs
- Third and fourth stage engines were SSMEs and STMEs
- Saturn stage mass fractions, interstage weights and instrument unit weights were used
- Payload was 200,000 lbm (90,700 kg) to TLI
- Rocket equation program was used to do sizing
- Mission velocity to TLI varied as a function of initial thrust-to-weight
  - -- If No1= 1.40 g, mission velocity= 40,700 ft/sec
  - -- If No1= 1.50 g, mission velocity= 40,200 ft/sec

Figure 2.3-9 NLS-Derived Series-Burn HLLV Groundrules and Constraints

ALTERNATE VEHICLE TANKAGE OPTIONS

- Case four was selected for detailed analysis
- Programs were written to estimate stage weights for three vehicle configurations
  - -- Configuration A
    - 38 foot stage diameter
    - Stages one and two used propellant tanks with elliptical end caps
    - Stage three used a cluster of seven propellant tanks (two for LOX and five for LH2)
  - -- Configuration B
    - 38 foot stage diameter
    - Lower propellant tank on all stages used toroidial end caps
  - -- Configuration C
    - 50 foot stage diameter
    - Lower propellant tanks on stages one and two used toroidal end caps
    - Stage three used a cluster of seven propellant tanks (two for LOX and five for LH2)

Figure 2.3-10 NLS-Derived Series-Burn HLLV Configuration Definition Summary

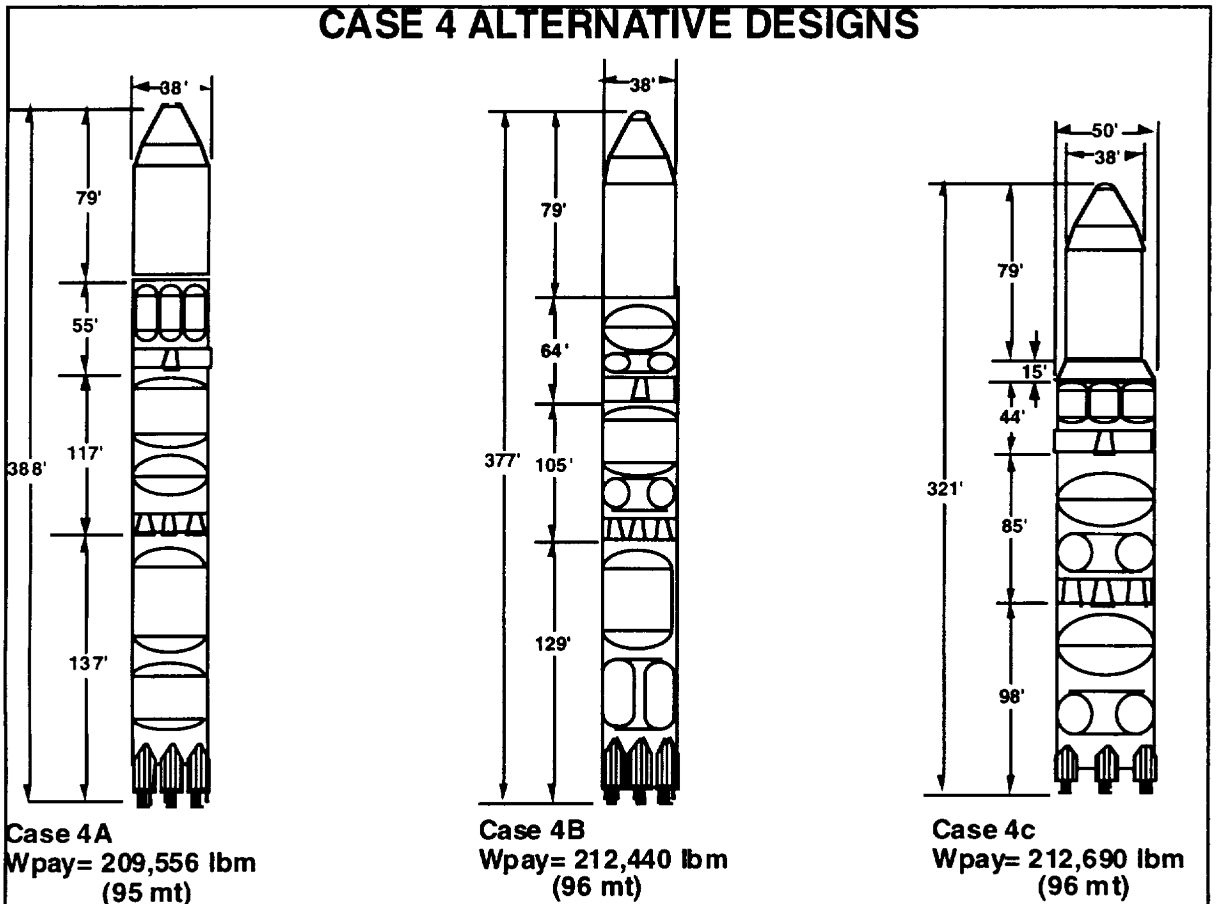


Figure 2.3-11 NLS-Derived Series-Burn (Case 4) HLLV Candidate Configurations

During this same time period, personnel at the Lockheed Space Operations Company (LSOC), under TA-2 sponsorship and NASA Kennedy Space Center (KSC) direction (out of KSC's Future Launch Systems Office), performed a series of SEI launch site facilitization and launch processing requirements assessments; addressing both single-launch and dual-launch FLO program scenarios, and Saturn-V-derived and NLS-derived candidate launch vehicle configurations. One particular study focused on the ground operations assessment of two alternate lunar HLLV configurations that both utilizing the lunar single launch concept. One configuration featured an ET-derived core (SSME engines) with seven LOX/RP-1 strap-on boosters and RD-170 engines. The other configuration utilized the same core stage with eight LOX/RP-1 strap-on boosters with F-1A engines. Results of this assessment indicated that there was no significant ground operations discriminator between the two proposed lunar HLLV configurations. The launch site processing scenario, shown here, is interchangeable between the two vehicle options. The predicted scheduled event burden was similar and the launch site station set (facility) solutions were identical. Both options satisfied the minimum lunar launch interval and launch manifest requirements. Figure 2.3-12 illustrates the basic ground operations scenario for processing of the lunar HLLVs.

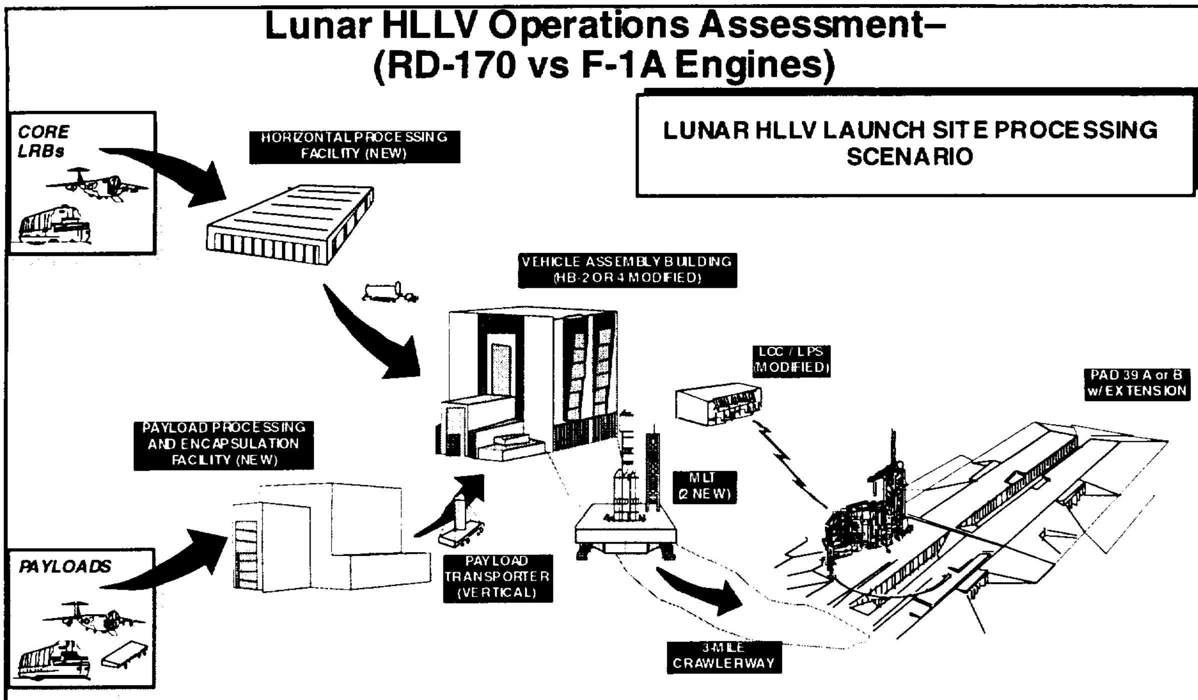


Figure 2.3-12 Lunar HLLV Launch Site Processing Scenario

A detailed assessment of the launch site operational impacts for the proposed lunar dual launch concept was also performed at the request of the KSC Future Launch Systems Office. The lunar dual launch concept required two successful HLLV launches within an 8-day maximum launch interval for each lunar cargo or piloted mission opportunity. The nominal interval between lunar cargo and piloted missions was 60 days. The candidate HLLV launch system was an NLS-derived core with four STMEs, two LOX/RP-1 strap-on boosters with two F-1A engines each, and a LOX/LH_2 TLI stage for the first launch or a hypergolic kickstage for the second launch in the two-launch-per-lunar-mission sequence. The issue of launch site schedule feasibility was addressed under a mixed-fleet manifest scenario of eight Space Shuttle, eight lunar, and two NLS-2 (medium-lift) flights annually. A preliminary ground processing scenario was developed, and bottoms-up processing timelines were estimated for each major flight hardware component. These timeline estimates, associated facility resources, and integrated ground processing logic were incorporated into a network-based project management system. The summary output of this effort is shown Figure 2.3-13. The lunar dual-launch concept launch interval requirements were prioritized and maintained. The lunar, Shuttle, and NLS annual flight rates were also achieved.

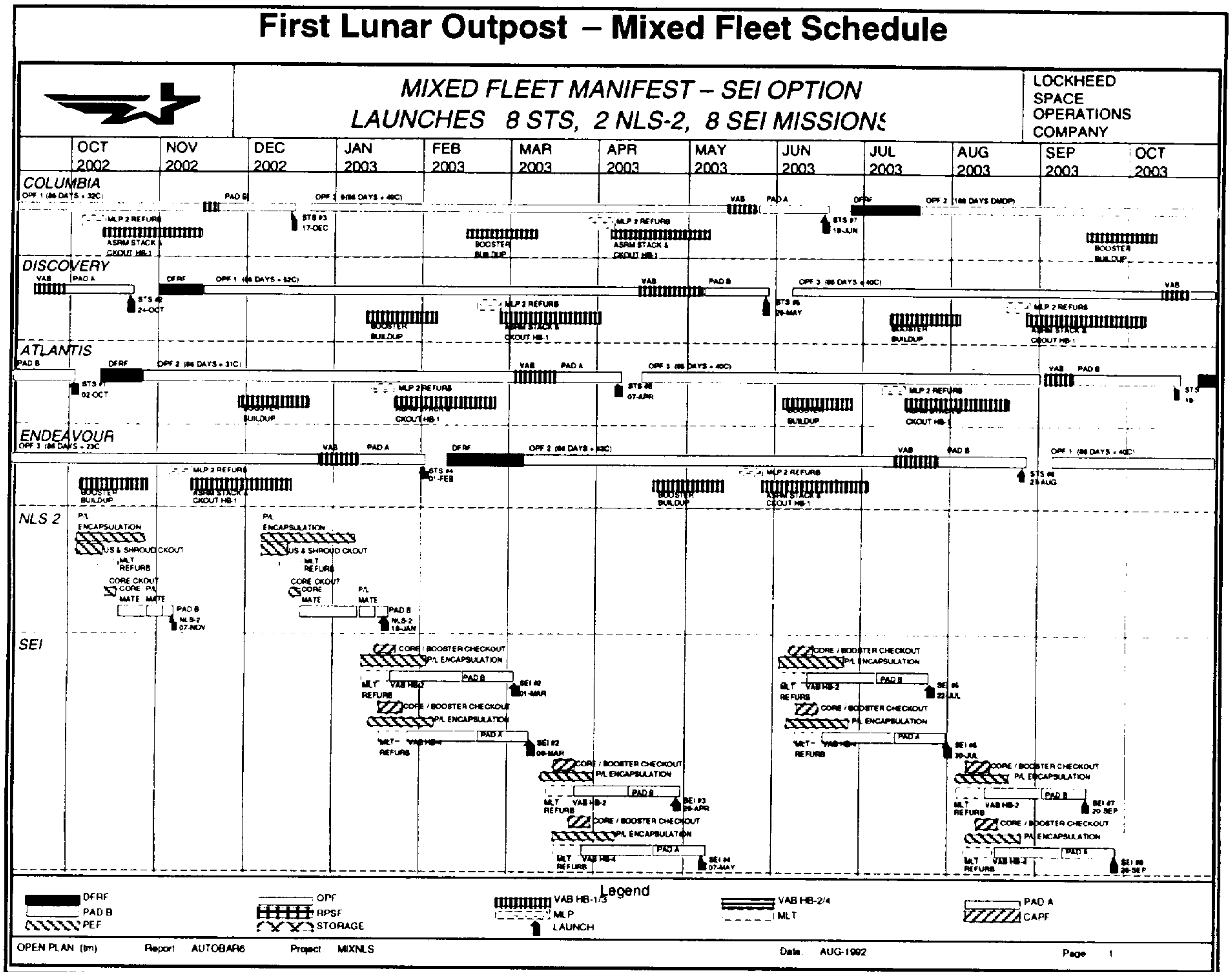


Figure 2.3-13 Mixed Fleet Ground Operations Timelines

2.4 Early Heavy Lift Launch Vehicle Derived HLLVs

During the period of October through December of 1992, a series of parallel-burn and series-burn launch vehicle configurations were defined that sought to maximize hardware commonalty with the Shuttle-derived Early Heavy Lift Launch Vehicle (EHLLV). Areas of hardware commonalty included use of the EHLLV core vehicle propellant tankage for the FLO vehicle, strap-on stages, and the use of Space Shuttle Main Engines (SSMEs) for the TLI stage. The EHLLV core vehicle's propellant tank design was directly derived from the Space Shuttle's External Tank. Each of the FLO vehicle concepts utilized a new 38-foot diameter payload shroud that was sized to encapsulate a lunar habitat lander having a transverse-mounted Space Station Freedom derived pressurized crew module. Figure 2.4-1 summarizes the major aspects of the sizing philosophy that was used to define the candidate lunar HLLV configurations.

General Sizing Philosophy

- **Minimize number of engines per stage while striving for "reasonable" stage thrust-to-weight ratio**
  --Simplifies propellant feed subsystem design
  --Helps to minimize stage structural mass
  --Helps to minimize stage unit cost
  --Helps to maximize stage reliability and lower FMEA/CIL items

- **Minimize stage dry mass**
  --Helps to minimize stage unit cost

- **Maximize stage-to-stage hardware design commonality**
  --Helps to minimize stage unit cost and manufacturing nonrecurring cost

- **Seek to have no more than one "new" engine design for a given vehicle configuration**
  --Helps to minimize DDT& E costs and programmatic risk

- **Consider technologies that would be available to support a 2005 first launch date**

- **Perform initial sizing sensitivity assessment of minimum-GLOW designs versus minimum-dry-weight solutions**

Figure 2.4-1 General Sizing Philosophy for EHLLV-Derived Configurations

Figure 2.4-2 shows the resulting candidate parallel-burn EHLLV-derived lunar mission launch vehicle concepts that were identified and assessed. The large arrow in the figure indicates the path of design evolution from the basic EHLLV core vehicle, with the evolution shown in terms of least relative DDT&E funding requirements to highest relative DDT&E requirements. Figures 2.4-3 through 7 summarize the qualitative pros and cons of the respective parallel-burn vehicle types.

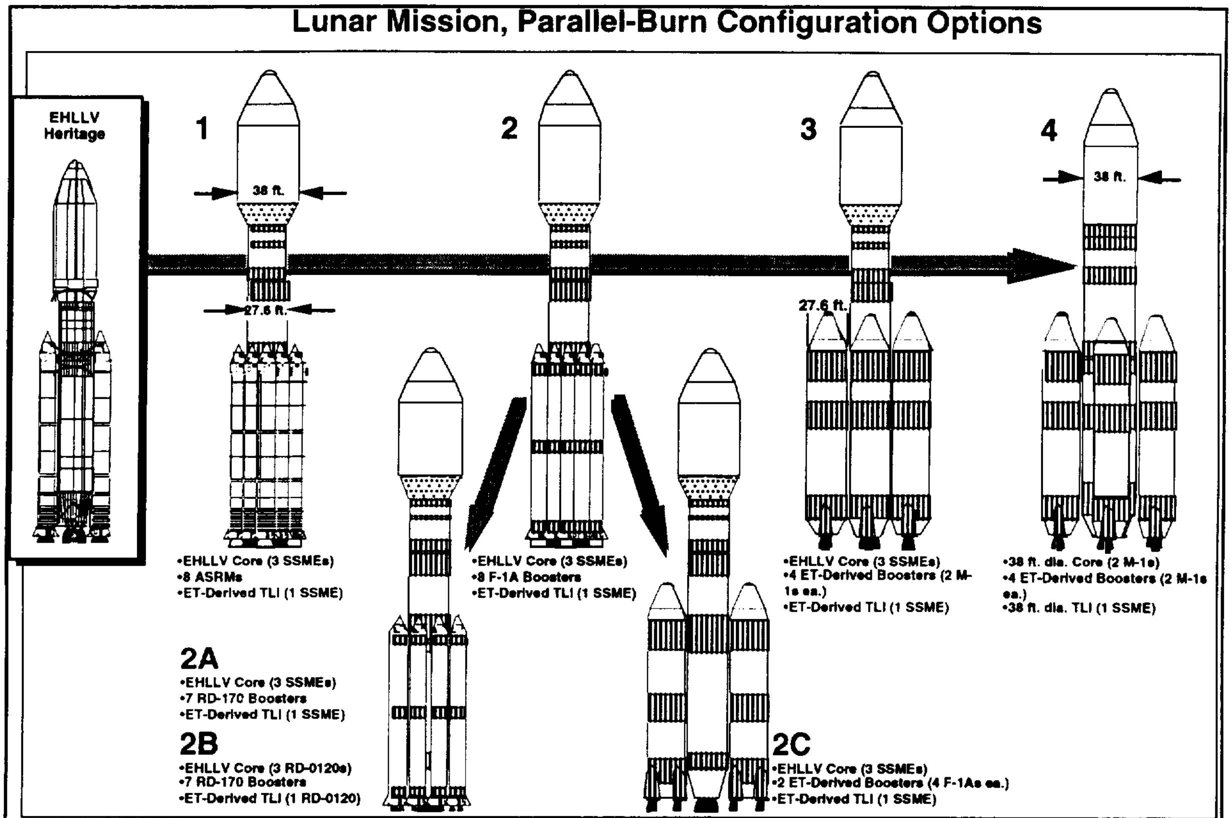
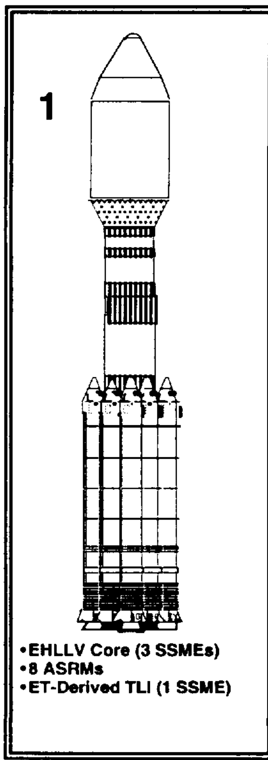


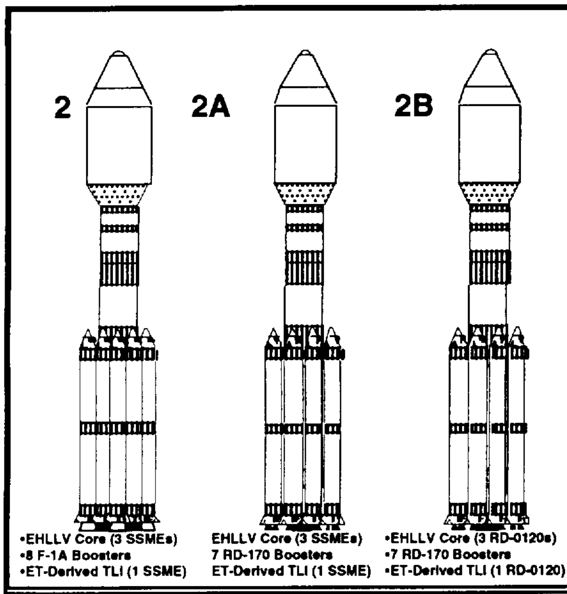
Figure 2.4-2 Candidate Parallel-Burn EHLLV-Derived Configurations



Pros:
- No new engines
- Booster simplicity
- Low DDT&E

Cons:
- Cannot lift 93 MT
- Booster stacking complexity
- Environmental issues
- Acceleration loads vs. core vehicle beef-up
- Separation dynamics
- Vehicle & ground load path complexity
- Mars evolution questionable
- Hammerhead vs. inert wt. hit

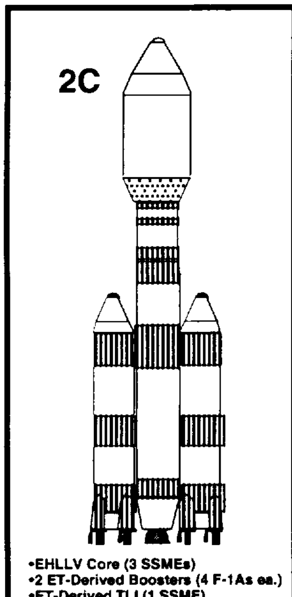
Figure 2.4-3 Pros/Cons of Parallel-Burn Configuration Using ASRM Strap-Ons



Pros:
- Core commonality
- Booster design simplicity
- Booster test & check-out simplicity
- Booster unit cost savings (larger lot buy)
- ELV family evolution from booster

Cons:
- New RP boosters engine
- Hammerhead vs. inert wt. hit
- Booster stacking
- Programmatic risk of CIS engines
- QA/QC uncertainties of CIS engines
- Mars evolution questionable
- Vehicle & ground load path complexity
- High FMEA/CIL count
- FRF feasibility questionable

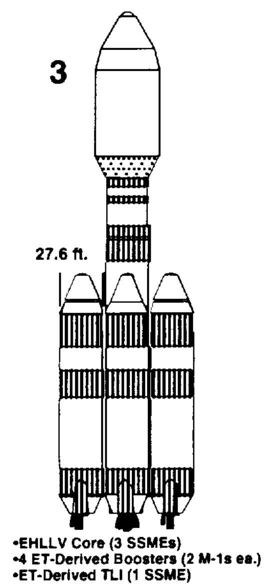
Figure 2.4-4 Pros/Cons of Parallel-Burn Configuration Using Single-Engine LRB Strap-Ons



Pros:
- Booster commonality with core
- Booster stand-alone ELV
- Mars evolution potential
- Fewer element interfaces
- Simple vehicle & ground load paths
- Fewer vehicle FMEAs/CILs

Cons:
- Booster MPS feed complexity
- Potentially higher booster unit cost (smaller lot buy)
- Hammerhead vs. core inert wt. hit

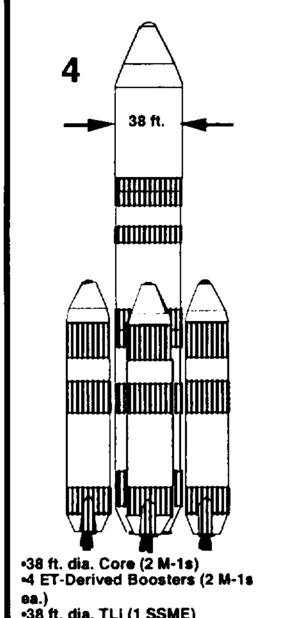
Figure 2.4-5 Pros/Cons of Parallel-Burn Configuration Using Two Multi-Engine LRB Strap-Ons



Pros:
- Booster commonality with core
- Common booster/core propellants
- Booster stand-alone ELV
- Mars evolution potential
- Fewer element interfaces
- Fewer vehicle FMEAs/CILs
- No environmental impacts
- Potentially lower booster unit cost (larger lot buy)

Cons:
- New booster engine
- Booster leak potential
- Lower booster density impulse
- Hammerhead vs. core inert wt. hit
- Vehicle & ground load path complexity

Figure 2.4-6 Pros/Cons of Parallel-Burn Configuration Using Four Multi-Engine LRB Strap-Ons



Pros:
- No hammerhead
- Common booster/core propellants
- Booster stand-alone ELV
- Mars evolution potential
- Fewer element interfaces
- Fewer vehicle FMEAs/CILs
- No environmental impacts
- Potentially lower booster unit cost (larger lot buy)

Cons:
- New core vs. booster commonality with ET
- New booster/core engine
- Booster leak potential
- Lower booster density impulse
- Vehicle & ground load path complexity

Figure 2.4-7 Pros/Cons of Parallel-Burn Configuration Using Four Multi-Engine LRB Strap-Ons with Large-Diameter Core

Figure 2.4-8 shows the resulting candidate series-burn EHLLV-derived lunar mission launch vehicle concepts that were identified and assessed. The large arrow in the figure indicates the path of design evolution from the basic EHLLV core vehicle, with the evolution shown in terms of least relative DDT&E funding requirements to highest relative DDT&E requirements. Figures 2.4-9 through 13 summarize the qualitative pros and cons of the respective series-burn vehicle types.

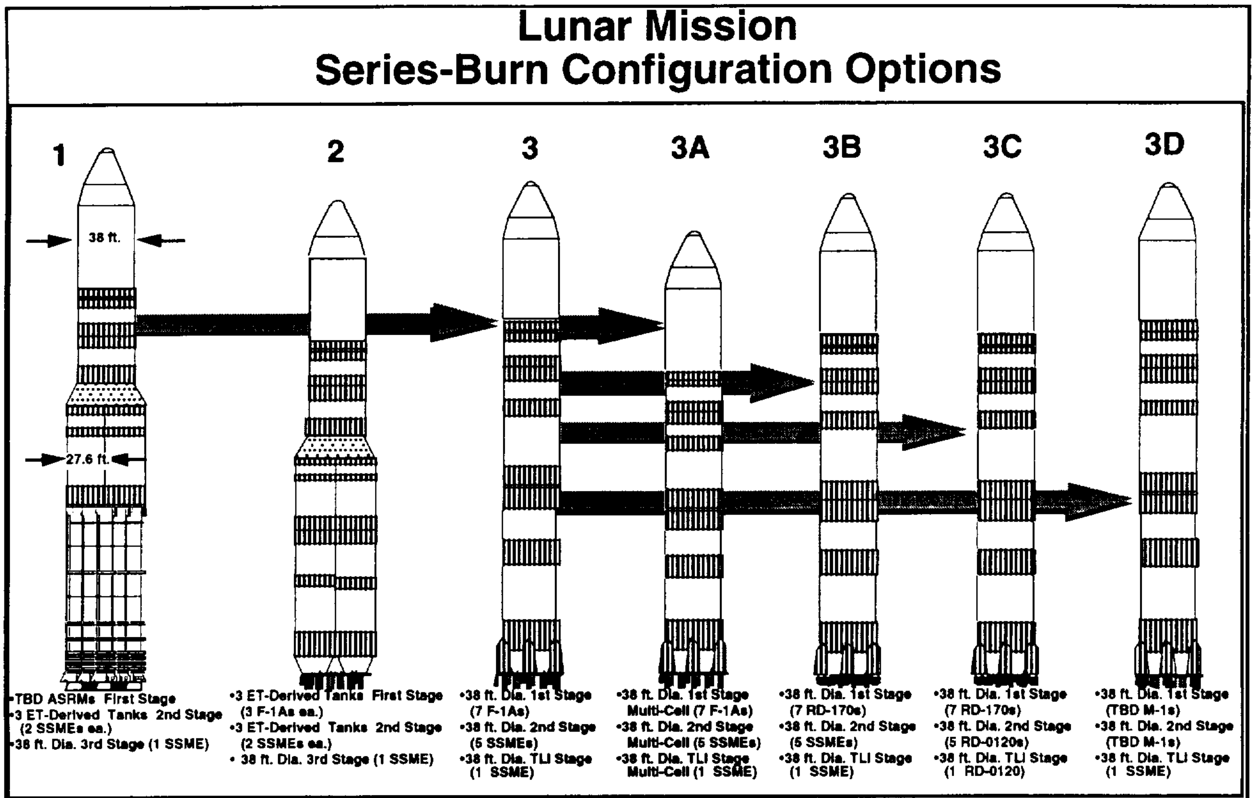
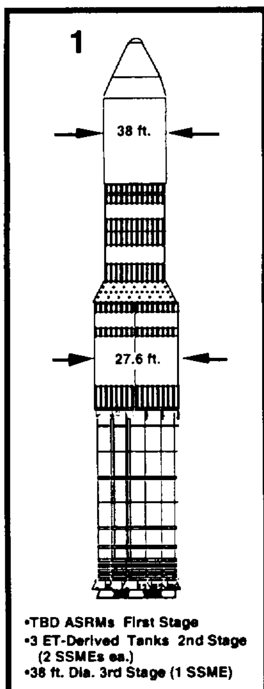


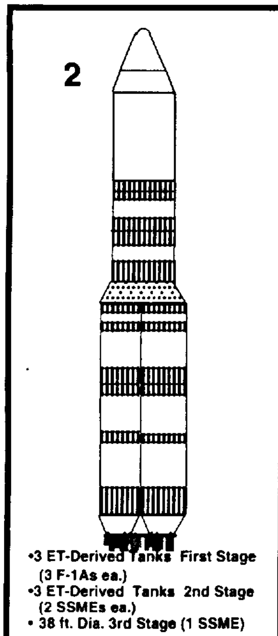
Figure 2.4-8 Candidate Series-Burn HLLV Configurations



Pros:
- No hammerhead
- No new engines
- Booster simplicity
- Existing stage elements vs. attach hardware development

Cons:
- New TLI tank design
- Stage stacking complexity
- Interstage complexity
- TLI stage tank complexity (probably multiple tanks)
- Environmental issues
- Acceleration loads vs. core vehicle beef-up
- Vehicle & ground load path complexity
- Many SSMEs
- Mars evolution questionable

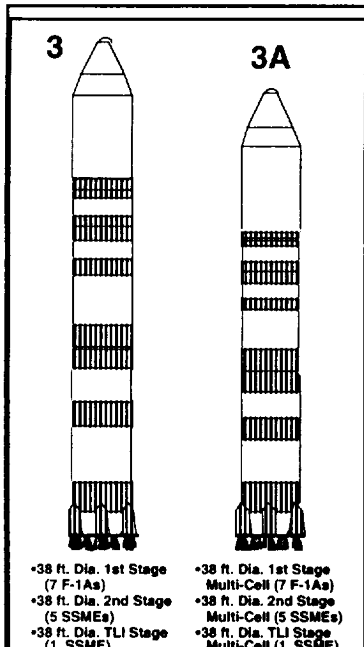
Figure 2.4-9 Pros/Cons of Series-Burn Configuration Using ASRM First Stage Cluster



Pros:
- No hammerhead
- First/Second stage tank commonality with ET
- Capable of testing each stage element independently
- Existing stage elements vs. attach hardware development
- Lessened environmental issues
- Mars evolution potential

Cons:
- New TLI tank design
- Stage stacking complexity
- Interstage complexity
- TLI stage tank complexity (probably multiple tanks)
- Vehicle load path complexity
- Many SSMEs

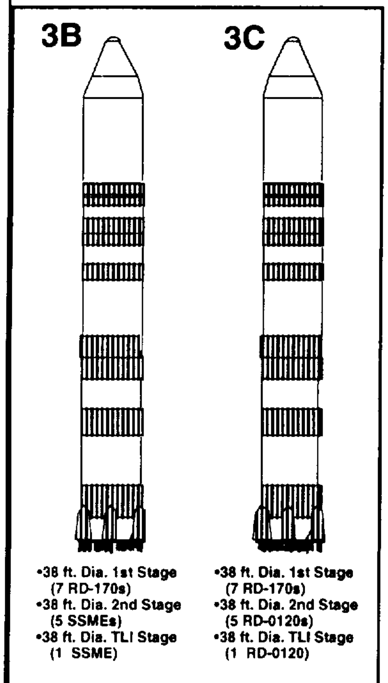
Figure 2.4-10 Pros/Cons of Series-Burn Configuration Using ET-Derived First Stage Cluster



Pros:
- No hammerhead
- Common stage diameter & tank design
- Fewer stage sub-elements & FMEAs/CILs
- Fewer engines per stage with simpler MPS
- Capable of testing each stage element independently
- Simplified load path
- Lessened environmental issues
- Mars evolution potential
- Less weld length for multi-cell
- Fewer stage pressurization subsystems
- No first stage leak issue

Cons:
- New stage design
- Many SSMEs
- Long total vehicle length (unless use multi-cell)
- TLI stage tank complexity (probably multiple tanks)
- Many SSMEs
- Complicated first/second stage MPS feed subsystems (unless use multi-cell)

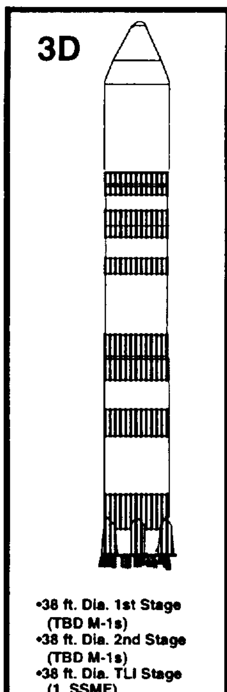
Figure 2.4-11 Pros/Cons of Series-Burn Configuration Using Constant Diameter Stages and Saturn-Derived Engines



Pros:
- Shorter first stage design (better Isp)
- Potentially cheaper engines & no development cost
- No hammerhead
- Common stage diameter & tank design
- Fewer stage sub-elements & FMEAs/CILs
- Fewer engines per stage with simpler MPS
- Capable of testing each stage element independently
- Simplified load path
- Lessened environmental issues
- Mars evolution potential
- Fewer stage pressurization subsystems
- No first stage leak issue

Cons:
- Foreign engines & RP source
- New stage design
- Long total vehicle length (unless use multi-cell)
- TLI stage tank complexity (probably multiple tanks)
- Complicated first/second stage MPS feed subsystems

Figure 2.4-12 Pros/Cons of Series-Burn Configuration Using Constant Diameter Stages and Russian Engines



Pros:
- Fewer engines for first/second stage (higher thrust)
- Simpler MPS feed subsystems
- Propellant commonality for all stages
- No hammerhead
- Common stage diameter & tank design
- Fewer stage sub-elements & FMEAs/CILs
- Fewer engines per stage with simpler MPS
- Capable of testing each stage element independently
- Simplified load path
- No environmental issues
- Mars evolution potential
- Fewer stage pressurization subsystems

Cons:
- Poor first stage density impulse (larger stage)
- Greater leak potential
- New stage design
- Long total vehicle length (unless use multi-cell)
- TLI stage tank complexity (probably multiple tanks)

Figure 2.4-13 Pros/Cons of Series-Burn Configuration Using Constant Diameter Stages and New LOX/LH_2 First Stage Engines

2.5 Alternative Propellant Tank Design Concepts

The sheer size of the lunar mission HLLV concepts, which were typically larger in height and diameter than the venerable Saturn V (365 feet and 33 feet respectively), presented both manufacturing and ground operations issues. As a result, LMMS researched methods for manufacturing large launch vehicle structures and uncovered a collection of works performed by MSFC structures personnel during the latter stages of the Apollo program that assessed innovative methods for reducing the size of, and labor associated with large launch vehicles. The efforts, which included the actual fabrication of full-scale propellant tank hardware, were the result of early Mars mission planning that had been proposed by Werner Von Braun, which involved launch vehicles that were wanting to be significantly larger than the Saturn V. Unfortunately, the majority of the documentation associated with these efforts were lost during the post-Apollo years, but LMMS obtained copies of one-of-a-kind personal copies of original engineering reports that were saved by two individuals involved with the original efforts. Two primary methods were assessed for the reduction of propellant tank length and diameter: semi-toroidal propellant tanks, and multi-cell propellant tanks.

Semi-Toroidal Propellant Tanks

The use of semi-toroidal and toroidal propellant tanks is not unique, nor new; they have been used for exotic classified upper stage applications, and have been used, and continue to be used very successfully, by the Russians. Semi-toroidal tanks, which are toroids with barrel sections added between the end caps to provide additional propellant volume (via length), produce large length reductions over conventional non-nested propellant tank cylinders (up to 50 ft. reductions for large vehicles), reduce the "stowed volume" within a given diameter than conventional designs, and provide higher volumetric efficiency. They also require a center structural post to allow thrust structure to help support the tanks for accelerational loads during ascent.

Multi-Cell Propellant Tanks

The studies performed by the MSFC structures personnel determined through analysis and production demonstration units that the multi-cell design concept was the most promising method for reducing both size and production labor costs for propellant tanks that were greater than 25 feet in diameter and approaching or greater than the size of the Saturn V propellant tanks. Upon reviewing the MSFC documentation, LMMS concluded that multi-cell propellant tanks could provide approximately a 10 percent dry mass reduction for ET-sized diameters, and could provide approximately a 25 percent dry mass reduction for Saturn V-sized diameter. Slosh baffles could become an integral part of the load-bearing web stiffeners instead of being purely parasitic dry mass. It was also found that if the number of propellant tank cells equaled the number of engines, the feed line complexity and propellant residuals could be dramatically reduced. From a production standpoint, use of multi-cell tank concepts could significantly reduce the total weld length over conventional designs and weld land depths could be up to one third less than that required for conventional tanks.

Figure 2.5-1 illustrates the relative size and weight comparison between conventional, semi-toroidal, and multi-cell propellant tank concepts for a typical lunar mission class HLLV. Figure 2.5-2 provides a detailed stage component weight comparison between the three concepts. The multi-cell concept was the clear winner, both in structural dry weight and linear weld land distance for Saturn V class (and larger) HLLVs. The additional benefit of the multi-cell internal structural webs doubling as slosh baffles made the design trade even more obvious. Figure 2.5-3 illustrates the three types of propellant tank design concepts.

Vehicle Comparison		
Tank Type	Structural Weight (lbm)	Length (ft)
Conventional	322,400	306
Toroidal	317,700	256
Multi-Cell	247,200	273

Figure 2.5-1 Comparison of Alternative Propellant Tank Designs

WEIGHT (lbf) COMPARISON			
Structure	Conventional	Semitoroidal	Multicell
Forward Skirt	29,000	19,000	15,000
Intertank Skirt	52,000	33,000	36,000
Fuel Tank	32,000	45,000	37,600
Lox Tank	67,400	75,700	63,600
Aft Skirt and Thrust Structure	142,000	145,000	95,000
Total	322,400	317,700	247,200

Figure 2.5-2 Detailed Comparison of Alternative Propellant Tank Designs

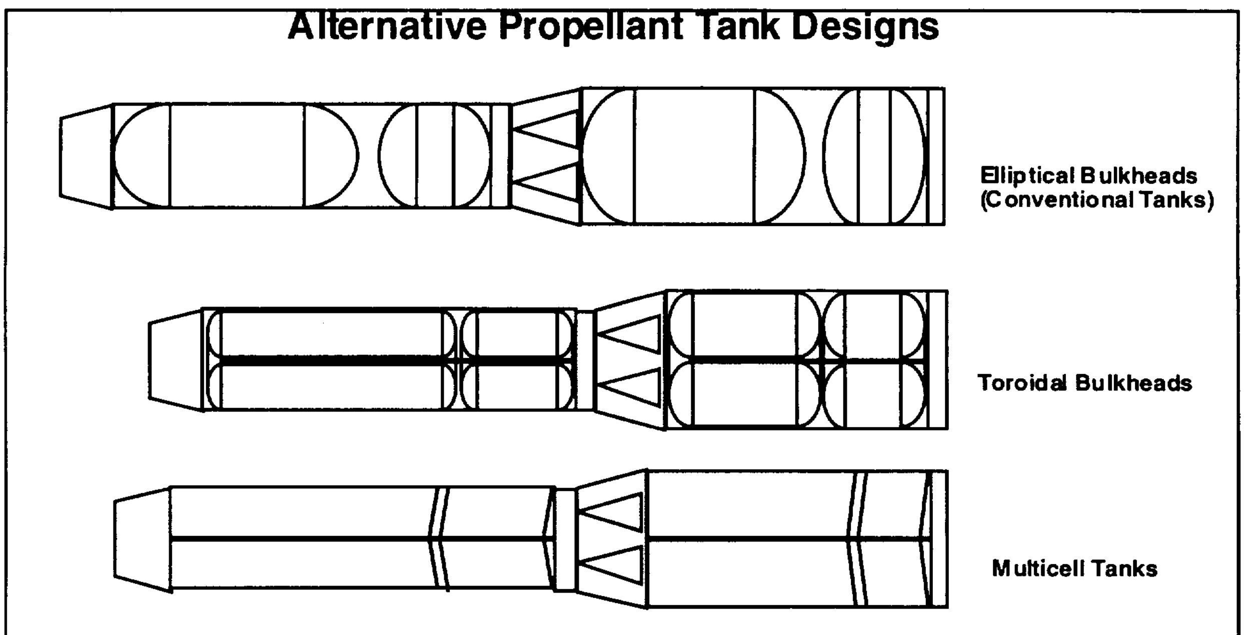


Figure 2.5-3 Alternative Propellant Tank Designs

2.6 Launch System Health Management System Requirements

During the latter stages of the HLLV concept assessment effort, Mr. Steven Black of Lockheed Martin Space Operations was tasked with defining a comprehensive set of launch system health management requirements that tied ground operations requirements with those of the launch vehicle. Mr. Black applied his uniquely extensive knowledge of both the Space Shuttle vehicle and ground operations subsystem functions and associated hardware and software, along with information provided from Lockheed Martin Sanders' state-of-the-art electronics/fault-diagnostics hardware experience to prepare a sixty page system health management requirements document that generically applied to any new liquid propellant launch system. A copy of this document is provided in Section 11 of Volume II.

VHM requirements were divided into the following four categories and further analyzed: 1) methodology; 2) vehicle management; 3) ground management, and; 4) information systems. These VHM categories were described through the phases of engineering development, component manufacturing and acceptance testing, vehicle manufacture/buildup and acceptance testing, launch site integration and launch commit, and mission/post-mission operations. Figure 2.6-1 illustrates at an overview level the system health management architecture.

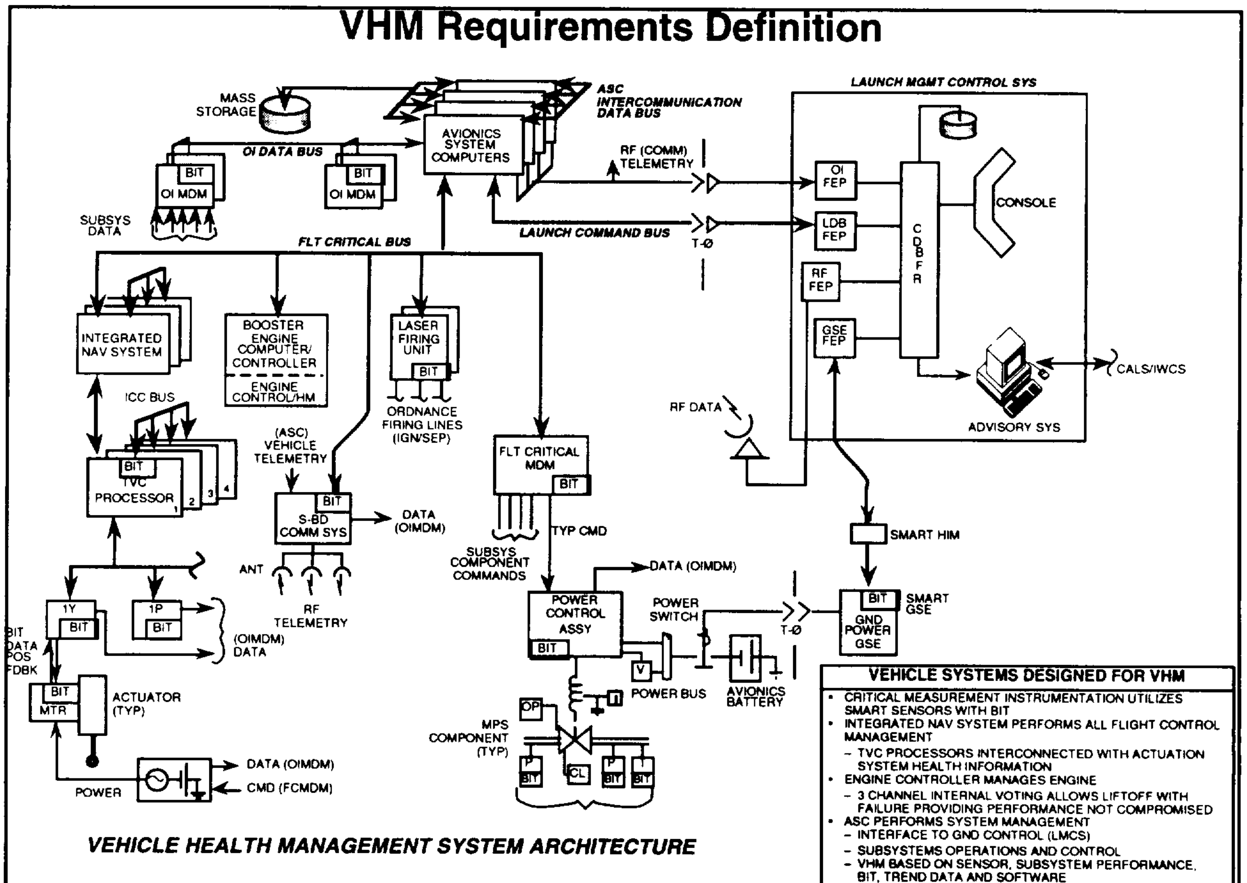


Figure 2.6-1 HLLV System Health Management Architecture

2.7 HLLV Engine Arrangements

One of the more complicated aspects of defining candidate HLLV configurations was the determination of the best first stage main engine orientations that minimized the boattail area to minimize base drag, maximized attitude control authority for both nominal and engine-out scenarios, minimized radiative and convective plume heating hot-spots, and provided sufficient gimbal angle deflection clearance when protecting for nozzle hard-over failure scenarios. Use of a computer aided design (CAD) tool allowed relatively quick and highly accurate assessments of various engine and strap-on layouts. A forty-page drawing package was developed using AutoCad, and provided to the TA-2 COTR, that contained alternate engine and gimbal arrangements for thirteen candidate lunar HLLV concepts. The HLLV concepts were grouped into parallel and series burn vehicle configurations. The drawings presented alternate methods of arranging the HLLV engines and actuators to minimize both total boattail area and engine gimbal overlap (assuming 5-8 degree gimbaling). Booster-to-booster and booster-to-core clearances were also calculated in the drawings to address the concerns of launch vehicle accessibility during preflight assembly. Figure 2.7-1 provides an example of the CAD drawings that were developed to establish engine clearances.

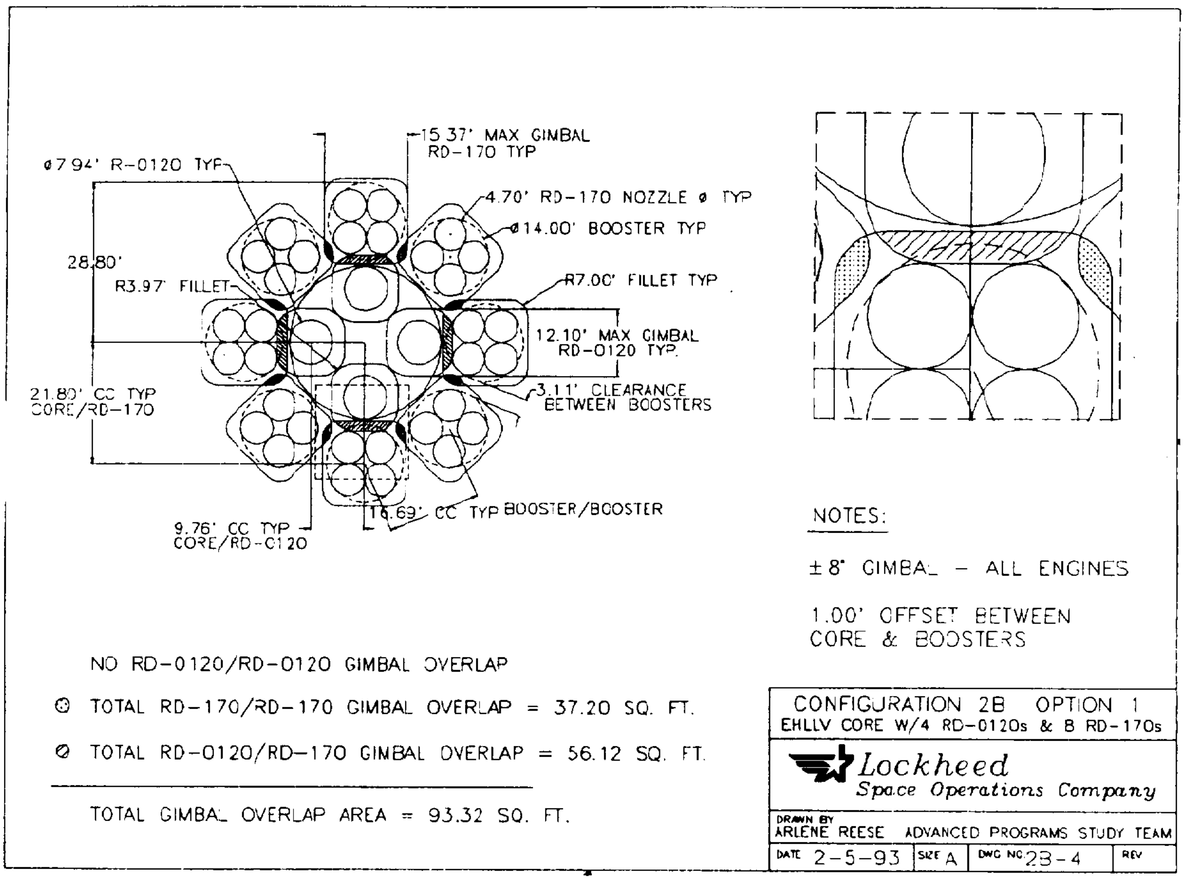


Figure 2.7-1 Typical HLLV Configuration Layout and Gimbal Angle Assessment

2.8 Super Red Team Support

During January and February of 1993, NASA Headquarters chartered an assessment of the requirements for a rapid assembly of Space Station Freedom, using HLLVs to augment the Space Shuttle assembly flights. LMSO was tasked under TA-2 by NASA KSC to assess ground operations scenarios for various mixed-fleet architectures. The HLLV concept that was utilized in the assessments was an EHLLV-derived core vehicle and either eight single-F-1A strap-on boosters or seven single-RD-170 strap-on boosters.

3.0 50-80K Vehicle Assessment

During January of 1993, the TA-2 team was tasked to define and assess in-line, two-stage ELVs that had an injected payload capability to LEO of 50-80 Klbm. The range of payload masses represented candidate concepts for the resupply of Space Station Freedom (SSF) pressurized and unpressurized logistics modules, as well as uncrewed cargo return vehicles (CRVs) and SSF crew transfer vehicles (known as the Personnel Launch System; PLS). The connection with TA-2's charter was that the stage elements were to be designed such that they could also be used as strap-on boosters for a lunar HLLV, or the stage elements could be used as in-line stage elements for a lunar HLLV. Figure 3.0-1 illustrates the evolutionary philosophy for applying the 50K vehicles to HLLV configurations.

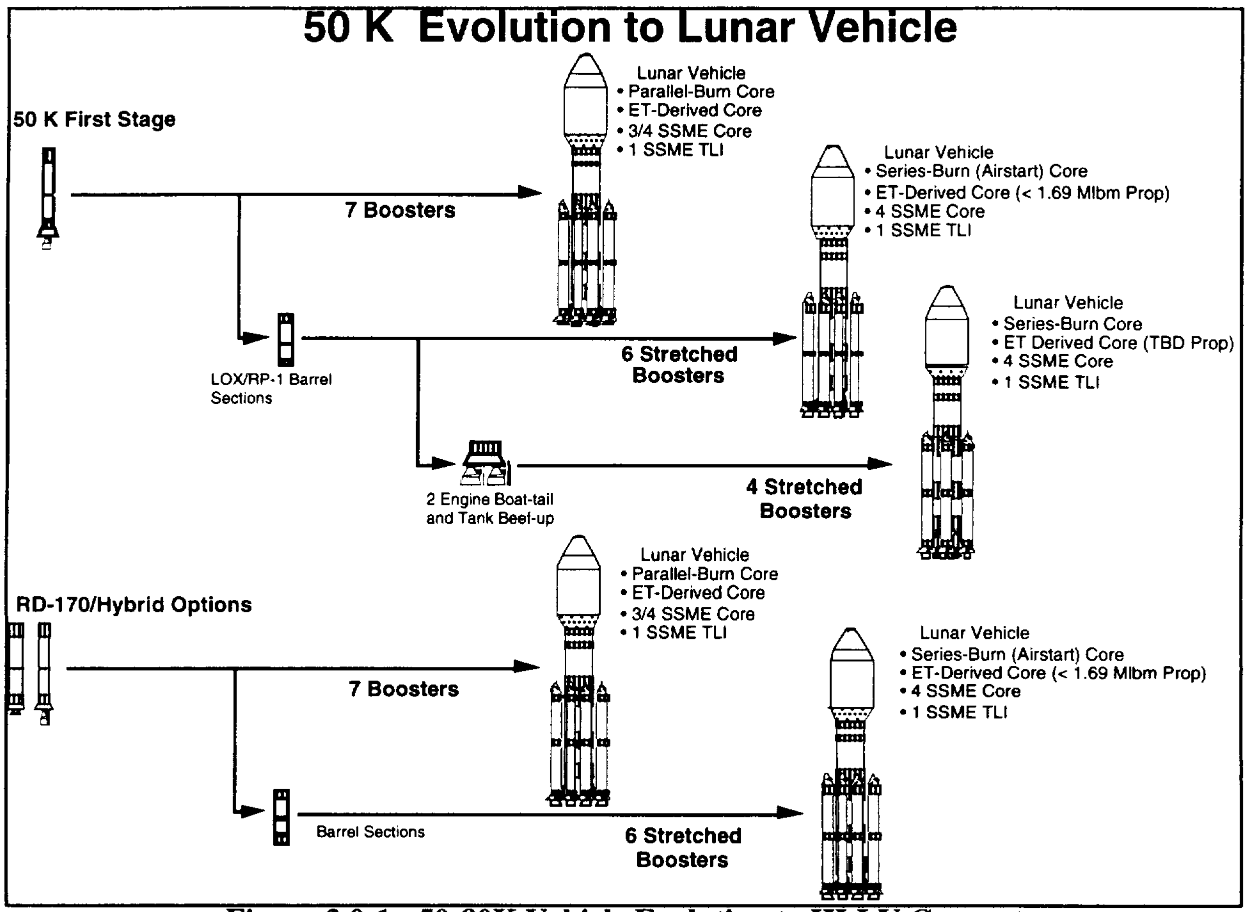


Figure 3.0-1 50-80K Vehicle Evolution to HLLV Concepts

During the April-May timeframe of 1993, the focus of the 50-80K concepts shifted from an application to HLLV to that of primary use for the resupply of the newly-defined Space Station Alpha, under the auspices of the Access to Space Option 2 Team, which was led by Uwe Hueter of the Marshall Space Flight Center. The focus of the Option 2 Team was the replacement of the Space Shuttle with a mixed fleet of ELVs that could support missions to carry cargo or crew up to and down from (via the CRV and PLS). For both of the 50-80K design efforts, a large number of different stage main propulsion concepts were assessed and an extensive array of design sensitivity trade studies were performed. The resulting vehicle assessment data were provided to both the TA-2 COTR and to the Option 2 Team.

3.1 50-80K Propulsion Matrix

Figure 3.1-1 summarizes the matrix of first and second stage main propulsion options that were assessed for the two-stage 50-80K vehicle concepts. Aerojet provided invaluable assistance in identifying candidate propulsion concepts and defining the associated performance characteristics, leveraging their extensive experience in liquid propulsion, as well as their developing experience with hybrid propulsion. Hybrid propulsion concepts were allowed for consideration due to the extended Initial Operational Capability (IOC) date being projected for the 50-80K vehicles, late 1990s to early 2000s. The hybrids provided the benefit of high density-impulse for atmospheric flight applications, as well as their inherent simplicity, as compared to bipropellant pump-fed liquid propulsion options, while still providing the capability for throttling (via mixture ratio control) and controlled shut-down (which was a man-rating safety consideration).

First Stage/Second Stage Options		
Liquid/Liquid *	**Hybrid/Liquid ***	**Solid/Liquid**
F-1A/LCSSME	Staged Combustion Hybrid/LCSSME	3 Segment ASRM/LCSSME
F-1A/J-2S	Staged Combustion Hybrid/Rubber STME	3 Segment ASRM/LCSSME
F-1A/SSME	Staged Combustion Hybrid/J-2S	2 Segment ASRM/J-2S
F-1A/RD-0120	Staged Combustion Hybrid/Vulcain	2 Segment ASRM/LCSSME
F-1A/Vulcain	Staged Combustion Hybrid/RD-0120	2 Segment ASRM/SSME
STME/LCSSME	Classical Hybrid/LCSSME	1 Segment ASRM/Centaur
STME/STME	Classical Hybrid/Rubber STME	
STME/RD-0120	Classical Hybrid/J-2S	
STME/Vulcain	Classical Hybrid/Vulcain	
M-1A/LCSSME	Classical Hybrid/RD-0120	
M-1A/RD-0120		
M-1A/Vulcain		
RD-170/LCSSME		
RD-170/J-2S		
RD-170/RD-0120		
RD-170/Vulcain		
LCSSME/LCSSME		
LCSSME/RD-0120		
LCSSME/Vulcain		

Note: * Configurations sized for 50 Klbm payload

Figure 3.1-1 50-80K Vehicle Stage Main Propulsion Options

Figure 3.1-2 summarizes the performance specifications of the candidate engines.

Engine Specifications

	M-1A	F-1A	STME	SSME (104% RPL)	RD-170
Sea Level Thrust (lbf)	1,300,000	1,800,000	551,430	390,000	1,632,000
Vacuum Thrust (lbf)	1,562,000	2,020,700	650,000	488,800	1,777,000
Sea Level Specific Impulse (sec)	344.5	269.7	364	364.8	309
Vacuum Specific Impulse (sec)	414.0	303.1	428.5	452.9	337
Chamber Pressure (psia)	1,000	1,161	2,250	3,110	3,560
Mixture Ratio	5.0	2.27	6.0	6.0	2.6
Area Ratio	20	16	45	77.5	36.87
Engine Mass (lbm)	20,200	19,000	9,974	6,990	21,510
Engine Length (ft)	19.08	18.36	13	14	13.12
Engine Diameter (ft)	12.58	11.96	12.1	8	12.20
Propellant	O2/H2	O2/RP-1	O2/H2	O2/H2	O2/Syn10

	RD-0120	Vulcain	J-2S	LCSSME (Altitude)	LCSSME (Sea Level)
Sea Level Thrust (lbf)	352,746	---	197,000	---	506,000
Vacuum Thrust (lbf)	440,925	230,000	265,000	326,600	600,000
Sea Level Specific Impulse (sec)	364	---	320	---	371
Vacuum Specific Impulse (sec)	455	431.6	436	451.9	440
Chamber Pressure (psia)	3,000	1,450	1,200	2,075	2,075
Mixture Ratio	6.0	5.2	5.5	6.0	6.0
Area Ratio	85.7	45	40	77.5	42
Engine Mass (lbm)	7,607	2,860	3,800	7,053	7,300
Engine Length (ft)	14.93	9.62	11.08	14	14
Engine Diameter (ft)	7.94	5.77	6.71	8	8
Propellant	O2/H2	O2/H2	O2/H2	O2/H2	O2/H2

Figure 3.1-2 50-80K Vehicle Stage Main Propulsion Specifications

3.2 50-80K Trade Studies

An extensive set of vehicle sizing sensitivity studies was performed for each of the first and second stage propulsion options in order to best optimize the respective launch vehicle configurations. Launch vehicle unit cost considerations were factored into the configuration sizing trades by the historically proven relationship that vehicle dry mass had a direct bearing on vehicle unit cost, for a fixed stage thrust-to-weight goal. The dry mass directly influenced the size and number of stage main engines, which dominated the stage unit cost, as well as influenced the complexity and quantity of thrust structure, primary structure, and propellant feed subsystems. Figures 3.2-1 through 9 illustrate the types of vehicle sizing sensitivity trade studies that were performed for each stage propulsion option. The resulting down-selected vehicle configurations were chosen through the best combination of results of the trade studies.

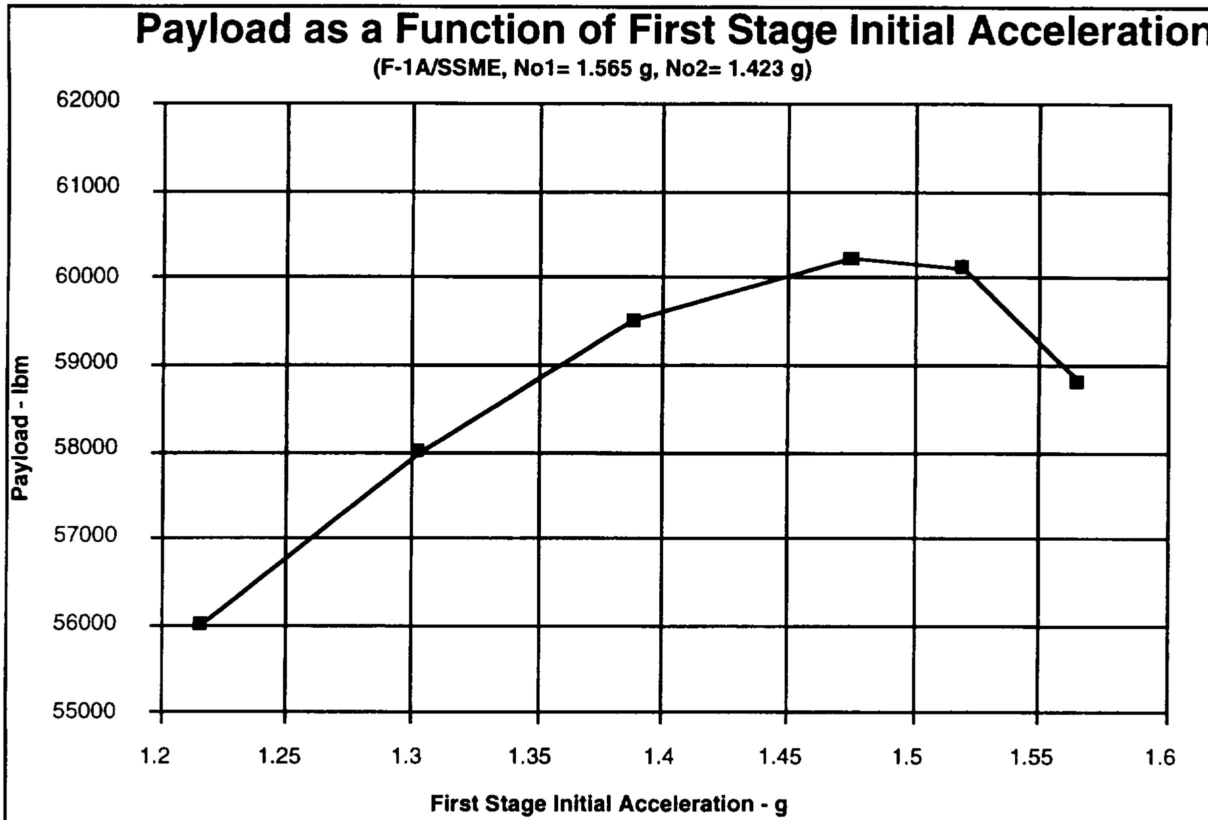


Figure 3.2-1 Payload Mass as a Function of First Stage Initial Acceleration

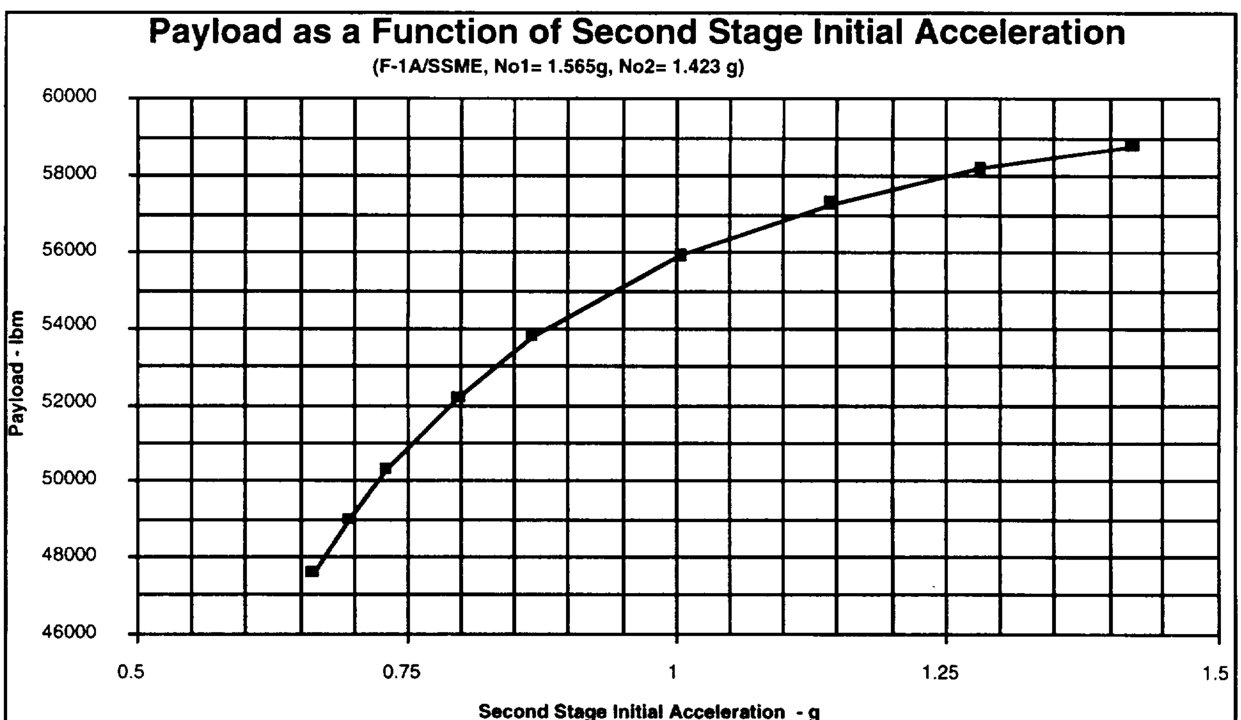


Figure 3.2-2 Payload Mass as a Function of Second Stage Initial Acceleration

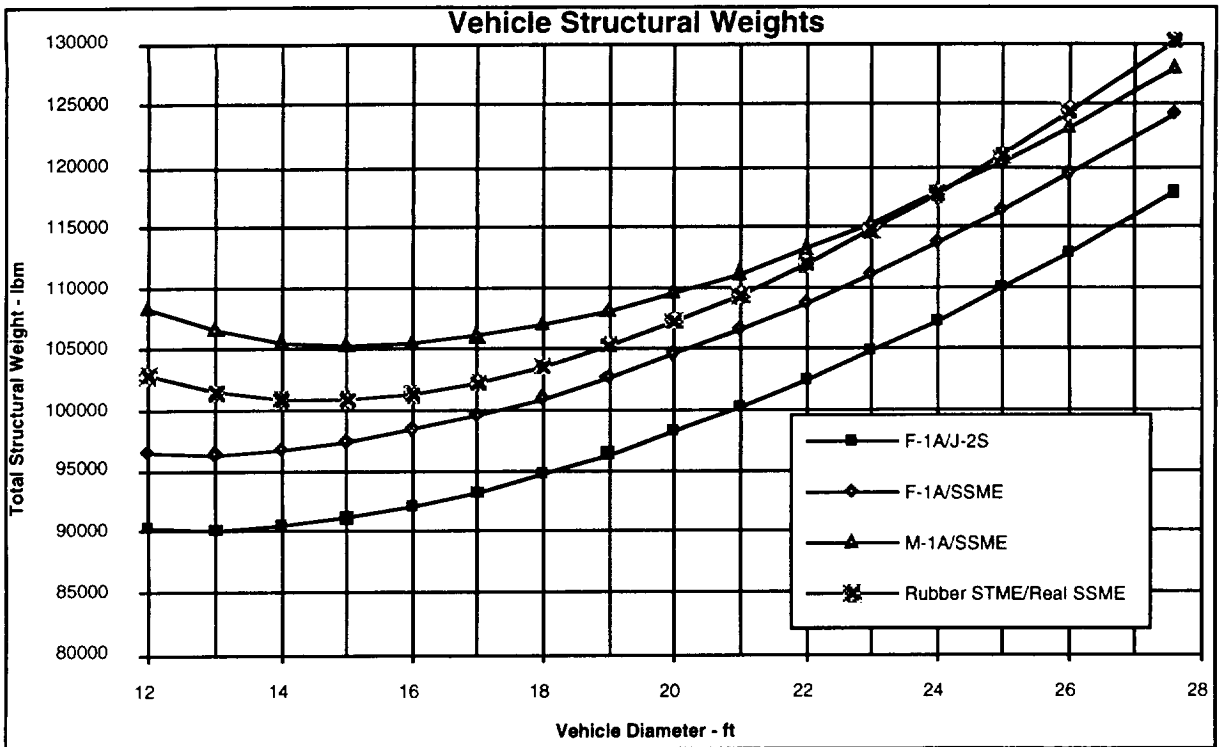


Figure 3.2-3 Total Vehicle Structural Weight as a Function of Vehicle Diameter

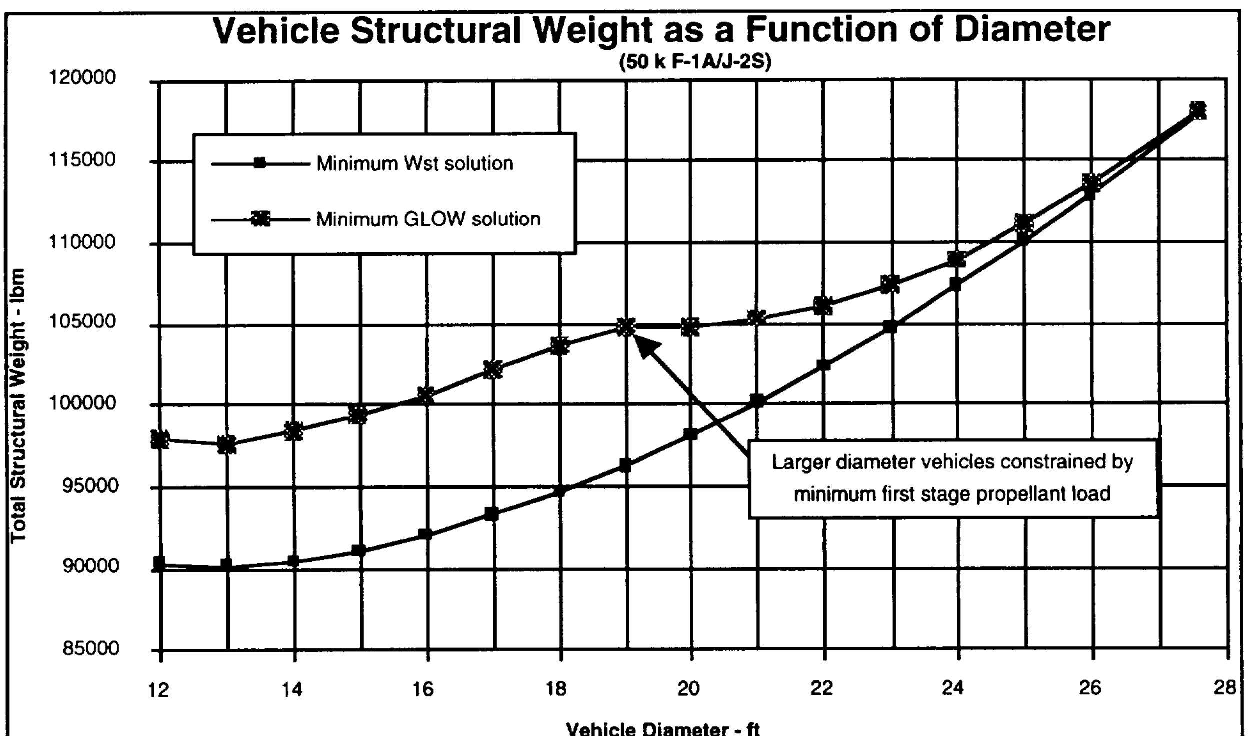


Figure 3.2-4 Total Vehicle Structural Weight Sensitivity to Minimum GLOW Solution and Minimum First Stage Structure Weight Solution

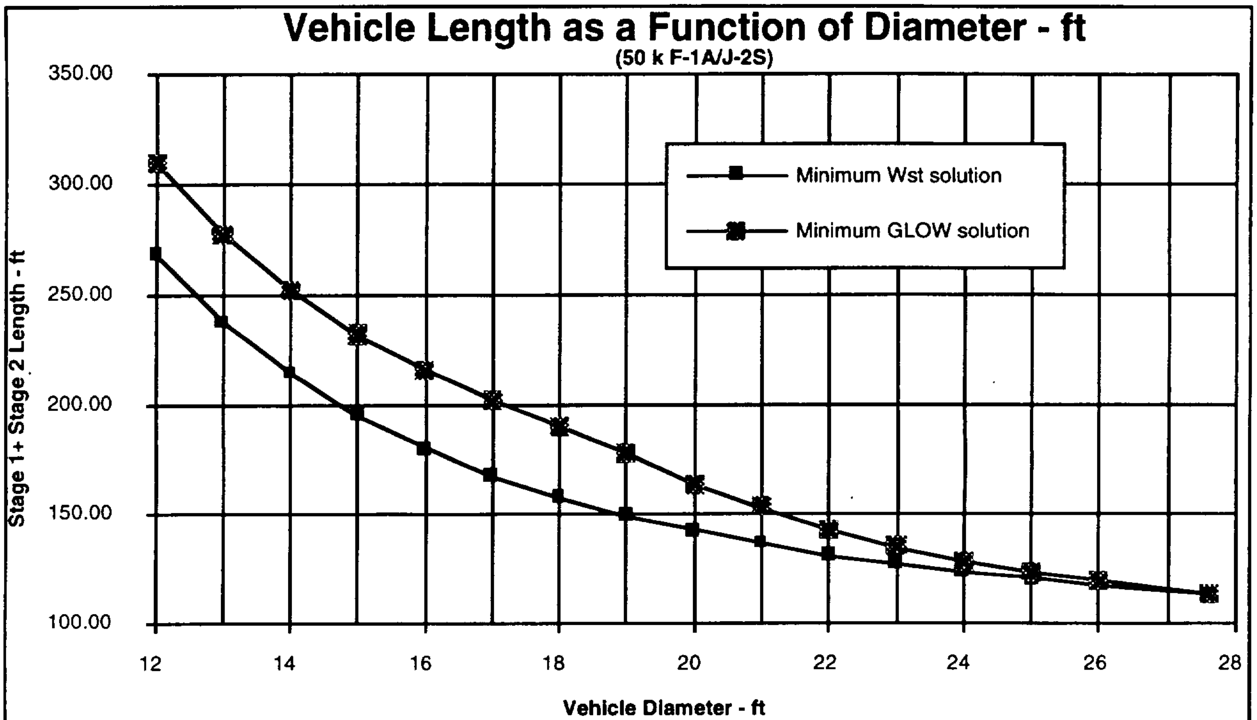


Figure 3.2-5 Sensitivity of Total Vehicle Length to Vehicle Diameter for Minimum First Stage Structural Weight Solutions and Minimum GLOW Solutions

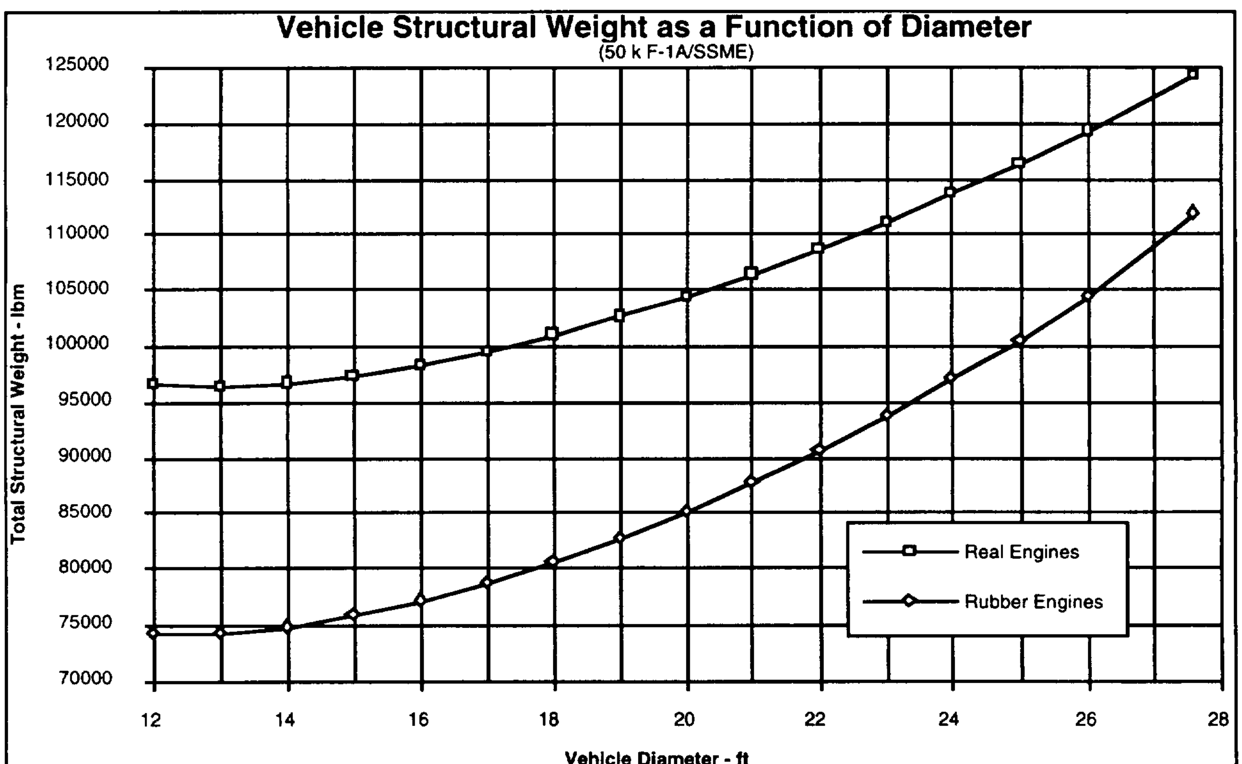


Figure 3.2-6 Sensitivity of Total Vehicle Structural Weight to Vehicle Diameter for Use of "Real" Engines Versus "Rubber" Engines

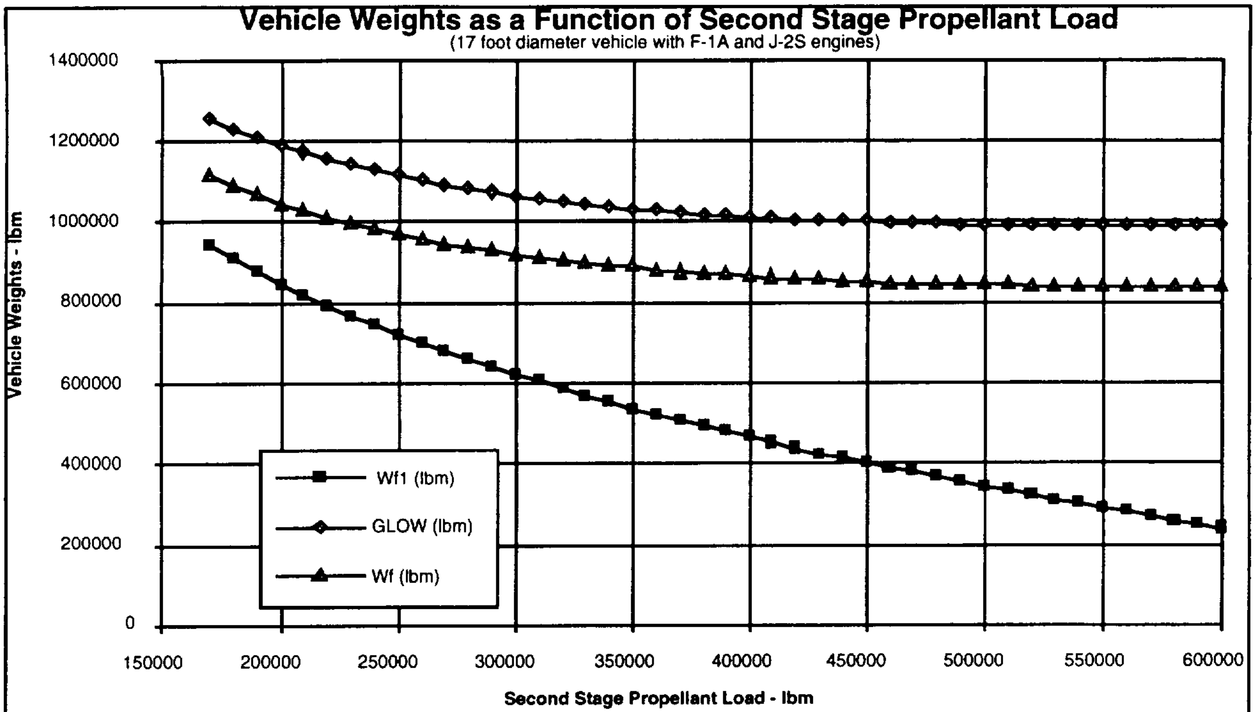


Figure 3.2-7 Sensitivity of First Stage Propellant Load, Total Propellant Load, and GLOW to Second Stage Propellant Load

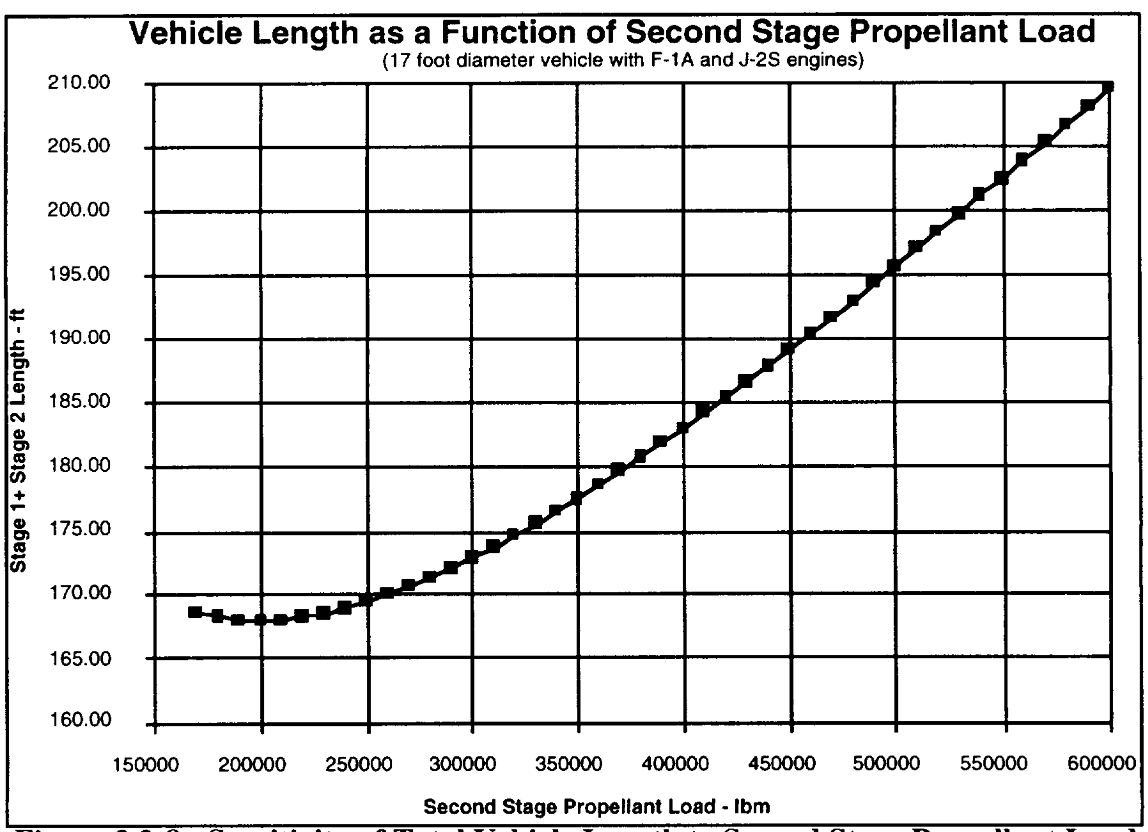


Figure 3.2-8 Sensitivity of Total Vehicle Length to Second Stage Propellant Load

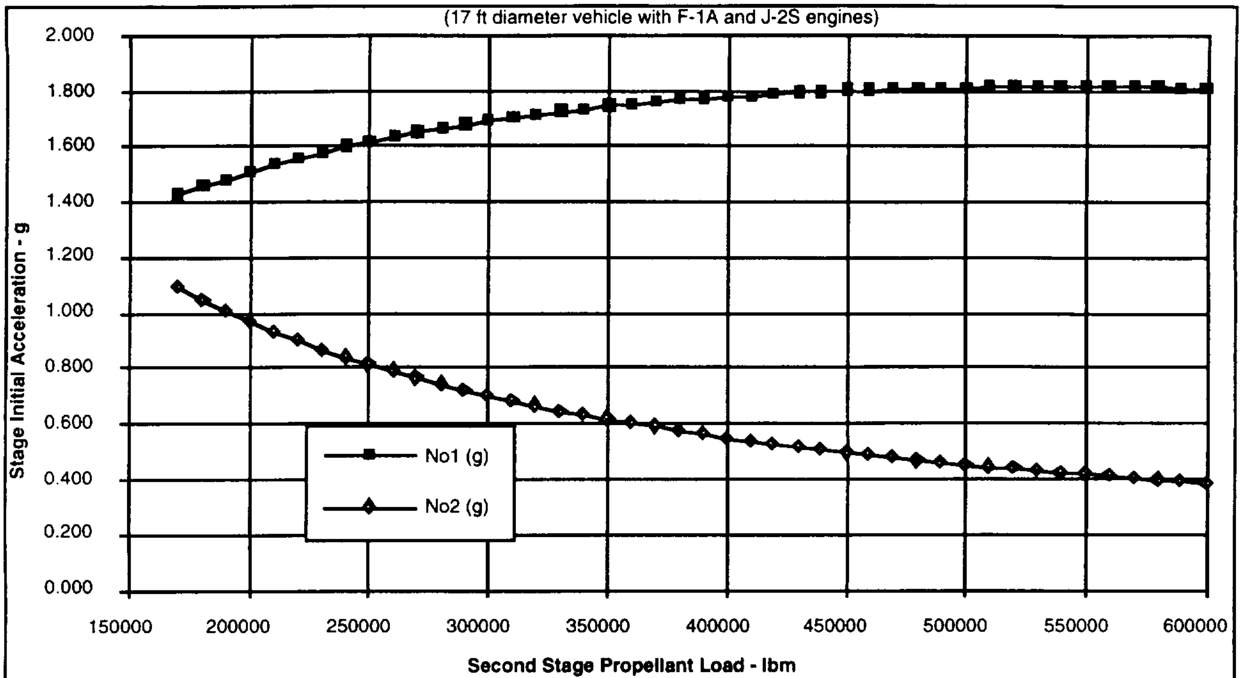


Figure 3.2-9 Sensitivity of First and Second Stage Initial Acceleration to Second Stage Propellant Load

The results of the vehicle sizing trade studies are presented in Section 3.3.

3.3 Candidate 50-80K Vehicle Configurations

Each of the candidate vehicle configurations were sized for an injected payload mass (to LEO) of 50 Klbm. Nominal 3-DOF ascent trajectory simulations were flown to confirm the adequacy of the vehicle sizing. Table 3.3-1 lists the resulting injected payload mass achieved by each configuration. The results are indicative of the high fidelity of the sizing algorithms that were utilized.

Vehicle Configuration Payload

Liquid/Liquid *	Payload (lbm) **
F-1A/LCSSME	48,249
F-1A/J-2S	54,893
F-1A/SSME	51,098
F-1A/RD-0120	48,599
F-1A/Vulcain	49,155
STME/LCSSME	48,321
STME/STME	48,034
STME/RD-0120	50,186
STME/Vulcain	49,986
M-1A/LCSSME	47,992
M-1A/RD-0120	49,471
M-1A/Vulcain	48,993
RD-170/LCSSME	49,878
RD-170/J-2S	50,166
RD-170/RD-0120	48,598
RD-170/Vulcain	50,598
LCSSME/LCSSME	48,222
LCSSME/RD-0120	49,339
LCSSME/Vulcain	49,071

Table 3.3-1 Payload Mass Confirmation via Trajectory Simulations

Hybrid/Liquid *	Payload (lbm) **
Staged Combustion Hybrid/LCSSME	51,773
Staged Combustion Hybrid/Rubber STME	54,836
Staged Combustion Hybrid/J-2S	50,610
Staged Combustion Hybrid/Vulcain	50,072
Staged Combustion Hybrid/RD-0120	51,354
Classical Hybrid/LCSSME	51,663
Classical Hybrid/Rubber STME	54,987
Classical Hybrid/J-2S	50,559
Classical Hybrid/Vulcain	49,962
Classical Hybrid/RD-0120	51,265

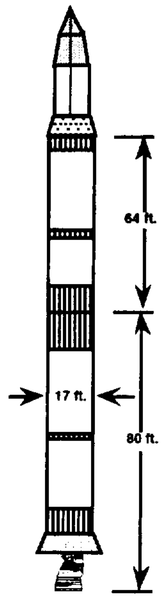
Solid/Liquid *	Payload (lbm) **
3 Segment ASRM/LCSSME	65,000
3 Segment ASRM/SSME	82,100
2 Segment ASRM/J-2S	43,600
2 Segment ASRM/LCSSME	49,300
2 Segment ASRM/SSME	56,600
1 Segment ASRM/Centaur	6,900

Note: * First Stage/Second Stage Propulsion Options
** Payloads verified by 3-DOF trajectory analysis

Table 3.3-1 Payload Mass Confirmation via Trajectory Simulations (Concluded)

Figures 3.3-1 through 5 summarize the definition of each candidate vehicle configuration.

50 k Vehicle, F-1A/ LCSSME



Length/Diameter= 8.5

Payload:	48,249 lbm (21.9 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 996,632 lbm

First Stage:

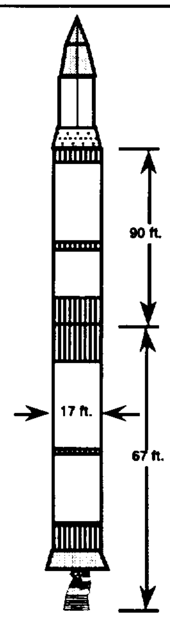
Inert Mass:	69,190 lbm
Usable Propellant:	665,000 lbm
Propellant Type:	LOX/RP-1
Engine Type/No.:	F-1A/1
Diameter:	17.0 ft
Thrust/Weight:	1.475 g
Throttle Setting:	81.8 %

Second Stage:

Inert Mass:	24,193 lbm
Usable Propellant:	190,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	LCSSME/1
Diameter:	17.0 ft
Thrust/Weight:	1.236 g

Figure 3.3-1 F-1A/Low-Cost-SSME Configuration Summary

50 k Vehicle, F-1A/ SSME



Length/Diameter= 9.2

Payload:	51,098 lbm (23.2 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 940,454 lbm

First Stage:

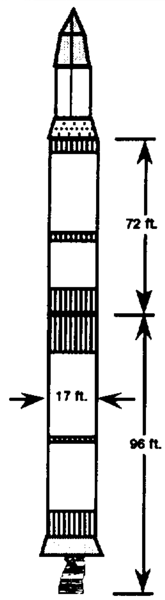
Inert Mass:	64,841 lbm
Usable Propellant:	478,000 lbm
Propellant Type:	LOX/RP-1
Engine Type/No.:	F-1A/1
Diameter:	17.0 ft
Thrust/Weight:	1.475 g
Throttle Setting:	77.0 %

Second Stage:

Inert Mass:	31,515 lbm
Usable Propellant:	315,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	SSME/1
Diameter:	17.0 ft
Thrust/Weight:	1.233 g

Figure 3.3-2 F-1A/SSME Configuration Summary

50 k Vehicle, F-1A/ J-2S



Length/Diameter= 9.9

Payload:	54,893 lbm (24.9 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,177,785 lbm

First Stage:

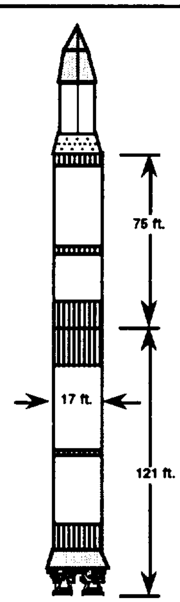
Inert Mass:	68,458 lbm
Usable Propellant:	819,635 lbm
Propellant Type:	LOX/RP-1
Engine Type/No.:	F-1A/1
Diameter:	17.0 ft
Thrust/Weight:	1.475 g
Throttle Setting:	96.1 %

Second Stage:

Inert Mass:	24,881 lbm
Usable Propellant:	210,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	J-2S/1
Diameter:	17.0 ft
Thrust/Weight:	0.930 g

Figure 3.3-3 F-1A/J-2S Configuration Summary

50 k Vehicle, STME/ STME



Length/Diameter= 11.5

Payload:	48,034 lbm (21.8 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 865,286 lbm

First Stage:

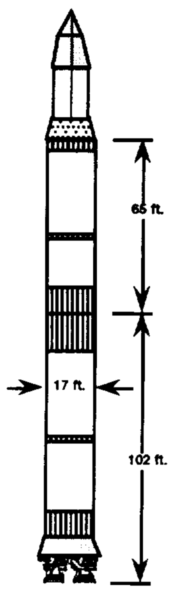
Inert Mass:	73,559 lbm
Usable Propellant:	460,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	STME/2
Diameter:	17.0 ft
Thrust/Weight:	1.272 g
Throttle Setting:	100.0 %

Second Stage:

Inert Mass:	33,693 lbm
Usable Propellant:	250,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	STME/1
Diameter:	17.0 ft
Thrust/Weight:	1.422 g
Throttle Setting	73.0 %

Figure 3.3-4 STME/STME Configuration Summary

50 k Vehicle, LCSSME/ LCSSME



Payload:	48,222 lbm (21.9 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW:	692,379 lbm

First Stage:

Inert Mass:	59,920 lbm
Usable Propellant:	365,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	LCSSME/2
Diameter:	17.0 ft
Thrust/Weight:	1.458 g
Throttle Setting:	100.0 %

Second Stage:

Inert Mass:	24,237 lbm
Usable Propellant:	195,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	LCSSME/1
Diameter:	17.0 ft
Thrust/Weight:	1.213 g

Figure 3.3-5 Low-Cost-SSME/Low-Cost-SSME Configuration Summary

There are two primary types of hybrid rocket motors that have been defined in recent years: a "classical" hybrid motor, and a staged combustion hybrid motor. The classical hybrid utilizes a solid fuel grain that does not contain any oxidizer. The oxidizer, typically LOX, is injected at the top of the solid fuel grain to support combustion. A technological draw-back of the classical hybrid motor is that uneven combustion and grain-stress can result from the injection of LOX directly onto the fuel grain. The staged combustion hybrid motor utilizes a solid propellant grain that contains a slight amount of solid oxidizer, usually ammonium-perchlorate, that provides the initiation of a fuel-rich combustion process. LOX is then injected down-stream of the fuel-rich combustion process, completing combustion at the desired mixture ratio. TA-2 propulsion partner Aerojet was tasked with defining candidate hybrid motor concepts for the 50-80K vehicles, when given the relative total first stage impulse requirement for a particular second stage propulsion candidate. Due to the sizing similarity between a classical hybrid motor and a staged combustion hybrid motor, Aerojet supplied motor data using the classical design for each of the second stage propulsion options. Aerojet also provided, for the sake of comparison, a staged combustion hybrid motor concept for use with a "rubber" STME second stage (i.e., the STME thrust was sized to meet the optimum second stage thrust-to-weight ratio). Figure 3.3-6 summarizes the groundrules and assumptions that were used to size the hybrid motor concepts.

Hybrid Booster Concepts

Performance and Sizing Groundrules and Assumptions

- **Staged Combustion Concepts Only**
  - -- Oxidizer Type: Ammonium Perchlorate (AP)
  - -- Fuel Grain Oxidizer Content: overall (Booster) MR is 2.5; the 2.5 is split 1.9 as LOX and the remaining 0.6 is AP within the grain

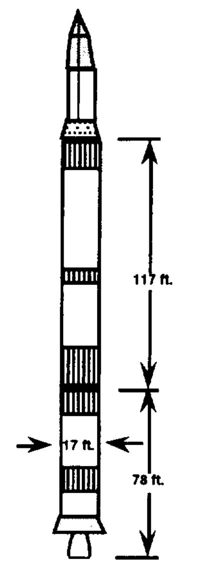
- **Staged Combustion and Classical Concepts:**
  - -- Grain Ignition Method: redundant, forward end, tri-ethyl aluminum
  - -- Igniter Weight: 50 lbf
  - -- Thrust Termination Method: termination of LOX flow
  - -- LOX Tank Pressurization Method: autogenous, warm GOX, turbine exhaust
  - -- LOX Tank Ullage Pressure: 60 psia
  - -- Motor Chamber Pressure: 1700 psia
  - -- LOX Injector Inlet Pressure: 2000 psia
  - -- Thrust Chamber Cooling Method: regenerative
  - -- TVC Method: electro-mechanical actuators
  - -- Minimum Throttle Setting: 75 % Rated Power Level

- **Residuals**
  - -- 2 % of solid propellant load for staged combustion
  - -- 10 % of solid propellant load for classical combustion

Figure 3.3-6 Hybrid Motor Sizing Groundrules and Assumptions

Figures 3.3-7 through 10 summarize the characteristics of the candidate hybrid motor vehicle concepts.

50 k Vehicle, Staged Combustion Hybrid/Rubber STME



Payload:	54,836 lbm (24.9 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,302,605 lbm

***First Stage*:**

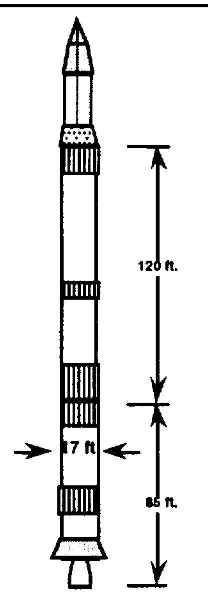
Inert Mass:	95,402 lbm
Usable Propellant:	620,000 lbm
Propellant Type:	LOX/PEBC
Engine Type/No.:	Staged Combustion Hybrid/1
Diameter:	17.0 ft
Thrust/Weight:	1.443 g
Sea Level Thrust	1,800,000 lbf
Throttle Setting	100.0 %

***Second Stage*:**

Inert Mass:	36,367 lbm
Usable Propellant:	446,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	Rubber STME/1
Diameter:	17.0 ft
Vacuum Thrust	425,894 lbf
Thrust/Weight:	0.800 g

Figure 3.3-7 Staged-Combustion-Hybrid/Rubber-STME Configuration Summary

50 k Vehicle, Classical Hybrid/Rubber STME



Payload:	53,733 lbm (24.4 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,314,534 lbm

***First Stage*:**

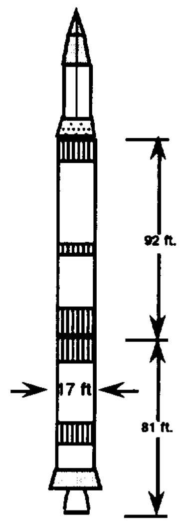
Inert Mass:	115,675 lbm
Usable Propellant:	650,000 lbm
Propellant Type:	LOX/HTDP
Engine Type/No.:	Classical Hybrid/1
Diameter:	17.0 ft
Thrust/Weight:	1.386 g
Sea Level Thrust	1,800,000 lbf
Throttle Setting	100.0 %

***Second Stage*:**

Inert Mass:	37,126 lbm
Usable Propellant:	458,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	Rubber STME/1
Diameter:	17.0 ft
Vacuum Thrust	436,101 lbf
Thrust/Weight:	0.800 g

Figure 3.3-8 Classical-Hybrid/Rubber-STME Configuration Summary

50 k Vehicle, Classical Hybrid/LCSSME



Length/Diameter= 10.1

Payload:	50, 499 lbm (22.9 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,114,359 lbm

First Stage:

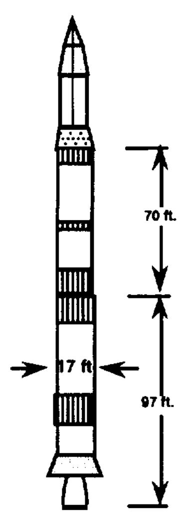
Inert Mass:	108,583 lbm
Usable Propellant:	600,000 lbm
Propellant Type:	LOX/HTDP
Engine Type/No.:	Classical Hybrid/1
Diameter:	17.0 ft
Thrust/Weight:	1.475 g
Sea Level Thrust	1,630,000 lbf
Throttle Setting	100.0 %

Second Stage:

Inert Mass:	30,277 lbm
Usable Propellant:	325,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	LCSSME/1
Diameter:	17.0 ft
Thrust/Weight:	0.806 g

Figure 3.3-9 Classical-Hybrid/Low-Cost-SSME Configuration Summary

50 k Vehicle, Classical Hybrid/Vulcain



Length/Diameter= 9.8

Payload:	47,887 lbm (21.7 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,207,921 lbm

First Stage:

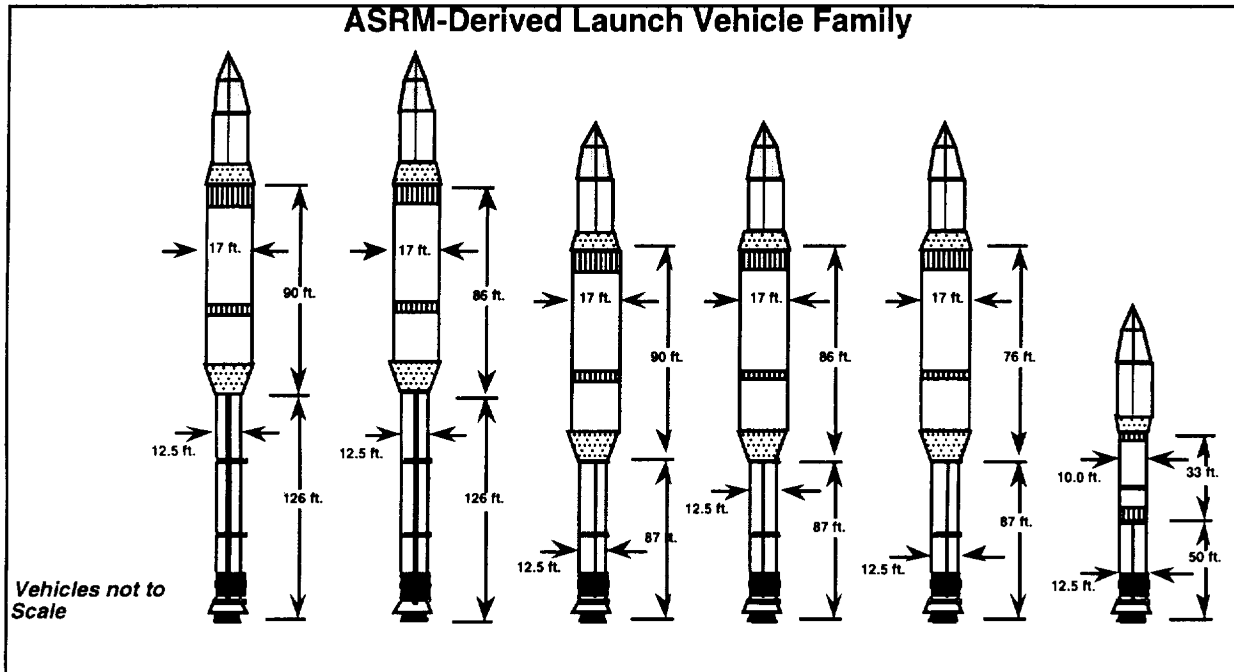
Inert Mass:	129,623 lbm
Usable Propellant:	790,000 lbm
Propellant Type:	LOX/HTDP
Engine Type/No.:	Classical Hybrid/1
Diameter:	17.0 ft
Thrust/Weight:	1.475 g
Sea Level Thrust	1,785,000 lbf
Throttle Setting	100.0 %

Second Stage:

Inert Mass:	20,411 lbm
Usable Propellant:	220,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	Vulcain/1
Diameter:	17.0 ft
Thrust/Weight:	0.792 g

Figure 3.3-10 Classical-Hybrid/Vulcain Configuration Summary

An additional family of candidate 50-80K vehicle concepts was defined that utilized partial-segment versions of the Space Shuttle's Advanced Solid Rocket Motor (ASRM), as a way to minimize stage development costs while providing high density-impulse during atmospheric flight. Figure 3.3-11 summarizes the ASRM-derived family of concepts. Figures 3.3-12 and 13 summarize the characteristics of two of the candidate configurations.

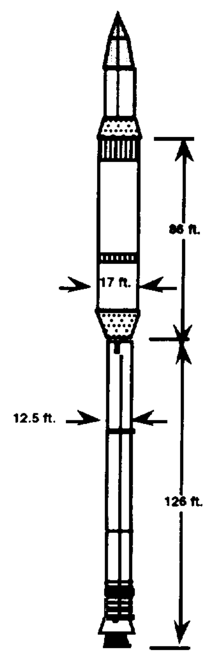


1 st Stage	3 Seg ASRM	3 Seg ASRM	2 Seg ASRM	2 Seg ASRM	2 Seg ASRM	1 Seg ASRM
2 nd Stage	1 SSME	1 LCSSME	1 SSME	1 LCSSME	1 J-2s	Centaur
Payload*	82,100 lbm	65,000 lbm	56,600 lbm	49,300 lbm	43,600 lbm	6,900 lbm

Note:
• Direct insertion into 15 x 220 nm, 28.5 deg. orbit

Figure 3.3-11 ASRM-Based 50-80K Configuration Summary

50 k Vehicle, 3 Segment ASRM/LCSSME



Payload:	65,000 lbm (29.5 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,736,481 lbf

***First Stage*:**

Inert Mass:	179,947 lbm**
Usable Propellant:	1,214,401 lbm
Propellant Type:	HTPB
Engine Type/No.:	3 Segment ASRM/1
Diameter:	12.5 ft
Thrust/Weight:	1.740 g
Sea Level Thrust	3,020,812 lbf

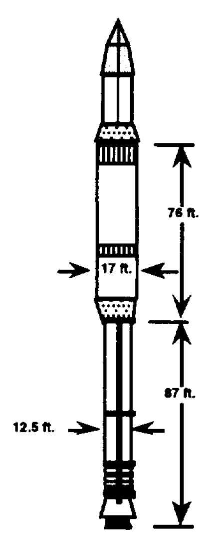
***Second Stage* :**

Inert Mass:	27,133 lbm
Usable Propellant:	250,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	LCSSME/1
Diameter:	17.0 ft*
Thrust/Weight:	0.955 g
Throttle Setting	100.0 %

Note:
* Diameter not optimized for best total vehicle L/D, inert weight, etc.; current diameter results in acceptable L/D
** First stage contains 42,300 lbm in excess of motor mass, this excess mass is used for extra booster stiffness and interstage masses

Figure 3.3-12 Three-Segment ASRM with Low-Cost SSME Upper Stage

50 k Vehicle, 2 Segment ASRM/J-2S



Payload:	43,600 lbm (19.8 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,293,267 lbf

***First Stage*:**

Inert Mass:	158,788 lbm**
Usable Propellant:	807,212 lbm
Propellant Type:	HTPB
Engine Type/No.:	2 Segment ASRM/1
Diameter:	12.5 ft
Thrust/Weight:	1.552 g
Sea Level Thrust	2,007,933 lbf***

***Second Stage* :**

Inert Mass:	23,667 lbm
Usable Propellant:	260,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	J-2S/1
Diameter:	12.5 ft *
Thrust/Weight:	0.998 g
Throttle Setting	100.0 %

Note:
* Diameter not optimized for best total vehicle L/D, inert weight, etc.; current diameter results in acceptable L/D
** First stage contains 40,500 lbm in excess of motor mass, this excess mass is used for extra booster stiffness and interstage masses
*** Thrust profile assumed to be a ratio of segment propellant loads

Figure 3.3-13 Two-Segment ASRM with J-2S Upper Stage

3.4 50-80K Vehicle Cost Assessments

In support of the Access to Space Option 2 team's assessment of ELV mixed-fleet architectures, ECON performed an assessment of the development and production costs of the 50-80K vehicle concepts that were defined by TA-2. Figure 3.4-1 summarizes the groundrules and assumptions used by ECON for their analysis. Figure 3.4-2 summarizes the development cost, in 1992 dollars, for each of the candidate configurations (by stage). Figure 3.4-3 summarizes the associated unit production costs for each of the candidate configurations (by stage); for the 101 vehicle flight sets and associated spares. Figure 3.4-4 tallies up the total program costs for each vehicle configuration, as shown by program phase.

The following conclusions were made as a result of the cost assessments. For similar 50K designs, the main propulsion system was the primary cost discriminator. As all main propulsion costs were provided by several differing sources, the commonalty of the associated groundrules and assumptions supporting the estimates are uncertain. Since the groundrules and assumptions of the primary cost discriminator (engine cost) may differ significantly, no direct comparisons were made between the estimates. This condition pointed out the need for consistent propulsion cost estimation methods. In general, solid and hybrid stages were cheaper than an equivalent liquid stage. Using a single large engine in place of multiple smaller ones resulted in lower stage unit cost.

Groundrules & Assumptions

- All costs presented in FY 92 $s
- NASA Code B new start escalation table used to normalize $s
- Current estimated costs include DDT&E and production
- ECON's weight-based cost estimating relationships utilized with various complexity factors
  -- Cost algorithms have been calibrated against MSFC Engineering Cost Group over the last several years
- Subsystem weights based on mass properties supplied by LMSC's vehicle sizing tools
- Weights included 10% contingency allocated to subsystems
- Mission model for 50K vehicle based on SSP/PLS mixed fleet model supplied by G. Austin/MSFC-PT01
  -- Total of 101 50K vehicle flights over 2003-2010 time horizon
- With exception of engines, all subsystems assumed 2 equivalent test articles
- All main propulsion cost data were throughputs for the estimate, no independent estimates of production costs were conducted
  -- Hybrid and ASRM data supplied by Aerojet
  -- SSME and LCSSME data supplied by Rocketdyne
  -- STME, F-1A, and J-2S data supplied by MSFC Engineering Cost Group
- No schedule impact assessed in costing
- State-of-the-art ranking assumed to be new drawings with known point-of-departure
  -- Engines assume most drawings exist
- Specification level set at manned space due to PLS Mission
- No government "wraps" included (40% typically used)

Figure 3.4-1 50-80K Vehicle Cost Assessment Groundrules and Assumptions

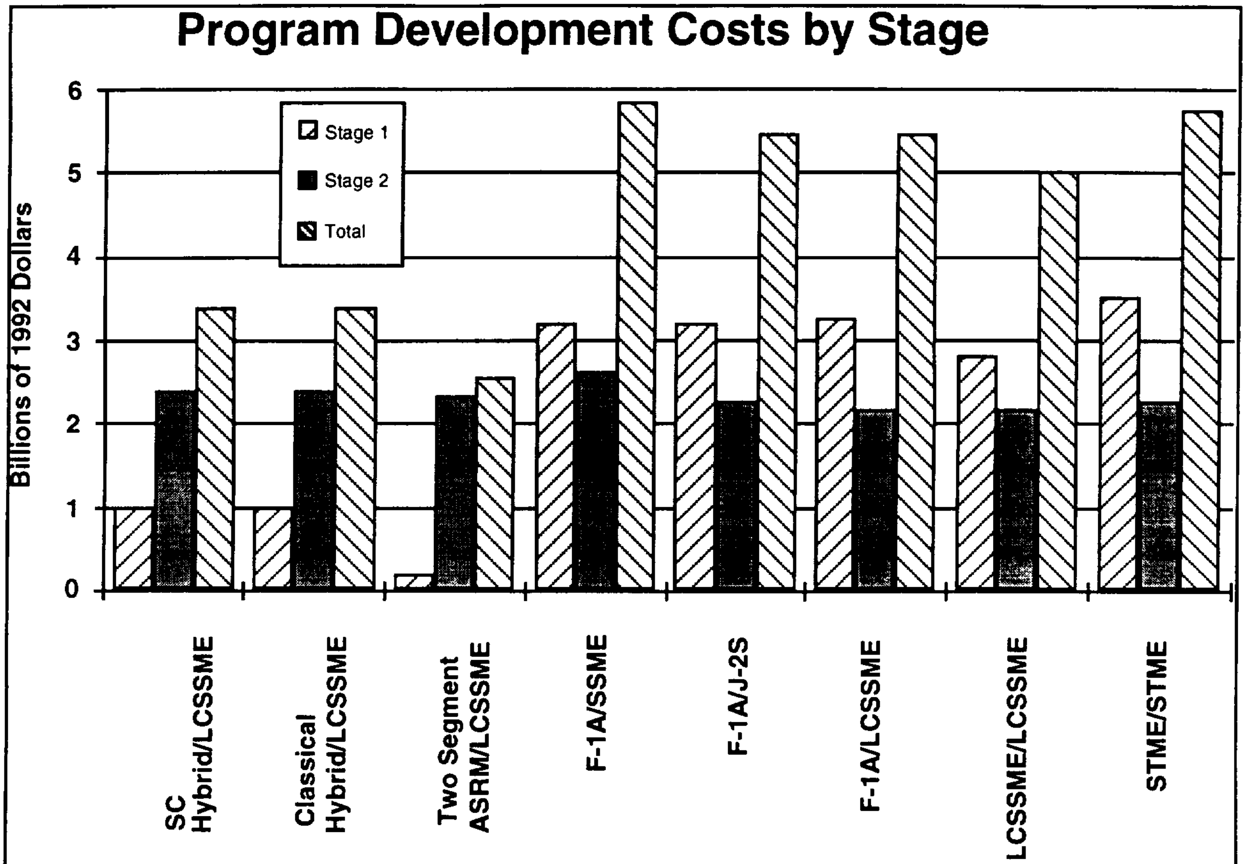


Figure 3.4-2 50-80K Vehicle Development Costs by Stage Element

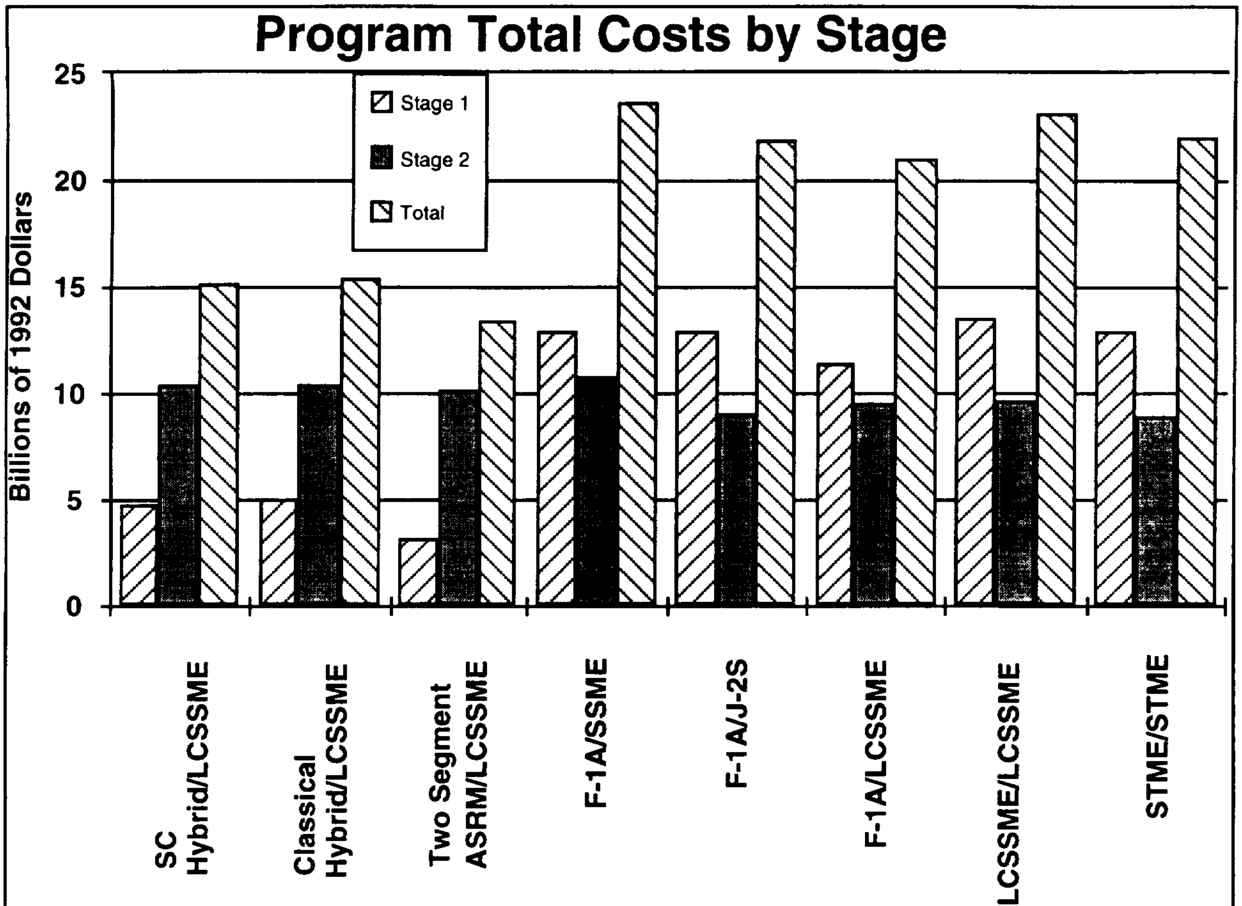


Figure 3.4-3 50-80K Vehicle Total Program Production Costs by Stage Element

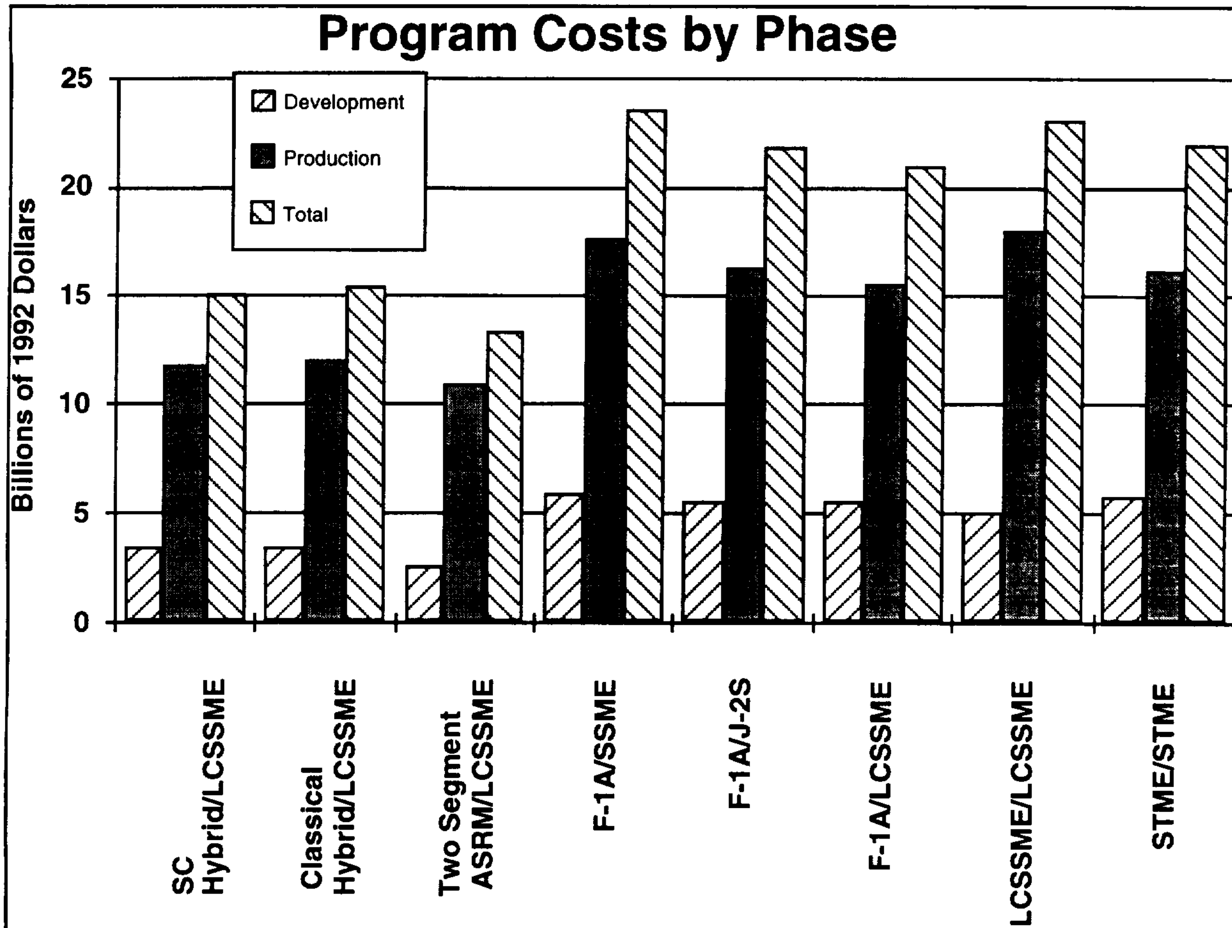


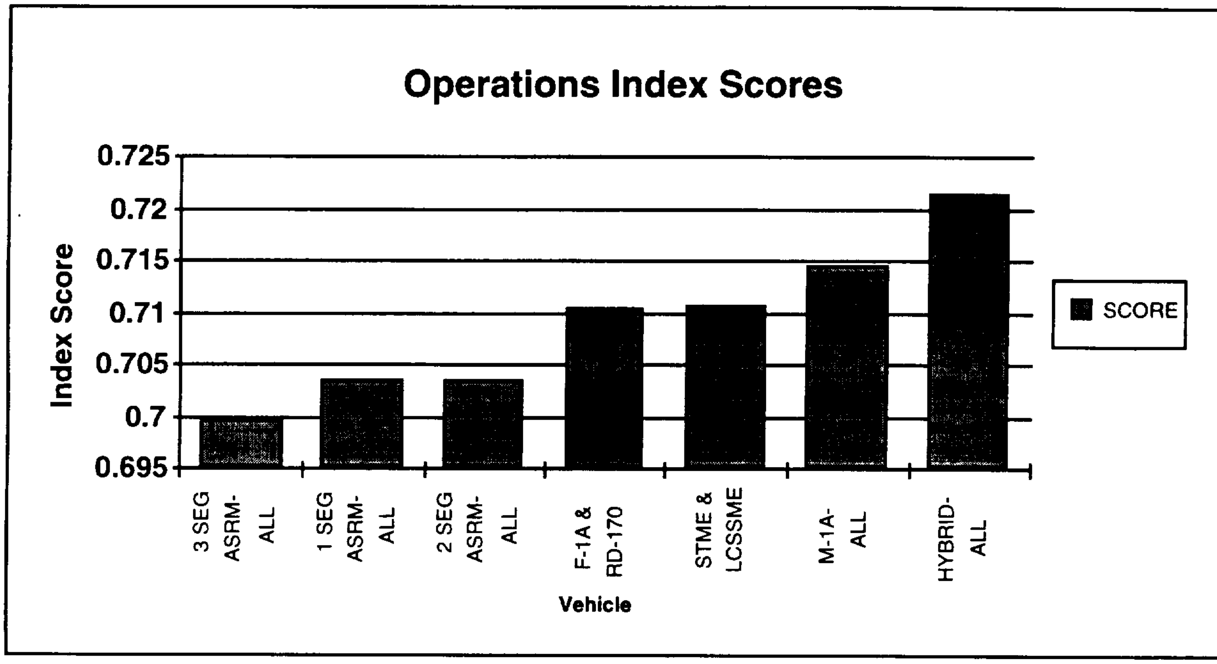
Figure 3.4-4 50-80K Vehicle Total Program Costs by Program Phase

3.5 50-80K Operability Assessments

During the definition and assessment of the candidate 50-80K vehicle configurations, LMSO was tasked to assess the relative operability of each candidate configuration. LMSO utilized a Ground Operations Index model to assess first-order ground operations figures of merit for the thirty-five 50/80K two-stage configurations generated by LMMS. The model was found to be most useful in providing a relative operability ranking among launch vehicle candidates when detailed configuration definition was limited or not available. Due to the relatively high degree of second stage commonalty between the thirty-five candidate configurations, the primary discriminator became the first and second stage engine selection (and inherent complexity).

The configuration figure-of-merit score was a weighted sum of the scores for a series of operability complexity factor utility parameters (number of stage elements, manned/ unmanned, processing concept, number of fluids, etc. Figure 3.5-1 illustrates that the segmented solid motor based (first stage) configurations scored slightly lower than the hydrocarbon based pump-fed (first stage) configurations due to the greater ground processing requirements of stacking the solid motor segments. The hybrid motor based configurations scored the highest in operability due to the simplicity of their monolithic-grain motors with only one turbopump (LOX pump). The use of two different propellant combinations between the first and second stages hurt the operability scores, while the use of similar propellants and simpler engine cycles (such as the LOX/LH2 M-1A large main engine) afforded higher operability scores. The operability scores were used as a first-order indication of relative differences in recurring costs between the various vehicle configurations.

Operability Assessment of 50/80K Vehicles



Note:
A higher score indicates better operability

NAME	SCORE
3 SEG ASRM-ALL	0.6997
1 SEG ASRM-ALL	0.7035
2 SEG ASRM-ALL	0.7035
F-1A & RD-170	0.7103
STME & LCSSME	0.7108
M-1A-ALL	0.7146
HYBRID-ALL	0.7214

Figure 3.5-1 50-80K Vehicle Operability Scores

4.0 Single Stage to Orbit Vehicle Assessments

The NASA-sponsored Access to Space teams completed their first round of vehicle assessments during the summer of 1993. The Access to Space Option 3 team, which assessed the requirements and viability of advanced technology based fully-reusable launch vehicle concepts as a replacement to the Space Shuttle fleet, had concluded that fully reusable single stage to orbit (SSTO) concepts were the most programmatically viable and held the potential for providing the lowest life cycle cost of all of the single and multi-stage reusable concepts that were assessed. Due to the limited amount of time and resources that the Option 3 team had to perform their vehicle concept assessments, they leveraged the extensive amount of concept-level data that had been developed by the Langley Research Center over a span of several years regarding vertical-takeoff/horizontal-landing winged SSTO configurations. As a result, the TA-2 team felt that an assessment of two other primary types of SSTO concepts, vertical-takeoff/vertical-landing (VTOL) side-entry configurations and vertical-takeoff/horizontal-landing (VTHL) lifting body configurations, would be a value-added contribution that the TA-2 team could make to help round out the Option 3 team's results. From June of 1993 to February of 1994, the TA-2 team developed complex SSTO vehicle sizing tools and defined and assessed VTOL and VTHL SSTO concepts over a large range of main engine and propellant combination options. A detailed account of the SSTO studies is provided in Sections 1 through 6 of Volume II of this final report. The conclusions of that study indicated that the lifting body VTHL concepts provided the largest range of design variability that could ensure a viable SSTO concept. The VTOL concepts were determined to be the highest in operational risk.

The SSTO concept assessments were the final launch vehicle assessment activities performed on the TA-2 contract. The remaining duties performed on the TA-2 contract were the completion of the Russian propulsion technology assessments, which are discussed in the next section.

5.0 Russian Propulsion Technology Assessments

Early in the HLLV configuration assessments, the NASA customer directed LMMS to obtain performance and technology data on candidate large Russian main engines. At that time, little factual information was available on actual Russian engine performance and no factual information was available regarding the technologies used by those engines. With the demise of the Soviet government and the formation of the Confederation of Independent States (CIS), Aerojet and Pratt & Whitney seized the opportunity to team with the major Russian and Ukrainian propulsion organizations with the intent of marketing CIS engines on U.S. launch vehicles. Aerojet signed teaming agreements with TRUD and CADB, and Pratt & Whitney signed a teaming agreement with NPO Energomash. Additional funds were provided to TA-2 to obtain performance data on the Aerojet team's existing CIS engines and to obtain both performance data and detailed technology data on the Pratt & Whitney team's CIS engines (both existing engines and their newly developing engines). Both Aerojet and Pratt & Whitney provided delegations of their respective CIS partners who visited MSFC for the preliminary exchange of technical information.

During the latter stages of the TA-2 contract, a large emphasis was placed by the NASA customer on the RD-170 and RD-180 bipropellant engines and the RD-701/704 tripropellant engine of the Pratt & Whitney team. Additionally, funding was provided to the TA-2 contract to obtain the results of preliminary hot-fire testing of a multi-element tripropellant injector that would be used on the RD-701/704. The RD-701 designation corresponded to the staged-combustion tripropellant main engine concept that Energomash had performed preliminary design definition of for a Russian SSTO project (that was canceled due to lack of funding). The RD-704 designation corresponded to a version of the RD-701 that was modified per NASA's preliminary requirements for an SSTO main propulsion concept. A summary of the tasks performed on Pratt & Whitney's subcontracted efforts to TA-2 are discussed in Section 14 of Volume II of this final report. The detailed results of those tasks were provided to LMMS as proprietary subcontract deliverables, which were in turn delivered to the NASA customer. Any inquiries regarding the data contained in those deliverables should be addressed to the TA-2 COTR, Mr. Gary W. Johnson, at the Marshall Space Flight Center. The results of the tripropellant injector hot-fire tests were the last contract data deliverables provided to the NASA customer on the TA-2 contract. Due to a series of test rig hardware failures during the injector testing, a time period of over two years elapsed from the start of test sponsorship by NASA, to the receipt of the test data results. A series of no-cost extensions from May of 1993 to September of 1995 were therefore required to the TA-2 contract in order to complete the contractual obligations.

6.0 Concluding Remarks

At the inception of the TA-2 contract, the hope of both the NASA customer and the LMMS team was to participate in a three-year effort to perform preliminary design and assessment (Pre-Phase A and Phase A) of extremely large, expendable launch vehicles that supported the requirements of a vigorous program for a human presence on the Moon and the eventual exploration of Mars. A very methodical and formal design process was identified by LMMS to perform those duties on the TA-2 contract. Unfortunately, NASA's space exploration program was entrained into a swirling vortex of wavering congressional and Executive Branch support and a projection of flat NASA budgets with no room for the growth of major new programs. The net result was to place both the TA-2 NASA customer and the TA-2 team on a roller-coaster ride of assessing a wide variety of launch vehicle concepts as NASA Headquarters searched for the definition of the Nation's future space transportation system requirements. The LMMS TA-2 team provided an extraordinary amount of technical data and high-quality engineering analysis effort, when considering the extenuating circumstances that the environment of that time period exerted.

It is hoped that the TA-2 analysis efforts documented in this final report will not have been a wasted effort, but will become a legacy for eventual application to NASA's future space transportation requirements.